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JAE 22/11/01 AERO-ENGINES - MANUFACTURER/CANADA/P&WC - PRATT & WHITNEY CANADA CORPORATION (Subsidiary of United Technologies Corporation)

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JAE 12/04/01 AERO-ENGINES - MANUFACTURER/CHINA/XRA - XIAN XR AERO ENGINE COMPONENTS CO LTD

JAE 12/04/01 AERO-ENGINES - MANUFACTURER/CHINA/XAE - XIAN AERO-ENGINE CORPORATION

JAE 12/04/01 *AERO-ENGINES - TURBOJET/CHINA/LMC WP7B

JAE 23/11/00 AERO-ENGINES - MANUFACTURER/CHINA/SARI - SHENYANG AVIATION ENGINE RESEARCH INSTITUTE

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JAE 31/03/98 AERO-ENGINES - MANUFACTURER/CHINA/CHINA NATIONAL AERO-TECHNOLOGY IMPORT AND EXPORT CORPORATION - CATIC

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JAE 22/03/02 AERO-ENGINES - MANUFACTURER/INTERNATIONAL/IAE - INTERNATIONAL AERO-ENGINES AG

JAE 22/03/02 *AERO-ENGINES - TURBOSHAFT/INTERNATIONAL/MTR 390

JAE 24/10/01 AERO-ENGINES - MANUFACTURER/INTERNATIONAL/EUROJET - EUROJET TURBO GmbH

JAE 18/09/01 *AERO-ENGINES - TURBOPROP/INTERNATIONAL/TP400

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JAE 04/07/01 AERO-ENGINES - MANUFACTURER/INTERNATIONAL/CFMI - CFM INTERNATIONAL SA

JAE 01/05/01 AERO-ENGINES - MANUFACTURER/INTERNATIONAL/ROLLS-ROYCE SNECMA

JAE 01/05/01 *AERO-ENGINES - TURBOFAN/INTERNATIONAL/SMR-95

JAE 01/05/01 AERO-ENGINES - MANUFACTURER/INTERNATIONAL/AEROSUD-MARVOL

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JAE 09/01/01 AERO-ENGINES - MANUFACTURER/JAPAN/IHI - ISHIKAWAJIMA-HARIMA JUKOGYO KABUSHIKI KAISHA (Ishikawajima-Harima Heavy Industries Co Ltd)

JAE 30/08/00 AERO-ENGINES - TURBOSHAFT/JAPAN/MITSUBISHI JUKOGYO KABUSHIKI KAISHA (Mitsubishi Heavy Industries Ltd)

JAE 30/08/00 AERO-ENGINES - MANUFACTURER/JAPAN/MITSUBISHI - MITSUBISHI JUKOGYO KABUSHIKI KAISHA (Mitsubishi Heavy Industries Ltd)

JAE 17/08/00 *AERO-ENGINES - TURBOJET/JAPAN/ISHIKAWAJIMA-HARIMA JUKOGYO KABUSHIKI KAISHA (Ishikawajima-Harima Heavy Industries Co Ltd)

JAE 17/12/99 AERO-ENGINES - TURBOFAN/JAPAN/IHI XF3-400

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JAE 17/12/99 *AERO-ENGINES - TURBOFAN/JAPAN/MITI/NAL FJR710

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JAE 01/05/01 *AERO-ENGINES - TURBOSHAFT/POLAND/PZL-10W

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/POLAND/TWD-10B

JAE 17/12/99 *AERO-ENGINES - TURBOSHAFT/POLAND/GTD-350

JAE 17/12/99 AERO-ENGINES - TURBOJET/POLAND/K-15

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JAE 31/03/98 *AERO-ENGINES - TURBOJET/POLAND/IL K-15

JAE 31/03/98 *AERO-ENGINES - TURBOFAN/POLAND/IL D-18A

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JAE 22/11/01 *AERO-ENGINES - TURBOFAN/RUSSIA/D-100

JAE 22/11/01 *AERO-ENGINES - TURBOFAN/RUSSIA/D-30KU-90

JAE 22/11/01 *AERO-ENGINES - TURBOFAN/RUSSIA/D-30KU

JAE 22/11/01 *AERO-ENGINES - TURBOFAN/RUSSIA/D-30

JAE 22/11/01 *AERO-ENGINES - TURBOSHAFT/RUSSIA/D-25V

JAE 22/11/01 *AERO-ENGINES - TURBOFAN/RUSSIA/D-21A1

JAE 22/11/01 AERO-ENGINES - MANUFACTURER/RUSSIA/`AVIADVIGATEL' - `AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

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JAE 24/10/01 *AERO-ENGINES - TURBOFAN/RUSSIA/AL-37FU

JAE 04/07/01 AERO-ENGINES - MANUFACTURER/RUSSIA/SOYUZ - TMKB (TUSHINSKOYE {TUSHINO} ENGINE DESIGN BUREAU) `SOYUZ'

JAE 04/07/01 *AERO-ENGINES - TURBOSHAFT/RUSSIA/VK-2500

JAE 04/07/01 AERO-ENGINES - MANUFACTURER/RUSSIA/SOYUZ - AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

JAE 04/07/01 AERO-ENGINES - MANUFACTURER/RUSSIA/RYBINSK MOTORS - RYBINSK MOTORS JSC

JAE 04/07/01 AERO-ENGINES - MANUFACTURER/RUSSIA/KMPO - KAZAN MOTOR-BUILDING PRODUCTION ASSOCIATION JSC

JAE 04/07/01 AERO-ENGINES - MANUFACTURER/RUSSIA/KLIMOV CORPORATION - ST PETERSBURG NPO IM KLIMOV

JAE 04/07/01 *AERO-ENGINES - TURBOFAN/RUSSIA/AL-55

JAE 04/07/01 *AERO-ENGINES - TURBOPROP/RUSSIA/AL-34

JAE 04/07/01 AERO-ENGINES - MANUFACTURER/RUSSIA/LYUL'KA SATURN - LYUL'KA SATURN INC

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JAE 01/05/01 *AERO-ENGINES - TURBOJET/RUSSIA/RD-36-41

JAE 01/05/01 AERO-ENGINES - TURBOFAN/RUSSIA/RD-43, VKS

JAE 01/05/01 AERO-ENGINES - TURBOJET/RUSSIA/R28V

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/RUSSIA/TVD-10

JAE 12/04/01 AERO-ENGINES - MANUFACTURER/RUSSIA/MOTOR - MOTOR, GNPP (STATE SCIENTIFIC PRODUCTION ENTERPRISE)

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/RUSSIA/GRANIT TVD-150

JAE 12/04/01 *AERO-ENGINES - TURBOSHAFT/RUSSIA/GTD-3

JAE 12/04/01 AERO-ENGINES - TURBOSHAFT/RUSSIA/TV7-117V (VK-3000)

JAE 12/04/01 AERO-ENGINES - MANUFACTURER/RUSSIA/TUMENSKIE - OAO (JOINT STOCK COMPANY) TUMENSKIE MOTORSTROITELY

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/RD-36-35

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JAE 12/04/01 *AERO-ENGINES - TURBOPROP/RUSSIA/NK-110

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/RUSSIA/NK-56, NK-64

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/RUSSIA/NK-25

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/RUSSIA/NK-62

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/RUSSIA/NK-86, NK-87

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/R95

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/RD-41

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/RUSSIA/VK-1500

JAE 12/04/01 AERO-ENGINES - TURBOFAN/RUSSIA/NK-118

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/RUSSIA/TV128-300

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JAE 12/04/01 *AERO-ENGINES - TURBOFAN/RUSSIA/R79

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/R29-300

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/R27V-300

JAE 12/04/01 AERO-ENGINES - TURBOJET/RUSSIA/R27-300

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/R15-300

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/R11

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/RD-9

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/AM-3, RD-3M

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/TURBOSHAFT/RUSSIA/TVD-1500 (RD-600)

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/RU-19

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JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/RD-38

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/RD-36-51

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/RD-7, VD-7

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/RUSSIA/AL-31

JAE 12/04/01 AERO-ENGINES - TURBOFAN/RUSSIA/R127-300

JAE 12/04/01 AERO-ENGINES - TURBOFAN/RUSSIA/R126-300

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/RUSSIA/R123-300

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/RUSSIA/RD-1700

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/RUSSIA/TVD-20

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/RUSSIA/NK-22

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JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/R35-300

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/R25

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/R195

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/R13

JAE 12/04/01 AERO-ENGINES - MANUFACTURER/RUSSIA/OMSK BARANOV - MOTOR-BUILDING ENTERPRISE NAMED FOR P I BARANOV

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/TURBOSHAFT/RUSSIA/TV-O-100

JAE 12/04/01 *AERO-ENGINES - TURBOJET/RUSSIA/TRDD-50

JAE 12/04/01 AERO-ENGINES - MANUFACTURER/RUSSIA/OMKB (OEDB) - OMSK ENGINE DESIGN BUREAU

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/RUSSIA/NK-321

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/RUSSIA/NK-12

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JAE 12/04/01 *AERO-ENGINES - TURBOFAN/RUSSIA/NK-88, NK-89

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JAE 12/04/01 AERO-ENGINES - MANUFACTURER/RUSSIA/GRANIT - MACHINE-BUILDING DESIGN BUREAU 'GRANIT'

JAE 09/01/01 *AERO-ENGINES - TURBOPROP/RUSSIA/TV7-117S Series 2

JAE 09/01/01 AERO-ENGINES - MANUFACTURER/RUSSIA/ROSVOOROUZHENIYE - STATE CORPORATION FOR IMPORT/EXPORT OF ARMS

JAE 09/01/01 AERO-ENGINES - MANUFACTURER/RUSSIA/CIAM - CENTRAL INSTITUTE OF AVIATION MOTORS

JAE 09/01/01 AERO-ENGINES - MANUFACTURER/RUSSIA/ASSAD - ASSOCIATION OF AERO-ENGINE MANUFACTURERS

JAE 09/01/01 *AERO-ENGINES - TURBOFAN/RUSSIA/NK-8

JAE 23/11/00 *AERO-ENGINES - TURBOJET/RUSSIA/RYBINSK MOTORS JSC

JAE 23/11/00 *AERO-ENGINES - TURBOFAN/RUSSIA/RYBINSK MOTORS JSC

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JAE 23/11/00 *AERO-ENGINES - TURBOFAN/RUSSIA/RYBINSK MOTORS JSC

JAE 23/11/00 AERO-ENGINES - MANUFACTURER/RUSSIA/NK - ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

JAE 23/11/00 AERO-ENGINES - MANUFACTURER/RUSSIA/CHERNYSHEV - V V CHERNYSHEV STATE MOSCOW MECHANICAL ENGINEERING PRODUCTION ENTERPRISE

JAE 30/08/00 *AERO-ENGINES - TURBOSHAFT/RUSSIA/MACHINE-BUILDING DESIGN BUREAU 'GRANIT'

JAE 30/08/00 *AERO-ENGINES - TURBOJET/RUSSIA/MACHINE-BUILDING DESIGN BUREAU 'GRANIT'

JAE 30/08/00 *AERO-ENGINES - TURBOPROP/RUSSIA/ST PETERSBURG NPO IM KLIMOV

JAE 30/08/00 *AERO-ENGINES - TURBOSHAFT/RUSSIA/ST PETERSBURG NPO IM KLIMOV

JAE 30/08/00 *AERO-ENGINES - TURBOFAN/RUSSIA/ST PETERSBURG NPO IM KLIMOV

JAE 17/08/00 AERO-ENGINES - MANUFACTURER/RUSSIA/SKBM - SAMARA MACHINE-BUILDING DESIGN BUREAU

JAE 17/08/00 AERO-ENGINES - MANUFACTURER/RUSSIA/MOTOROSTROITEL - MOTOROSTROITEL JSC

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JAE 17/12/99 AERO-ENGINES - TURBOFAN/RUSSIA/RD-35

JAE 17/12/99 AERO-ENGINES - MANUFACTURER/RUSSIA/MANUFACTURER DETAILS

JAE 17/12/99 AERO-ENGINES - MANUFACTURER/RUSSIA/MANUFACTURER DETAILS

JAE 26/10/99 *AERO-ENGINES - TURBOJET/RUSSIA/VK-1

JAE 17/09/99 *AERO-ENGINES - TURBOPROP/TURBOSHAFT/RUSSIA/AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'/TV-116-300

JAE 17/09/99 AERO-ENGINES - TURBOSHAFT/RUSSIA/AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'/TV-O-100-300

JAE 17/09/99 AERO-ENGINES - MANUFACTURER/RUSSIA/SMPO - SAMARA MOTOR PRODUCTION ORGANISATION/MANUFACTURER DETAILS

JAE 17/09/99 *AERO-ENGINES - TURBOFAN/RUSSIA/KMPO - KAZAN MOTOR-BUILDING PRODUCTION/AL-35

JAE 17/09/99 AERO-ENGINES - TURBOPROP/RUSSIA/KLIMOV CORPORATION - ST PETERSBURG NPO IM KLIMOV/PK206

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JAE 17/09/99 AERO-ENGINES - TURBOPROP/RUSSIA/KLIMOV CORPORATION - ST PETERSBURG NPO IM KLIMOV/PK6A

JAE 17/09/99 AERO-ENGINES - MANUFACTURER/RUSSIA/V/O AVIAEXPORT plc/MANUFACTURER DETAILS

JAE 17/09/99 AERO-ENGINES/RUSSIA/Introduction

JAE 16/09/98 AERO-ENGINES - TURBOPROP/RUSSIA/TVD-450

JAE 16/09/98 AERO-ENGINES - TURBOSHAFT/RUSSIA/GTE-400

JAE 31/03/98 AERO-ENGINES - TURBOSHAFT/TURBOPROP/RUSSIA/TVD-10B

JAE 31/03/98 AERO-ENGINES - TURBOFAN/RUSSIA/AL-32

JAE 31/03/98 *AERO-ENGINES - TURBOJET/RUSSIA/AL-21

JAE 31/03/98 *AERO-ENGINES - TURBOJET/RUSSIA/AL-7

JAE 31/03/98 AERO-ENGINES - TURBOPROP/RUSSIA/TVA-3000

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JAE 22/03/02 *AERO-ENGINES - TURBOFAN/SLOVAKIA/DV-2

JAE 22/03/02 AERO-ENGINES - MANUFACTURER/SLOVAKIA/PSLM - POVAZSKÉ STROJÁRNE LETECKÉ MOTORY, a.s.

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JAE 31/03/98 AERO-ENGINES - MANUFACTURER/SOUTH AFRICA/ATLAS AVIATION - ATLAS

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JAE 30/08/00 AERO-ENGINES - MANUFACTURER/SPAIN/ITP - INDUSTRIA DE TURBO PROPULSORES SA

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JAE 22/03/02 *AERO-ENGINES - TURBOFAN/SWEDEN/RM8

JAE 22/03/02 AERO-ENGINES - MANUFACTURER/SWEDEN/VOLVO - VOLVO AERO CORPORATION

JAE 11/01/02 *AERO-ENGINES - TURBOFAN/SWEDEN/RM12

JAE 17/08/00 AERO-ENGINES - TURBOFAN/SWEDEN/VOLVO AERO CORPORATION

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JAE 18/04/02 *AERO-ENGINES - TURBOFAN/UKRAINE/AI-222

JAE 22/03/02 *AERO-ENGINES - TURBOSHAFT/UKRAINE/AI-450

JAE 04/07/01 AERO-ENGINES - TURBOFAN/UKRAINE/AI-V

JAE 04/07/01 *AERO-ENGINES - TURBOFAN/UKRAINE/AI-22

JAE 04/07/01 AERO-ENGINES - MANUFACTURER/UKRAINE/IVCHENKO PROGRESS ZMKB - IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

JAE 12/04/01 AERO-ENGINES - MANUFACTURER/UKRAINE/MOTOR SICH - jsc MOTOR SICH

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/UKRAINE/D-27

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/UKRAINE/D-436

JAE 12/04/01 AERO-ENGINES - TURBOPROP/UKRAINE/AI-30

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JAE 12/04/01 *AERO-ENGINES - PROPFAN/UKRAINE/D-236

JAE 12/04/01 *AERO-ENGINES - TURBOSHAFT/UKRAINE/D-136

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/UKRAINE/D-36

JAE 12/04/01 *AERO-ENGINES - TURBOFAN/UKRAINE/D-18T

JAE 12/04/01 *AERO-ENGINES - TURBOPROP/UKRAINE/AI-24

JAE 30/08/00 *AERO-ENGINES - TURBOFAN/UKRAINE/IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

JAE 30/08/00 AERO-ENGINES - TURBOFAN/UKRAINE/IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

JAE 17/09/99 AERO-ENGINES - MANUFACTURER/UKRAINE/ZAPOROZHYE MOTOR-BUILDING COMPLEX/MANUFACTURER DETAILS

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JAE 31/03/98 *AERO-ENGINES - TURBOSHAFT/UKRAINE/D-127

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JAE 11/01/02 *AERO-ENGINES - TURBOFAN/UK/ROLLS-ROYCE 535

JAE 11/01/02 AERO-ENGINES - TURBOFAN/UK/V2500

JAE 11/01/02 AERO-ENGINES - TURBOFAN/UK/TF41

JAE 11/01/02 AERO-ENGINES - TURBOSHAFT/UK/RTM322

JAE 11/01/02 AERO-ENGINES - TURBOFAN/UK/RB199

JAE 11/01/02 AERO-ENGINES - TURBOJET/UK/Olympus

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JAE 11/01/02 AERO-ENGINES - TURBOFAN/UK/JSF

JAE 11/01/02 AERO-ENGINES - TURBOFAN/UK/EUROJET TURBO EJ200

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JAE 11/01/02 AERO-ENGINES - TURBOFAN/UK/BR700 series

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JAE 11/01/02 AERO-ENGINES - MANUFACTURER/UK/ROLLS-ROYCE - ROLLS-ROYCE plc

JAE 11/01/02 *AERO-ENGINES - TURBOJET/UK/ROLLS-ROYCE VIPER

JAE 11/01/02 *AERO-ENGINES - TURBOPROP/UK/ROLLS-ROYCE TYNE

JAE 11/01/02 *AERO-ENGINES - TURBOFAN/UK/ROLLS-ROYCE TAY

JAE 11/01/02 *AERO-ENGINES - TURBOFAN/UK/ROLLS-ROYCE SPEY

JAE 11/01/02 *AERO-ENGINES - TURBOFAN/UK/ROLLS-ROYCE RB211

JAE 11/01/02 *AERO-ENGINES - TURBOFAN/UK/ROLLS-ROYCE PEGASUS

JAE 11/01/02 *AERO-ENGINES - TURBOFAN/UK/ROLLS-ROYCE CONWAY

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JAE 11/01/02 *AERO-ENGINES - TURBOJET/UK/ROLLS-ROYCE NENE

JAE 11/01/02 *AERO-ENGINES - TURBOSHAFT/UK/ROLLS-ROYCE GNOME

JAE 11/01/02 *AERO-ENGINES - TURBOSHAFT/UK/ROLLS-ROYCE GEM

JAE 11/01/02 *AERO-ENGINES - TURBOSHAFT/UK/ROLLS-ROYCE GAZELLE

JAE 11/01/02 *AERO-ENGINES - TURBOJET/UK/ROLLS-ROYCE DERWENT

JAE 11/01/02 *AERO-ENGINES - TURBOPROP/UK/ROLLS-ROYCE DART

JAE 11/01/02 *AERO-ENGINES - TURBOJET/UK/ROLLS-ROYCE AVON (MILITARY)

JAE 11/01/02 *AERO-ENGINES - TURBOJET/UK/ROLLS-ROYCE AVON (CIVIL)

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JAE 22/03/02 AERO-ENGINES - MANUFACTURER/USA/TELEDYNE CONTINENTAL MOTORS - TCM TURBINE ENGINES

JAE 22/03/02 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC CF6-80C2

JAE 22/03/02 *AERO-ENGINES - TURBOPROP/USA/HONEYWELL TPE331

JAE 22/03/02 *AERO-ENGINES - TURBOSHAFT/USA/HONEYWELL LTC4, T55

JAE 22/03/02 *AERO-ENGINES - TURBOFAN/USA/HONEYWELL AS900

JAE 22/03/02 *AERO-ENGINES - TURBOFAN/USA/HONEYWELL TFE731

JAE 22/03/02 *AERO-ENGINES - TURBOPROP/ - TURBOSHAFT/USA/HONEYWELL LTS101 AND LTP101

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JAE 22/03/02 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC CF6-80E1

JAE 22/03/02 *AERO-ENGINES - TURBOFAN/USA/CFE738

JAE 22/03/02 *AERO-ENGINES - TURBOPROP/USA/SOLOY DUAL PAC

JAE 22/03/02 *AERO-ENGINES - TURBOPROP/USA/SOLOY TURBINE PAC

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JAE 11/01/02 *AERO-ENGINES - TURBOFAN/USA/ROLLS-ROYCE AE 3007

JAE 11/01/02 *AERO-ENGINES - TURBOPROP/USA/ROLLS-ROYCE AE 2100

JAE 11/01/02 AERO-ENGINES - MANUFACTURER/USA/ROLLS-ROYCE - ROLLS-ROYCE CORPORATION

JAE 11/01/02 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC GE90

JAE 11/01/02 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC F414

JAE 11/01/02 *AERO-ENGINES - TURBOPROP/ - TURBOSHAFT/USA/ROLLS-ROYCE MODEL 250

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JAE 11/01/02 *AERO-ENGINES - TURBOPROP/USA/ROLLS-ROYCE 501

JAE 24/10/01 *AERO-ENGINES - TURBOJET/USA/TURBINE ENGINES J69

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JAE 23/10/01 *AERO-ENGINES - TURBOFAN/USA/PRATT & WHITNEY PW4000

JAE 18/09/01 AERO-ENGINES - MANUFACTURER/USA/USAF - UNITED STATES AIR FORCE

JAE 18/09/01 *AERO-ENGINES - TURBOPROP/ - TURBOSHAFT/USA/LHTEC T800

JAE 18/09/01 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC TF34 and CF34

JAE 04/07/01 AERO-ENGINES - MANUFACTURER/USA/GENERAL ELECTRIC - GE AIRCRAFT ENGINES

JAE 04/07/01 AERO-ENGINES - MANUFACTURER/USA/ENGINE ALLIANCE - ENGINE ALLIANCE, A JOINT COMPANY OF GEAE AND P&W

JAE 04/07/01 *AERO-ENGINES - TURBOFAN/USA/WILLIAMS ROLLS FJ44

JAE 04/07/01 *AERO-ENGINES - TURBOFAN/USA/PRATT & WHITNEY F119

JAE 04/07/01 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC F110

JAE 04/07/01 *AERO-ENGINES - TURBOFAN/USA/ENGINE ALLIANCE GP7000

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JAE 04/07/01 AERO-ENGINES - MANUFACTURER/USA/WILLIAMS - WILLIAMS INTERNATIONAL

JAE 04/07/01 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC CF6

JAE 01/05/01 AERO-ENGINES - MANUFACTURER/USA/GE/P&W - GE AIRCRAFT ENGINES

JAE 09/01/01 AERO-ENGINES - TURBOFAN/TURBOJET/USA/P&W/TCM

JAE 09/01/01 *AERO-ENGINES - TURBOPROP/TURBOSHAFT/USA/GENERAL ELECTRIC CT7

JAE 09/01/01 *AERO-ENGINES - TURBOFAN/USA/WILLIAMS INTERNATIONAL FJX

JAE 09/01/01 *AERO-ENGINES - TURBOJET/USA/GENERAL ELECTRIC YF120

JAE 23/11/00 *AERO-ENGINES - TURBOFAN/USA/GE AIRCRAFT ENGINES

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JAE 30/08/00 *AERO-ENGINES - TURBOFAN/USA/UNITED TECHNOLOGIES PRATT & WHITNEY

JAE 30/08/00 *AERO-ENGINES - TURBOFAN/USA/UNITED TECHNOLOGIES PRATT & WHITNEY

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JAE 30/08/00 *AERO-ENGINES - TURBOFAN/USA/UNITED TECHNOLOGIES PRATT & WHITNEY

JAE 30/08/00 *AERO-ENGINES - TURBOFAN/USA/UNITED TECHNOLOGIES PRATT & WHITNEY

JAE 17/08/00 *AERO-ENGINES - TURBOSHAFT/USA/GENERAL ELECTRIC T700

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JAE 17/12/99 *AERO-ENGINES - TURBOFAN/USA/WILLIAMS INTERNATIONAL FJ33

JAE 17/12/99 AERO-ENGINES - TURBOSHAFT/USA/WILLIAMS INTERNATIONAL TSX

JAE 26/10/99 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC F404

JAE 17/09/99 AERO-ENGINES - TURBOFAN/USA/UNITED TECHNOLOGIES PRATT & WHITNEY/PRATT & WHITNEY PW7000

JAE 17/09/99 *AERO-ENGINES - TURBOFAN/USA/UNITED TECHNOLOGIES PRATT & WHITNEY/PRATT & WHITNEY JTF22, F100

JAE 17/09/99 *AERO-ENGINES - TURBOFAN/USA/UNITED TECHNOLOGIES PRATT & WHITNEY/PRATT & WHITNEY JT3D

JAE 17/09/99 AERO-ENGINES - MANUFACTURER/USA/UNITED TECHNOLOGIES PRATT & WHITNEY/International participation

JAE 17/09/99 AERO-ENGINES - TURBOFAN/USA/UNITED TECHNOLOGIES PRATT & WHITNEY/V2500

JAE 17/09/99 AERO-ENGINES - TURBOFAN/USA/UNITED TECHNOLOGIES PRATT & WHITNEY/JSF

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JAE 17/09/99 AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC - GE AIRCRAFT ENGINES/CFMI CFM56

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JAE 19/03/99 AERO-ENGINES/USA/TF33

JAE 19/03/99 AERO-ENGINES/USA/TF30

JAE 19/03/99 AERO-ENGINES/USA/T73

JAE 19/03/99 *AERO-ENGINES - TURBOFAN/USA/PRATT & WHITNEY JTF10A, TF30

JAE 19/03/99 *AERO-ENGINES - TURBOJET/USA/PRATT & WHITNEY JT12

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JAE 19/03/99 AERO-ENGINES/USA/F100

JAE 19/03/99 AERO-ENGINES/USA/F120

JAE 19/03/99 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC F118

JAE 19/03/99 AERO-ENGINES/USA/CT58

JAE 16/09/98 AERO-ENGINES/USA/CF34

JAE 16/09/98 AERO-ENGINES - MANUFACTURER/USA/ALLISON ENGINE COMPANY

JAE 31/03/98 *AERO-ENGINES - TURBOJET/USA/WRIGHT J65

JAE 31/03/98 AERO-ENGINES - MANUFACTURER/USA/WRIGHT AERONAUTICAL - WRIGHT

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JAE 31/03/98 *AERO-ENGINES - TURBOJET/USA/PRATT & WHITNEY JT11D-20

JAE 31/03/98 *AERO-ENGINES - TURBOJET/USA/PRATT & WHITNEY JT8B

JAE 31/03/98 *AERO-ENGINES - TURBOJET/USA/PRATT & WHITNEY JT4A

JAE 31/03/98 *AERO-ENGINES - TURBOJET/USA/PRATT & WHITNEY JT3

JAE 31/03/98 *AERO-ENGINES - TURBOSHAFT/USA/PRATT & WHITNEY JFTD12

JAE 31/03/98 AERO-ENGINES - TURBOJET/USA/J75

JAE 31/03/98 AERO-ENGINES - TURBOJET/USA/J60

JAE 31/03/98 AERO-ENGINES - TURBOJET/USA/J58

JAE 31/03/98 AERO-ENGINES - TURBOJET/USA/J57

JAE 31/03/98 AERO-ENGINES - TURBOJET/USA/J52

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JAE 31/03/98 *AERO-ENGINES - TURBOSHAFT/USA/GENERAL ELECTRIC CT58

JAE 31/03/98 *AERO-ENGINES - TURBOSHAFT/USA/GENERAL ELECTRIC T58

JAE 31/03/98 *AERO-ENGINES - TURBOJET/USA/GENERAL ELECTRIC J85

JAE 31/03/98 *AERO-ENGINES - TURBOJET/USA/GENERAL ELECTRIC J79

JAE 31/03/98 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC CJ805-23

JAE 31/03/98 *AERO-ENGINES - TURBOJET/USA/GENERAL ELECTRIC CJ610

JAE 31/03/98 *AERO-ENGINES - TURBOFAN/USA/GENERAL ELECTRIC CF700

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AERO-ENGINES - MANUFACTURER, BELGIUM

Date Posted: 04 July 2001

Jane's Aero-Engines 10

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General Manager, Commerce and Programmes: Jean-Christophe Dalla Toffola
Manager, External Communications: Pierre Vierset
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e-mail: pvierset@techspace-aero.be

TA designs, produces and supports equipment for aircraft and spacecraft propulsion. It began producing jet engines in 1949 (when its name was FN). It mass-produced the Pratt & Whitney F100 for European F-16s, and supports these engines for 12 air forces. Other products include LP compressors, bearing compartments, discs, casings, turbine blades and vanes, engine lubrication systems and major parts for spacecraft rocket engines. It also provides turnkey tailor-made jet-engine test facilities.

Customers include SNECMA (CFM56 and GE90), Pratt & Whitney (F100, PW4000 and PW6000), GE (CF34-10 and F110), Honeywell (AS900 and ALF502/507) and IAE (V2500). Shareholders are

SNECMA (51 per cent), Walloon Region (30 per cent) and Pratt & Whitney (19 per cent). The workforce numbers 1,230 and sales in 1998 were Euro200 million (US$240 million).

UPDATED

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AERO-ENGINES - TURBOSHAFT, CANADA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

PRATT & WHITNEY CANADA

(Subsidiary of United Technologies Corporation)

P&WC PW200

The launch of a completely new turboshaft engine for helicopters was announced by P&WC in 1983. The basic design was planned to be simple and robust, adaptable to almost any application, and to provide for considerable power growth. Thus, while the initial model was rated in the 478 kW (600 hp) class, it was planned from the outset that PW200 models would cover the power range 373 to 671 kW (500 to 900 shp). Models currently in production are:

PW206A/E

Initial version, selected in 1989 to power the Boeing-McDonnell Douglas Explorer 900/902. Certificated in December 1991 at a T-O rating of 463 kW (621 shp), but recertificated at higher powers (see below) in November 1993. Helicoper deliveries began in late 1994, and service operation began in June 1995.

PW206B

Configured with different gearbox and angled output for the Eurocopter EC135. Selected in 1991, certificated by Transport Canada in February 1996, first helicoper delivery August 1996.

PW206C

Very similar to PW206A, but configured for Agusta A109 POWER. Certificated by Transport Canada on 21 December 1995 and first production engines shipped in January 1996. Also scheduled for production by P&WC (Rus) as RK206S (see under Klimov, Russia). In August 1999 two XRK206S engines powered the first prototype KVZ Ansat, the production of which is planned to have PW207 engines.

PW206D

Generally similar to previous versions but with slightly different configuration. Selected June 1996 for Bell 427 and also for Kamov Ka-115. Certificated 1998.
Certification scheduled for late 1997.

PW206E

Increased OEI ratings for MD Explorer. Certificated August 1997.

PW207D

Growth version of PW206D. T-O rating 529.5 kW (710 shp). Certificated November 1998 for Bell 427. The proposed RK207 is envisaged as the engine of the production KVZ Ansat.

A number of other versions of the PW200 family are being discussed with manufacturers of high-performance single-engine helicopters and intermediate twins. Remarkably, in 1998 all models were cleared to operate at significantly increased levels of power. The basic core engine has the potential to reach 746 kW (1,000 shp).

All versions have a TBO starting at 3,000 hours, rising later to 3,500, with no scheduled hot-section inspection. All are covered by the Total Customer Support package, with warranty for 3 years or 2,500 hours.

By May 2000 PW2000 engines powered 178 helicopters in 30 countries. They had then accumulated over 208,000 hours of operating time.

Type

Free-turbine turboshaft.

Intake

Amidships, between gearbox and combustion chamber, radially inwards through a mesh screen. Service line buried in intake struts.

Compressor

Single-stage centrifugal in titanium alloy. Pressure ratio 8.0.

Combustion Chamber

Folded annular reverse-flow, fed through multiple curved pipes from compressor diffuser. Around the casing, 12 air-blast fuel nozzles project radially inwards. Two igniters project diagonally inwards at the

far end, around the LP turbine. The flame tube curves inwards in a 180~ bend to meet the HP turbine nozzles.

Compressor Turbine

Small-diameter high-speed single stage, with blades dovetailed into disc. Cold-junction thermocouple to sense gas temperatures.

Power Turbine

Single stage, with separate blades dovetailed into disc.

Output

PW206A and 206C, front-mounted combined reduction and accessory spur gearbox, with 6,000 rpm output, in aluminium case incorporating the oil tank; PW206B, bevel gearbox giving angled outpt at 5,898 rpm, again with integral oil tank with sight glass on left side.

Accessories

Gearbox pads provide for a starter/generator, alternator, tachometer generator and hydraulic pump.

Starting

Usually electric, by starter/generator. Dual capacitor-discharge igniters.

Control System

FADEC, with a dedicated permanent-magnet alternator and manual back-up. Includes a phase-shift output torquemeter, speed sensors in both shafts and a gas-temperature measurement system.

Fuel Specification

JP-1, JP-4, JP-5 or a range of gasolines.

Dimensions

Length:	
PW206A, 206C, 206D	912 mm (35.9 in)
PW206B	1,042 mm (41.0 in)
Width (all)	500 mm (19.7 in)
Height:	
PW206A, 206C, 206D	566 mm (22.3 in)
PW206B	627 mm (24.7 in)

Weight, Dry

PW206A, 206C	107.5 kg (237 lb)
PW206B	112 kg (247 lb)
PW206D	110.1 kg (242.7 lb)

Performance Ratings

30 sec OEI:	
PW207D	611.5 kW (820 shp)
2.5 min:	
PW206A	514 kW (690 shp)
PW206B, 206C, 206D	545 kW (732 shp)
PW206E	504 kW (676 shp)
PW207D	596.6 kW (800 shp)
T-O and OEI:	
PW206A, 206C, 206D	477 kW (640 shp)
PW206B	500 kW (700 shp)
PW206E	482 kW (646 shp)
PW207D	529.5 kW (710 shp)
Continuous:	
PW206A, 206D	410 kW (550 shp)
PW206B	419 kW (562 shp)
PW206C	423 kW (567 shp)
PW206E	426 kW (572 shp)
PW207D	466 kW (625 shp)

Specific Fuel Consumption

T-O as above:	
PW206A	91.7 Ág/J (0.543 lb/h/shp)
PW206B	92.6 Ág/J (0.548 lb/h/shp)
PW206C	91.6 Ág/J (0.542 lb/h/shp)
PW206D	90.6 Ág/J (0.536 lb/h/shp)
Continuous:	
PW206A	94.6 Ág/J (0.560 lb/h/shp)
PW206B	95.7 Ág/J (0.566 lb/h/shp)
PW206C	94.0 Ág/J (0.556 lb/h/shp)
PW206D	93.0 Ág/J (0.550 lb/h/shp)

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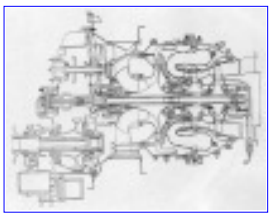
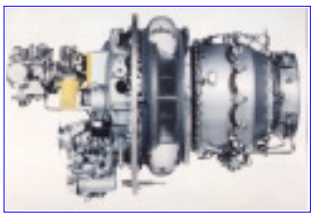
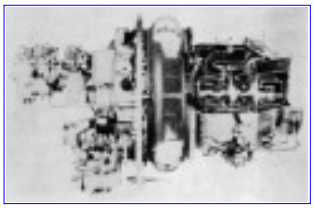


Longitudinal section through basic PW200

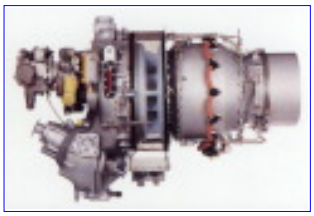


PW206A

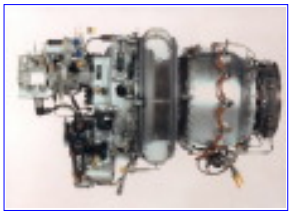


Part cutaway PW206A

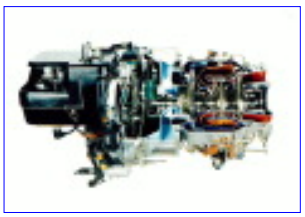
PW206A in MD Explorer 900



PW206B



PW206C



Cutaway PW207D (2000)

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AERO-ENGINES - TURBOFAN, CANADA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

PRATT & WHITNEY CANADA

(Subsidiary of United Technologies Corporation)

P&WC JT15D

Following a comprehensive study of small turbofan performance, detail design of the JT15D began in June 1966. The first engine was started on the bench on 23 September 1967, and flight testing began on 22 August 1968, with an engine in a pod hung under an Avro CF-100. Since then, numerous versions have been developed:

JT15D-1

Initial production version, with T-O rating of 9.79 kN (2,200 lb st). Powers Cessna Citation and early Aérospatiale Corvettes. First delivery August 1969 for Citation 500 prototype. TBO 3,500 hours.

JT15D-1A

Minor changes to power Citation I. Certification 1976. TBO 3,500 hours. Note: Sierra Industries Eagle II conversion replaces these engines by Williams FJ44-2A.

JT15D-1B

Further improvements, without change in ratings. Certificated July 1982, and replaced -1A in Citation I in 1983. TBO 3,000 hours. By July 2000 the JT15D-1 family had accumulated over 9.1 million hours on 1,753 engines.

JT15D-4

First growth version, with T-O rating 11.12 kN (2,500 lb st). Single axial core booster added, rotating with the fan. Certificated September 1973, though the first engines were delivered for Corvette production in August 1972. This engine also powers the Citation II and Mitsubishi Diamond I. TBO 3,500 hours.

JT15D-4B

Altitude-optimised variant, with unchanged ratings. Powers the Citation S/II. Certificated 1983. TBO 3,500 hours.

JT15D-4C

Oil system for sustained flight under negative *g*, and full-throttle electronic fuel control. Ratings unchanged. Certificated 1982. Powers Agusta S.211. TBO 1,500 hours.

JT15D-4D

Flat-rated (maxima unchanged) for improved hot/high performance. Certificated 1983. Powers Diamond IA. TBO 3,500 hours. By July 2000 the JT15D-4 family had accumulated over 11.3 million hours on 2,195 engines.

JT15D-5

{Second-stage growth version, with T-O rating 13.2 kN (2,965 lb st). New fan with higher pressure ratio and greater mass flow, and improved core booster, HP compressor, HP turbine rotor and electronic control. Development started 1977, first flight April 1978 and certificated in 1983. Powers Cessna NT-47A, Diamond II and Beechjet 400A. TBO 3,600 hours. Original choice for VisionAire Vantage with T-O rating 12.9 kN (2,900 lb st).

JT15D-5A

Hydromechanical fuel control. T-O rating 12.9 kN (2,900 lb st). Certificated 1988. Powers Citation V. TBO 3,500 hours.

JT15D-5B

Dash 5A engine modified to suit demands of Beech T-1A Jayhawk. Ratings unchanged. Certificated 1990. TBO 4,500 hours.

JT15D-5C

Oil system for sustained flight under negative *g*. T-O rating 14.19 kN (3,190 lb st). Powers Agusta S.211A and Northrop Grumman Pegasus UCAV-N (Unmanned Combat Air Vehicle, US Navy). Certificated 1991. TBO 2,500 hours.

JT15D-5D

New fan with integrally bladed rotor with wide-chord blades rotating against abradable Al/Kevlar case, broad-chord integrally bladed booster, single-crystal HP turbine rotor blades and restaggered LP turbine blades. T-O rating 13.54 kN (3,045 lb st) with improved fuel economy. Certificated 1993. TBO 3,900 hours. Powers Citation V Ultra and UC-35A, and selected in June 1999 for VisionAire Vantage.

JT15D-5F

T-O rating 12.9 kN (2,900 lb st). Certificated 1993. Powers Beech TCX. TBO 3,000 hours. By late 2001 a total of over 2,400 JT15D-5 engines had accumulated over 5.95 million hours.

By July 2001 P&WC had delivered over 6,000 JT15D engines. These had logged over 28 million hours in more than 2,700 aircraft of 1,800 operators in 80 countries. The high-time engine had accumulated almost 29,000 hours.

The following description refers generally to the JT15D-1B, except where otherwise indicated:

Type

Two-shaft turbofan.

Intake

Direct pitot intake without inlet guide vanes. Hot-air anti-icing for nose bullet.

Fan

Single-stage axial fan, aerodynamically related to that of the JT9D but on a much smaller scale. Forged disc fitted with 28 solid titanium blades, secured by dovetail fixings riveted to disc. Blades have part-span shrouds. Casing, which forms the engine air intake, of forged stainless steel. Circular splitter ring behind fan, held between two rows of 33 inner wrapped-sheet stators and single row of 66 outer stator blades. Mass flow 34 kg (75 lb)/s. Bypass ratio 3.3. Fan pressure ratio 1.5.

Compressor

Primary airflow enters eye of single-stage titanium centrifugal compressor. Single-sided impeller, with 16 full vanes and 16 splitter canes, secured to shaft by special bolt and key-washer. Two-piece casing with diffuser in form of pipes containing straightening vanes. Mass flow 8 kg (17.5 lb)/s. Overall pressure ratio 9.9. JT15D-4 compressor airflow augmented by axial boost stage between fan and compressor, to about 9 kg (19.8 lb)/s, with OPR 11.5. JT15D-5 OPR 12.6, JT15D-5D OPR 13.5.

Combustion Chamber

Annular reverse-flow type. Outer casing of heat-resistant steel; flame tube of nickel alloy, supported on low-pressure turbine stator assembly. Spark igniters at 5 and 7 o'clock positions (viewed from rear).

Turbine

Single-stage HP turbine with 71 solid blades held in fir-tree roots in thick-hub disc of refractory alloy; two-stage LP turbine with nickel alloy discs, first stage with 61 blades and second stage carrying 55 blades, in each case in fir-tree roots. LP fan shaft drives fan, with ball thrust bearing behind fan and roller gear and intershaft bearings; HP shaft drives centrifugal compressor, with front ball thrust bearing and rear roller bearing. Gas temperatures 960ºC before turbine, 562ºC after turbine.

Jetpipe

Nickel alloy cone and sheet-metal pipe. Provision made for adjusting the area to match engines and to trim performance.

Mounting

Hard or soft, according to customers' choice. Four main pods on front casing, arranged two on each side at 30º above and below horizontal. One rear mount at top on either side of centreline.

Accessories

Package under front of engine driven by power offtake from front of HP shaft.

Starting

Air-turbine starter or electric starter/generator.

Control System

Engine-driven sandwich-mounted pump delivering through FCU, flow divider and dual manifolds at 45.7 kg/cm^2 (650 lb/sq in); DPL-1 pneumatic control unit mounted on pump.

Fuel Specification

JP-1, JP-4, JP-5 conforming to PWA Spec 522.

Oil System

Integral oil system, with gear-type pump delivering at up to 5.62 kg/cm^2 (80 lb/sq in). Capacity 7.87 litres (2.08 US gallons, 1.73 Imp gallons. JT15D-5C capacity 9 litres (2.4 US gallons; 2.0 Imp gallons).

Oil Specification

Oil to PWA 521-B.

Dimensions

Diameter: JT15D-1	693 mm (27.3 in)
JT15D-4 (all), -5, -5A, -5C	686 mm (27.0 in)
JT15D-5B, -5D, -5F	711 mm (28.0 in)
Length overall: JT15D-1	1,506 mm (59.3 in)
JT15D-4, -5A, -5C	1,531 mm (60.4 in)
JT15D-4B, -4C, 5B, -5D, -5F	1,600 mm (63.0 in)

Weight, Dry

JT15D-1	232.5 kg (512.6 lb)

JT15D-1A	233.15 kg (514 lb)
JT15D-1B	235.42 kg (519 lb)
JT15D-4	253 kg (558 lb)
JT15D-4B	258 kg (569 lb)
JT15D-4C	261 kg (575 lb)
JT15D-4D	255 kg (562 lb)
JT15D-5	292.6 kg (645 lb)
JT15D-5A,-5B, -5F	288 kg (635 lb)
JT15D-5C	302 kg (666 lb)
JT15D-5D	284.4 kg (627 lb)

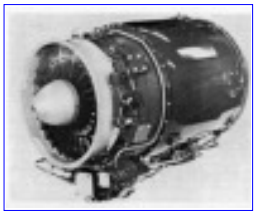
Performance Ratings

T-O: See model listing	
Max continuous:	
JT15D-1, -1A, -1B	9.3 kN (2,090 lb st)
JT15D-4, -4B, -4C, -4D	10.56 kN (2,375 lb st)
JT15D-5, -5A, -5B	12.9 kN (2,900 lb st)
JT15D-5C	14.19 kN (3,190 lb st)
JT15D-5D	13.54 kN (3,045 lb st)

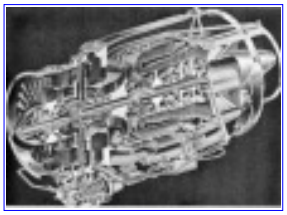
Specific Fuel Consumption (T-O)

JT15D-1, -1A, -1B	15.30 mg/Ns (0.540 lb/h/lb st)
JT15D-4, -4B, -4C, -4D	15.92 mg/Ns (0.562 lb/h/lb st)
JT15D-5	15.69 mg/Ns (0.554 lb/h/lb st)
JT15D-5A, -5B, -5F	15.61 mg/Ns (0.551 lb/h/lb st)
JT15D-5C	16.23 mg/Ns (0.573 lb/h/lb st)
JT15D-5D	15.86 mg/Ns (0.560 lb/h/lb st)

UPDATED

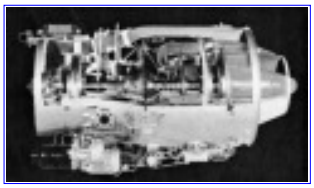


JT15D-1

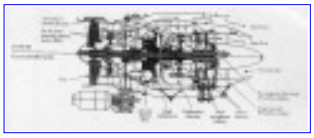


Cutaway drawing of JT15D-1

JT15D-5



Cutaway JT15D-5



Longitudinal sections showing differences between JT15D-5D (upper half) and earlier Dash-5 versions (lower half)

JT15D-5D

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AERO-ENGINES - TURBOFAN, CANADA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

PRATT & WHITNEY CANADA

P&WC PW800

At the Farnborough airshow in September 2000 Pratt & Whitney Canada disclosed the fact that it was working on a demonstrator engine called the Advanced Technology Fan Integrator (ATFI), intended to lead to an engine suitable for the 50- to 90-passenger market, which implied the thrust range 44.48-84.53 kN (10,000-19,000 lb st). The following is what we reported at that time:

Having got off to what appears to have been a false start with the JV (Joint Venture) with Snecma (see under JV in International section), Pratt Canada is now determined to make up for lost time. CEO Ouimet said ``No point in arriving on the market in third place; we intend to leapfrog the competition". He sees the competition as the GE CF34, with a possible second player in RR Deutschland.

The emphasis will be on rugged simplicity. He even predicted ``a double-digit improvement in life operating costs". Astonishingly, the first ATFI engine is to go on test in early 2001, indicating that a large effort has been committed to it - especially on the vital gearbox - for several years past. Ouimet said ``I would expect, for a start, a couple of new airframes to be developed or adapted around this engine".

The following are preliminary details:

Type

High-bypass ratio geared turbofan.

Fan

Single-stage, with advanced blades, driven by reduction gear from LP turbine.

Compressor

Robust, high work per stage, very few separate parts. Based on that of PW308.

Combustor

Minimum emissions, again based on that of PW308.

HP Turbine

Based on PW308.

LP Turbine

Based on PW308, with geared drive to match high-speed turbine to much slower fan.

Performance ratings

Aimed at bracket 44.48 to 84.53 kN (10,000 to 19,000 lb st).

Nothing more was said until the Paris airshow, on 19 June 2001. The following is the announcement in its entirety:

`Pratt & Whitney Canada Corp. (P&WC) is bringing a game changer to the marketplace with its new family of geared turbofan engines, the PW800. Leveraging the vast experience and advanced technology from the Pratt & Whitney family, the PW800 will set new standards in performance and aircraft level operating economics in the industry.

`"We believe that geared turbofans are the next big step in engine performance, efficiency and economy for the regional airline market," says Gilles Ouimet, President and Chief Executive Officer. "The PW800 represents a new family of environmentally friendly `green' engines that will burn less fuel, be quieter and cleaner, while requiring far fewer parts than current engines."

`Spanning the 10,000 to 20,000 lb thrust range, the PW800 will power the next-generation 50- to 100+-passenger regional jets and future large business aircraft. The new PW800 family will complete the Pratt & Whitney product line by bridging the gap between the PW300 series for small regional and business jets, in the 5,000 to 8,000 lb thrust range, and the PW6000 family, in the 20,000+ lb thrust range.

`"We see the evolution of the requirements for the next-generation regional and business aircraft to be essentially driven by their operating economics as well as environmental concerns," says Mr Ouimet. "Accordingly, the PW800 will bring fundamental change to the market by delivering the best technology and highest value to our customers at the lowest cost."

`The PW800's demonstrator, the Advanced Technology Fan Integrator (ATFI), completed its first run in March, setting the stage for the flight test programme in the coming year. The low spool technologies and operability demonstration are currently well advanced, and a core technologies demonstration for the ATFI is being planned for the second half of 2002.

`Mr Ouimet says that discussions are ongoing with potential aircraft manufacturers for the PW800. "We are very excited about the interest it has generated already, and anticipate the product to be available within 36 months of a programme `go'."

`The power plant will burn approximately 10 per cent less fuel than competitive engines, with very low emissions aiming beyond Zurich 5 requirements to cater for the more stringent limits anticipated in

the future. Its noise level will target a -28 dB cumulative noise margin relative to Stage 3, and will also be comfortably below Stage-4 limitations.

`The PW800 programme and its demonstrator, the ATFI, is being undertaken by the total Pratt & Whitney Group, leveraging the strengths of both Pratt & Whitney Canada's extensive regional airline experience, together with that of Large Commercial Engines' with the large carriers. Also participating as partners are MTU Aero Engines from Germany, supplying the low-pressure turbine for the engine, and FiatAvio from Italy, responsible for the fan drive gearbox assembly, intermediate case and accessory gearbox.

`The PW800 engine family features a reduction gearbox that allows the fan to run at a slower speed than the low-spool compressor and turbine, permitting all components to operate at their optimum speeds for maximum efficiency. The slower fan speed contributes to very low noise levels, while the higher turbine and LP compressor speeds lead to an engine configuration with fewer stages and smaller turbines to do the same work. Having fewer parts relative to a conventional, ungeared design will significantly reduce in-service operating costs for our customers.

`This new engine family also features an advanced high-performance fan design, which brings the dual advantage of better performance derived from increased fan flow capacity and reduced noise levels, enabling the power plant to meet the increasingly strict rules imposed by airports around the world.'

That is about all P&WC was prepared to say in mid-2001, and the images it released at that time give only a hazy idea of the ATFI and PW800. The Editor asked Mr Ouimet whether the extra weight, complexity and potential source of trouble of the gearbox were really more than counterbalanced by the gains. He replied: "Unquestionably. Compared with previous engines in this thrust class the PW800 will have significantly fewer parts". He was not prepared to elaborate, though the Editor suggested that a slower fan is bound to mean one that is bigger and heavier, and a gearbox transmitting several thousand horsepower is not only heavy, and probably with no redundant load path, but it also needs a considerable flow of oil to carry away the heat (lost energy) from the meshing gears and bearings.

However, the parent company in Connecticut has been working on geared turbofans for many years, and claims to have been achieving a major breakthrough in gearbox heat loss. This is outlined in the entry on the PW8000, in the USA section.

Nobody can fail to be deeply impressed by P&WC, which so far has enjoyed fantastic success and (apart from the JV) hardly put a foot wrong. Over the years the members of the engineering team at Montreal have by sheer toil and ability climbed to the position where they stand on an equal footing with their colleagues in the parent in Connecticut. For many years the two teams have worked closely on geared engines, the parent company's PW8000 being aimed at a higher thrust-class than the PW800. Other things being equal, smaller engines mean smaller risk, so in 2001 it looks as if the PW800 may be likely to reach the marketplace before the PW8000. Its service experience would then underpin the larger engine.

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ATFI (2002)

PW800
(2002)

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AERO-ENGINES - TURBOSHAFT, CANADA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

PRATT & WHITNEY CANADA

(Subsidiary of United Technologies Corporation)

P&WC PW100T

Just as the PT6T was derived from the PT6 turboprop, so is the PW100T family of turboshaft engines being developed from the PW100 turboprop to meet the power needs of large helicopters. Thanks to the development of the PW150 family the derived helicopter engines could provide shaft power up to about the 5,966 kW (8,000 shp) level. The problem in marketing is that, astonishingly, no large helicopters are being developed outside Russia.

PW127T

Initial rating (T-O and OEI) probably to be 2,580 kW (3,480 shp), with considerably greater power available for 2.5 min. Under development to power the prototype Mil Mi-38, which is being manufactured by KAPO at Kazan under contract to the M L Mil design authority, with major participation by Eurocopter. The transmission is the responsibility of the Krasnyi Oktyabr plant in St Petersburg. It is expected that the PW127T will be the standard powerplant of export versions of the Mi-38. For the former-Soviet market Klimov is developing the TVA-3000.

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AERO-ENGINES - TURBOFAN/ - TURBOPROP/ - TURBOSHAFT, CANADA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

PRATT & WHITNEY CANADA

(Subsidiary of United Technologies Corporation)

P&WC PW6XX

Presumably eventually to be restyled as the PW600 family, this designation covers a range of projects for future turbofans, turboprops and turboshaft engines. Announced at the US National Business Aircraft Association (NBAA) show in October 1999, this totally new family of engines is to provide `power for the entire GA market'.

CEO David Caplan said that the broad outline of the core had been decided after discussion with aircraft manufacturers and hinted that, when detailed design began in early 2000, versions of the engine would already be linked to projected future aircraft. He said that a compressor was already on test, and that a complete core would run by the end of 1999. He further stressed that, given a choice between advanced technology and low cost, the PW6XX would be slanted towards low cost.

The following summarises what was disclosed in October 1999:

Type

Simple two-shaft engine.

Inlet

Varies with type of output.

Fan

Turbofan only, single axial stage.

Compressor

Single-stage axial followed by single-stage centrifugal on same shaft.

Combustion Chamber

Folded annular.

HP Turbine

Single stage driving compressor.

LB Turbine

Single Stage Driving Fan Or Output shaft, contra-rotating, no stators needed.

Jetpipe

Turbofan, mixer nozzle with 16 lobes; others, a plain pipe.

Accessories

Driven off HP shaft, on underside of turbofan, on aft face of gearbox of turboprop and on front of turboshaft.

Weight, Dry

Turbofan	159 to 181 kg (350 to 400 lb)
Turboprop	136 to 159 kg (300 to 350 lb)

Performance Rating

(T-O, S/L)

Turbofan	c4.5 to c9.0 kN (1,000 to 2,000 lb st) class
Turboprop	447 to 671 kW (600 to 900 shp)

Specific Fuel Consumption

Turboprop	20 per cent below today's engines

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PW6XX turbofan mockup (2000)

PW6XX turbofan mockup (2000)

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AERO-ENGINES - TURBOSHAFT, CANADA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

PRATT & WHITNEY CANADA

(Subsidiary of United Technologies Corporation)

P&WC LOW-COST ENGINE

The following is the complete text of a statement issued at the Paris Air Show on 14 June 1999:

``Pratt & Whitney Canada Inc. (P&WC) has launched a series of concept studies for an all-new engine family specifically designed for the general aviation and small business aircraft markets. This new engine family is targeted to significantly reduce ownership cost, while providing the legendary reliability and durability standards of P&WC engines.

``Extensive end user market studies and consultation with potential aircraft manufacturers in 1998 have indicated the emerging need for a turbofan engine in the 1,000 - 2,500 lb thrust range, as well as a derivative turboprop engine using the same core. Low acquisition cost is one of the key drivers highlighted by these studies.

``P&WC has identified aerodynamic designs aimed at minimising parts count, engine operating cycles assuring optimal use of materials, and installation effects which simplify system integration. These technologies will be demonstrated and analysed at P&WC by the end of 1999, not only to quantify performance but also to assess their contribution to the cost objective. The technology program will run in parallel with various concept design and market studies towards a possible engine launch based on market conditions".

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AERO-ENGINES - TURBOSHAFT, CANADA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

PRATT & WHITNEY CANADA

(Subsidiary of United Technologies Corporation)

P&WC PT6T TWIN-PAC

First run in July 1968, the Twin-Pac comprises two PT6 turboshaft engines side by side and driving a combining gearbox. The high-time Twin-Pac had logged 24,970 hours by mid-1996. Some operators have been granted TBO extensions to 7,250 hours on power sections and 6,500 hours on reduction gearboxes. By October 1998 over 7,200 PT6T engines had accumulated over 29 million hours.

PT6T-3

T-O rating 1,342 kW (1,800 shp). For Bell 212, UH-1N and CH-135, Agusta-Bell AB212 and 212ASW and California/Sikorsky S-58T.

In these applications, shaft power is limited by the transmission. In the Model 212 the 1,342 kW (1,800 shp) PT6T-3 is restricted to a T-O rating of 962 kW (1,290 shp) and 843 kW (1,130 shp) for continuous power. In the S-58T the limits are 1,122 kW (1,505 shp) at T-O and 935 kW (1,254 shp) for continuous operation.

PT6T-3B

PT6T-3 with some T-6 hardware and improved single-engine performance. Bell 212, 412 and 412SP.

PT6T-3BE

PT6T-3B with upgraded combining reduction gearbox and modified torque control unit. For Bell 412HP and AB 412HP.

PT6T-3BF

Offers 5.7 per cent increase in 30-min OEI rating over PT6T-3B.

PT6T-3BG

Offers 5.7 per cent increase in 30-min OEI rating over PT6T-3BE.

PT6T-3D

Improved engine for 412EP. Certificated August 1993.

PT6T-3DF

In response to requests for increased hot/high performance, offers a 5.7 per cent increase in 30-min OEI thermodynamic rating relative to -3D.

PT6T-6

Improved compressor-turbine nozzle guide vanes and rotor blades. S-58T and AB 212.

PT6T-6B

Upgraded, combining reduction gearbox and modified torque control. AB 412HP.

The following features differ from the PT6:

Type

Coupled free turbine turboshaft.

Intake

Additional inertial particle separator to reduce ingestion. High-frequency compressor noise suppressed.

Output

Combining gearbox comprises three separate gear trains, two input and one output, each contained within an individual sealed compartment and all interconnected by driveshafts. Overall reduction ratio 5.

Accessories

Starter/generator and tachogenerator on accessory case at front of each power section. Other drives on gearbox, including power turbine governors and tachogenerators, and provision for blowers and aircraft accessories.

Starting

Electrical, with cold weather starting down to -54~C.

Control System

As PT6 with manual back-up, and dual manifold for cool starts. Automatic power sharing and torque limiting.

Fuel Specification

JP-1, JP-4 and JP-5.

Oil Specification

PWA Spec 521. For military engines, MIL-L-7808 and -23699.

Dimensions

Length	1,671 mm (65.8 in)
Width	1,105 mm (43.5 in)
Height	828 mm (32.6 in)

Weight, Dry

(standard equipment):

PT6T-3	298 kg (656 lb)
PT6T-3B, -6	299 kg (660 lb)
PT6T-3BE, -3BG, -6B	305 kg (673 lb)
PT6T-3BF	303 kg (668 lb)
PT6T-3D	317 kg (690 lb)
PT6T-3DF	309 kg (680.5 lb)

Performance Ratings

T-O (5 min):	
Total output, at 6,600 rpm:	
PT6T-3 (all)	1,342 kW (1,800 shp)
PT6T-6	1,398 kW (1,875 shp) (to 21~C)
Single power section only, at 6,600 rpm:	
PT6T-3, -3B, -3BF	671 kW (900 shp)
PT6T-6, -6B, -3BE (2.5 min)	764 kW (1,025 shp)
PT6T-3D (2.5 min)	820 kW (1,100 shp)
30 min power (single power section), at 6,600 rpm:	
PT6T-3B, -3BE, -6	723 kW (970 shp)
Cruise A:	
Total output, at 6,600 rpm:	
PT6T-3, -3B, -3BE	932 kW (1,250 shp)
PT6T-6	1,014 kW (1,360 shp)
Single power section only, at 6,600 rpm:	
PT6T-3, -3B, -3BF	466 kW (625 shp)
PT6T-3D, -6	500 kW (670 shp)
Cruise B:	

Total output, at 6,600 rpm:	
PT6T-3, -3B, -3BE, -3BF, -3BG	820 kW (1,100 shp)
PT6T-6	891 kW (1,195 shp)
Single power section only, at 6,600 rpm:	
PT6T-3, -3B, -3BF	410 kW (550 shp)
PT6T-6	440 kW (590 shp)
Ground idle, at 2,200 rpm	44.7 kW (60 shp) max

Specific Fuel Consumption

At 2.5 min rating (single power section):	
PT6T-3B, -3BE, -3BF	100.7 Ág/J (0.596 lb/h/shp)
PT6T-6, -6B	101.6 Ág/J (0.602 lb/h/shp)

Oil Consumption

Max (for both gas generators)	0.18 kg (0.4 lb)/h

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PT6T-3D

PT6T cutaway display engine

PT6T-3B in Bell 212

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AERO-ENGINES - MANUFACTURER, CANADA

Date Posted: 22 November 2001

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P&WC - PRATT & WHITNEY CANADA CORPORATION (Subsidiary of United Technologies Corporation)

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Pratt & Whitney Canada is a wholly-owned subsidiary of United Technologies Corporation, Connecticut, USA, and is the P&W Group member responsible for engines for general aviation and regional transport.

Originally a mere support centre for P&W piston engines, in the late 1950s it launched a small turboprop, the PT6. From then on commercial success could fairly be described as explosive. Initially, P&WC concentrated on the bottom end of the size scale, but today it sees its remit as extending up to 16,000 pounds thrust, where the PW6000 takes over'.

Sales in 2000 exceeded C$2.4 billion, comprising C$1.0 billion for engines and C$1.4 billion for aftermarkets. Of this total, C$2.1 billion was exported. A total of 51,100 engines had been delivered (1,924 in 2000), logging more than 375 million hours in over 21,100 currently active aircraft in more than 180 countries. Employees totalled 8,960, of whom 6,980 were in Canada.

Headquartered near Montreal, P&WC has facilities in Ontario, Nova Scotia and Alberta. In some of its programmes it is partnered by MTU of Germany, a specialist in LP turbines, and together with that partner operates a customer support centre at Ludwigsfelde, Berlin. Another partner is FiatAvio of Italy. In February 1996 it inaugurated a customer support centre in Singapore. In April 1996 it opened a customer support centre (a joint venture with two local companies) in South Africa.

In August 1993 P&WC signed an agreement to form a joint venture with Klimov of St Petersburg (see under Russia) to build engines for the Russian Federation and Associated States (CIS) market. Out of this has evolved a company called Pratt & Whitney (Rus). Based in St Petersburg, Russia, this is a wholly owned subsidiary of P&WC, which purchased Klimov's share.

In July 1994 P&WC established a joint venture with the Polish company WSK-PZL-Kalisz, and now wholly owns the resulting P&W (Kalisz), which makes precision parts for P&WC engines. In March 1998, it completed the formation of a joint venture with SAEC of China; the new company, Southern Pratt & Whitney Aero-Engine Co, is described under China.

P&WC also produces APUs, and its Industrial & Marine division produces aero-derived engines for surface applications.

P&WC offers customers a range of warranties, a Total Customer Support package, Engine Operating Cost Protection plans and an Early Operator Protection plan. It also operates an Eagle Service Plan, available over a wide range of corporate engines, which is a pre-budget maintenance plan under the terms of which P&WC assumes responsibility for scheduled overhaul and HSI, unscheduled maintenance outside warranty, unscheduled LRU/accessory maintenance and any required product-support improvements at shop visits. It also runs a school for customer personnel.

2nd Quarter 2001 Statistics

	PT6A	PT6B/C/D/T	PW100	PW200	JT15D	PW300	PW500	APUs
Engines delivered	29,453	7,504	4,700	690	6.008	894	927	1,014
Certified A/C appl.	60	19	29	9	12	5	3	2
Applications	104	19	13	4	17	4	3	2
Different operators	5,547	432	291	152	1,818	183	264	38
Airline operators/Aircraft	454/1,852	n/a	246/1,796	n/a	47/55	9/67	1/1	41/554
A/C in operation	12,188	1,834	1,948	230	2,709	339	300	547
Different countries	160	95	91	34	80	27	28	28
Highest total time - hours	64,410	25,850	35,490	4,110	28,980	6,960	3,510	29,300
Operating time - hours	230,419,300	32,337,600	66,782,000	344,100	28,071,000	1,498,900	426,500	15,003,500

Total engines delivered	51,190
Total a/c in operation	20,140
Total operating hours	374,953,500

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AERO-ENGINES - TURBOPROP, CANADA

Date Posted: 22 November 2001

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PRATT & WHITNEY CANADA

(Subsidiary of United Technologies Corporation)

P&WC PW100

During the 1970s, Pratt & Whitney Canada became increasingly convinced that it should design a completely new free-turbine engine in the thermodynamic power class of 1,865 kW (2,500 shp), though limited by its gearbox to 1,119 kW (1,500 shp). Such an engine was expected to take over the market carved out by the RR Dart, especially 30/40-seat passenger aircraft (now extended to 70-seat). The engine was launched in 1979, with the designation PT7, when it was picked by Embraer for the EMB-120.

From the outset it was planned as a modular turbomachine entirely separate from the reduction-gear module, with the air fed via an upward curving duct from an inlet under the propeller. Another early choice was to use tandem centrifugal compressors, as in the Dart, but to drive each by its own turbine, with a third shaft passing down the centre linking the power turbine and gearbox. Flight development in the nose of a Viscount began in February 1982, by which time the engine had been redesignated in the Pratt & Whitney style as the PW100 family, with the final two digits denoting the power in hundreds of horsepower. Since then, the following versions have been announced:

PW118

T-O rated at 1,411 ekW; 1,342 kW (1,892 ehp; 1,800 shp) at 1,300 propeller rpm to 33ºC. EMB-120 Brasilia. Certificated March 1986.

PW118A

T-O rated at 1,411 ekW; 1,342 kW (1,892 ehp; 1,800 shp) at 1,300 propeller rpm to 42.1ºC. EMB-120 Brasilia. Certificated June 1987.

PW118B

T-O rated at 1,411 ekW; 1,342 kW (1,892 ehp; 1,800 shp) at 1,300 propeller rpm to 44.9ºC. EMB-120 Brasilia.

PW119B

T-O rated at 1,702 ekW; 1,626 kW (2,282 ehp; 2,180 shp) at 1,300 propeller rpm to 31.1ºC. Dornier 328. Certificated June 1993.

PW119C

T-O rated at 1,702 ekW; 1,626 kW (2,282 ehp; 2,180 shp) at 1,300 propeller rpm to 36.1ºC. Dorner 328.

PW120

T-O rated at 1,566 ekW; 1,491 kW (2,100 ehp; 2,000 shp) at 1,200 propeller rpm to 27.7ºC. Aérospatiale/Alenia ATR 42 and Snow SA-210TA. Certificated December 1983.

PW120A

T-O rated at 1,566 ekW; 1,491 kW (2,100 ehp; 2,000 shp) at 1,200 propeller rpm to 27.9ºC. Dash 8-100. Certificated September 1984.

PW121

T-O rated at 1,679 ekW; 1,603 kW (2,252 ehp; 2,150 shp) at 1,200 propeller rpm to 25.7ºC. Dash 8-100 and ATR 42. Certificated February 1987.

PW121A

T-O rated at 1,718 ekW; 1,640 kW (2,304 ehp; 2,200 shp) at 1,200 propeller rpm to 25ºC. ATR 42.

PW123

T-O rated at 1,866 ekW; 1,775 kW (2,502 ehp; 2,380 shp) at 1,200 propeller rpm to 35ºC. Dash 8-300. Certificated June 1987.

PW123AF

T-O rated at 1,866 ekW; 1,775 kW (2,502 ehp; 2,380 shp) at 1,200 propeller rpm to 35ºC. CL-215/415. Certificated February 1990.

PW123B

T-O rated at 1,958 ekW; 1,864 kW (2,626 ehp; 2,500 shp) at 1,200 propeller rpm to 30.3ºC. Dash 8-300. Certificated November 1991.

PW123C

T-O rated at 1,687 ekW; 1,603 kW (2,262 ehp; 2,150 shp) at 1,200 propeller rpm to 25.5ºC. Dash 8-200.

PW123D

T-O rated at 1,687 ekW; 1,603 kW (2,262 ehp; 2,150 shp) at 1,200 propeller rpm to 45ºC. Dash 8-200.

PW123E

T-O rated at 1,866 ekW; 1,775 kW (2,502 ehp; 2,380 shp) at 1,200 propeller rpm to 40.6ºC. Dash 8-315.

PW124

Growth version, T-O rating 1,880 ekW; 1,790 kW (2,522 ehp; 2,400 shp) to 34.4ºC. BAe (Jetstream) ATP and Fokker 50.

PW124B

PW124 with PW123 turbomachinery to suit four-blade propeller at 1,200 rpm, same rating. ATR 72. Certificated May 1988.

PW125B

Growth PW124 with T-O rating of 1,958 ekW; 1,864 kW (2,626 ehp; 2,500 shp) at 1,200 propeller rpm to 30ºC. Fokker 50. Certificated May 1987.

PW126

Growth engine, maximum contingency 2,078 ekW; 1,978 kW (2,786 ehp; 2,653 shp) at 1,200 propeller rpm to 32.4ºC. Jetstream ATP. Certificated May 1987.

PW126A

Growth 124A with T-O rating of 2,084 ekW; 1,985 kW (2,795 ehp; 2,662 shp) at 1,200 propeller rpm to 29.2ºC. Jetstream ATP. Certificated June 1989. In September 1998 TBO on the PW126 and PW126A was extended from 7,000 to 8,000 hours.

PW127

T-O rated at 2,147.6 ekW; 2,051 kW (2,880 ehp; 2,750 shp) at 1,200 propeller rpm to 32ºC. Improved turbines drive higher airflow compressor. ATR 72 and ATR 42-500. EIS January 1993.

PW127A

T-O rated at 1,864 kW (2,500 ehp). An-140.

PW127B

T-O rated at 2,147.6 ekW; 2,051 kW (2,880 ehp; 2,750 shp) at 1,200 propeller rpm to 30ºC. Selected for hot and high performance Fokker 50.

PW127C

T-O rated at 2,147.6 ekW; 2,051 kW (2,880 ehp; 2,750 shp) at 1,200 propeller rpm to 30.2ºC. XAC Y7-200A.

PW127D

T-O rated at 2,147.6 ekW; 2,051 kW (2,880 ehp; 2,750 shp) at 1,200 propeller rpm to 33ºC. Jetstream 61. Certificated January 1994.

PW127E

T-O rated at 1,876 ekW; 1,790 kW (2,516 ehp; 2,400 shp) at 1,200 propeller rpm to 45ºC. ATR 72-500 and proposed 42MP.

PW127F

T-O rated at 2,147.6 ekW; 2,051 kW (2,880 ehp; 2,750 shp) at 1,200 propeller rpm to 34.9ºC. Selected for both passenger and cargo versions of Ilyushin Il-114PC produced by Tashkent APO, Uzbekistan.

PW127G

T-O rated at 2,177 kW (2,920 shp) or 1,972 kW (2,645 shp) with APR. Thermodynamic power 2,646.5 ekW (3,549 ehp) military or 2,580 ekW (3,460 ehp) civil. Selected with Hamilton RF-568F-5 six-blade propeller for CASA C-295. Certificated third quarter 1997, to enter service 2000.

PW127H

T-O rated at 2,051 kW (2,750 shp) or 1,972 kW (2,645 shp) with APR. Selected 1999 to power Il-114-100.

PW127J

Upgraded 127C, with same powers as PW127F, to power Y7-200A.

PW150A

T-O rated at 4,095 ekW; 3,781 kW (5,492 ehp; 5,071 shp) at 1,020 propeller rpm to 37.4ºC. Growth derivative for high-speed 50/80-seat aircraft. Increased mass flow through new three-stage axial

compressor and single-stage centrifugal, driven independently by new air-cooled single-stage HP and LP turbines; advanced combustion system, additional turbine cooling and high-power low-speed reduction gearbox. First engine run 9 June 1996, first flight on B-720 October 1996. Certificated by Transport Canada June 1998. Entered service February 2000 on Dash 8Q Series 400 at a flat rating of 3,781 kW (5,071 shp) to 37.4ºC, driving Dowty six-blade low-speed propellers, and as alternative engine for export versions of Tupolev Tu-136 rated at 2,634 kW (3,600 shp) to 40ºC driving Hartzell six-blade propellers.

By July 2000 P&WC had delivered over 4,600 PW100 turboprop engines. These had logged over 60 million hours in 1,960 aircraft of 298 operators in 94 countries. Airline operators numbered 220, with 1,670 PW100-powered aircraft. The high-time engine had accumulated 25,210 hours. More than half the operators have instituted On Condition Maintenance. In early 1999 a PW124B of LOT Polish Airlines completed 11,327 hours on wing, then requiring only a hot-section refurbishment.

The following description applies to all PW100 series engines except for the PW150A:

Type

Free-turbine turboprop.

Intake

S-bend duct. A secondary duct forms a flowing bypass to prevent foreign object ingestion.

Compressor

Two centrifugal impellers in series, each driven by its own turbine. Air guided through ring of curved pipes from LP diffuser to HP entry. Pressure ratio 10.9 (PW118), 11.4 (PW120), 13.2 (PW119), 13.8 (PW123), 14.4 (PW125), 14.7 (PW127), 18.0 (PW150).

Combustion Chamber

Annular reverse flow type, with 14 air blast fuel nozzles around periphery and two spark igniters.

HP Turbine

Single-stage with 47 air-cooled blades. Single-stage LP with 53 solid blades.

Power Turbine

Two-stage, first with 68 blades and second with 74, all with shrouded tips.

Output

Twin-layshaft gearbox with propeller shaft offset above turbomachine. Maximum propeller speed 1,200 rpm.

Accessories

Pads driven by HP compressor, for starter/generator, hydromechanical fuel control and hand turning. Pads on reduction gearbox for alternator, hydraulic pump, propeller control module, overspeed governor and electric auxiliary pump. Electric torque signal and auto power augmentation.

Starting

Electric starter/generator.

Control System

Hydromechanical control and electronic power management. PW150A, FADEC integrated with EMS (Engine Monitoring System).

Fuel Specification

JP-1, JP-4, JP-5 to PWA Spec 522.

Oil System

One pressure pump and two scavenge pumps, all driven off HP rotor. Integral tank, capacity 9.44 litres (2.5 US gallons, 2.08 Imp gallons).

Oil Specification

CPW202 or PWA521 Type II.

Dimensions

Length:	
PW118, 118A, 119B	2,057 mm (81 in)
others, except PW150A	2,134 mm (84 in)
PW150A	2,423 mm (95.4 in)
Width:	
PW118-121	635 mm (25 in)
others, except PW150A	660 mm (26 in)
PW150A	767 mm (30.2 in)
Height: PW118-121	787 mm (31 in)
others, except PW150A	838 mm (33 in)
PW150A	1,105 mm (43.5 in)

Weight, Dry

PW118	391 kg (861 lb)
PW118A, -118B	394 kg (866 lb)
PW119B	415.5 kg (916 lb)
PW120	417.8 kg (921 lb)
PW120A	423 kg (933 lb)
PW121	425 kg (936 lb)
PW123, 123B, 123AF, -123C, -123D, 123F	450 kg (992 lb)
others, except PW150A	481 kg (1,060 lb)
PW150A	690 kg (1,522 lb)

Performance Ratings

(S/L, static)

T-O: See under model listing	
Max cruise:	

PW118	1,188 ekW; 1,127 kW (1,593 ehp; 1,512 shp) at 1,300 rpm to 19.8ºC
PW118A	1,188 ekW; 1,127 kW (1,593 ehp; 1,511 shp) at 1,300 rpm to 29.4ºC
PW118B	1,188 ekW; 1,127 kW (1,593 ehp; 1,512 shp) at 1,200 rpm to 29.6ºC
PW119B, 119C	1,357 ekW; 1,293 kW (1,820 ehp; 1,734 shp) at 1,200 rpm to 54.8ºC
PW120	1,271 ekW; 1,207 kW (1,704 ehp; 1,619 shp) at 1,200 rpm to 15ºC
PW120A	1,295 ekW; 1,231 kW (1,737 ehp; 1,651 shp) at 1,200 rpm to 15ºC
PW121, 121A	1,330 ekW; 1,268 kW (1,784 ehp; 1,700 shp) at 1,200 rpm to 15ºC
PW123, 123E, 125B	1,593 ekW; 1,514 kW (2,136 ehp; 2,030 shp) at 1,200 rpm to 22.2ºC
PW123AF	as PW123 but to 22.0ºC
PW123B	as PW123 but to 22.6ºC
PW123C	1,532 ekW; 1,454 kW (2,054 ehp; 1,950 shp) at 1,200 rpm to 26.1ºC
PW123D	as PW123C but to 34.4ºC
PW124B	1,639 ekW; 1,557 kW (2,198 ehp; 2,088 shp) at 1,200 rpm to 24.6ºC
PW126	1,634 ekW; 1,553 kW (2,192 ehp; 2,083 shp) at 1,200 rpm to 26.3ºC
PW126A	1,633 ekW; 1,551 kW (2,190 ehp; 2,081 shp) at 1,200 rpm to 27.2ºC
PW127	1,668 ekW; 1,589 kW (2,237 ehp; 2,132 shp) at 1,200 rpm to 22.6ºC
PW127B, 127C	1,667 ekW; 1,589 kW (2,237 ehp; 2,132 shp) at 1,200 rpm to 20.5ºC
PW127D	1,667 ekW; 1,589 kW (2,237 ehp; 2,132 shp) at 1,200 rpm to 24.7ºC
PW127E	1,667 ekW; 1,589 kW (2,237 ehp; 2,132 shp) at 1,200 rpm to 25.5ºC
PW127F, 127J	as PW127E but to 22.6ºC
PW127G	as PW127E but to 24.2ºC
PW127H	as PW127E but to 25.7ºC
PW150A	3,193 ekW; 2,942 kW (4,283 ehp; 3,947 shp) at 1,020 rpm to 25.8ºC

Specific Fuel Consumption

T-O rating:	
PW118	84.2 µg/J (0.498 lb/h/ehp)
PW118A	85.2 µg/J (0.504 lb/h/ehp)
PW119B, 119C	82.8 µg/J (0.490 lb/h/ehp)
PW120, 120A	82.0 µg/J (0.485 lb/h/ehp)
PW121	80.6 µg/J (0.477 lb/h/ehp)
PW121A, 127E	80.1 µg/J (0.474 lb/h/ehp)
PW123, 123AF, 123E	79.4 µg/J (0.470 lb/h/ehp)
PW124B	79.1 µg/J (0.468 lb/h/ehp)
PW123C, 123D	81.6 µg/J (0.483 lb/h/ehp)
PW123B, 125B	78.2 µg/J (0.463 lb/h/ehp)
PW126	78.1 µg/J (0.462 lb/h/ehp)
PW126A	77.9 µg/J (0.461 lb/h/ehp)
PW127, 127B, 127C, 127F, 127H, 127J	77.6 µg/J (0.459 lb/h/ehp)
PW127G	76.6 µg/J (0.453 lb/h/ehp)
PW150A	73.2 µg/J (0.433 lb/h/ehp)

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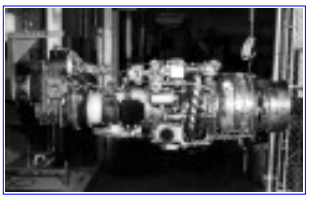
PW118 (2000)



PW120

Cutaway PW100

PW127 (2000)

Longitudinal section through PW150A

PW150A
(2000)

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AERO-ENGINES - TURBOSHAFT, CANADA

Date Posted: 22 November 2001

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PRATT & WHITNEY CANADA

(Subsidiary of United Technologies Corporation)

P&WC PT6B, PT6C, PT6D

These are the designations of the commercial turboshaft versions of the PT6A turboprop, with a generally identical gas generator but lower-ratio output gearbox. The following models have been announced:

PT6B-9

T-O power 410 kW (550 shp) to 25ºC. Certificated May 1965. Powered Lockheed 286 and Piasecki 16H.

PT6B-16

T-O power 515 kW (690 shp). Powered Nord 511.

PT6B-34

T-O power 559 kW (750 shp). Powered a Westland Lynx.

PT6B-35F

Selected for Lear Fan.

PT6B-36

T-O rating 732 kW (981 shp) to 15ºC with 2.5 min OEI rating 770 kW (1,033 shp) to 15ºC. Reverse-drive gearbox giving 6,409 rpm output. Powers Sikorsky S-76B.

PT6B-36A

PT6B-36 with different ratings.

PT6B-36B

PT6B-36 with improved hot section for longer life. Ratings unchanged from PT6B-36A.

PT6B-37A

Thermodynamic rating 747.5 kW (1,002 shp). Several new features (see below). Selected in February 1996 to power Agusta A119 Koala. Certificated July 1999.

PT6C-67A

T-O rating 1,380 kW (1,850 shp). Selected in November 1996 for Bell-Agusta BA 609 tilt-rotor. FAA certification due (engine) December 2001, (aircraft) 2003.

PT6C-67B

This is the first of a new PT6 family of turboshaft engines derived from the PT6A turboprop. Power in the 895 kW (1,200 shp) class. Selected in September 1996 for PZL-Swidnik W-3 Sokól. Certification October 1998.

PT6C-67C

T-O rating 1,252 kW (1,679 shp). Selected to power Agusta Bell AB139 Skyhorse.

PT6D-114A

T-O power 507 kW (680 shp) to 39.3ºC. Powers Soloy Pathfinder 21.

By June 2000 P&WC had delivered 7,415 PT6 turboshaft power sections. These had logged over 30.2 million hours in 1,766 helicopters of 400 operators in 95 countries. The high-time engine had accumulated 25,850 hours.

Type

Free-turbine turboshaft.

Intake

Annular, radially inwards, with protective screen.

Compressor

As PT6A, three-stage axial followed by centrifugal.

Combustion Chamber

As PT6A, folded annular reverse-flow.

Compressor Turbine

As PT6A, single-stage axial.

Power Turbine

As lower powered PT6A versions, single-stage axial.

Output

Offset spur-gear reduction gearbox, with drive facing to either direction (usually under the gas generator). PT6B-37A incorporates a freewheel clutch and provides drives to both front and rear at 4,373 rpm for main and tail rotors.

Control System

Basically hydromechanical, as PT6A, with inter-turbine thermocouple system. PT6B-37A has automatic fuel control, with manual override, and a phase-shift torquemeter. PT6C-67B electronic.

Dimensions

Length:	
PT6B-36 family (excl accessories)	1,504 mm (59.2 in)
PT6B-37A	1,636 mm (64.4 in)
PT6D-114A	1,341 mm (52.8 in)
Width:	
PT6B-36 family	495 mm (19.5 in)
PT6B-37A	564 mm (22.2 in)
PT6D-114A	483 mm (19.0 in)
Height (all)	about 894 mm (35.2 in)

Weight, Dry

PT6B-36	169 kg (372 lb)
PT6B-36A	171 kg (378 lb)
PT6B-36B	175 kg (386 lb)
PT6B-37A	172 kg (380 lb)
PT6D-114A	135 kg (297 lb)

Performance Ratings

T-O: See model listing	
Continuous:	
PT6B-36	649 kW (870 shp) to 15ºC
PT6B-36A, 36B	654 kW (877 shp) to 15ºC
PT6B-37A	671 kW (900 shp) to 15ºC

Specific Fuel Consumption

T-O as above:	
PT6B-36	100.5 µg/J (0.594 lb/h/shp
PT6B-36A, 36B	98.2 µg/J (0.581 lb/h/shp)
PT6B-37A	98.7 µg/J (0.584 lb/h/shp)
PT6D-114A	102.6 µg/J (0.607 lb/h/shp)

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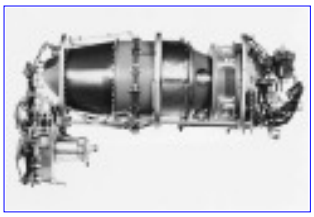
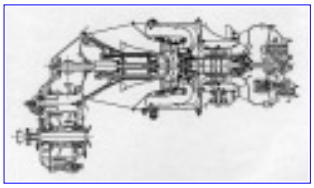


PT6B-36B

PT6B-114A (2000)

PT6B-37A



Longitudinal section through PT6B-37A

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AERO-ENGINES - TURBOPROP, CANADA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

PRATT & WHITNEY CANADA

(Subsidiary of United Technologies Corporation)

P&WC PT6A

US military designations: T74 and T101

In 1958 Pratt & Whitney Canada could see that its business, mainly centred on the 600 hp Wasp piston engine, would continue to decline. The company asked a Montreal bank for a loan to help develop a small gas-turbine, the PT6, to replace the obsolescent engine. The bank, seeing few obvious customers for the proposed engine, politely refused to help.

By July 2001 P&WC had delivered almost 29,500 PT6A engines. These had logged over 230.4 million hours in 12,188 aircraft of 5,547 operators in 160 countries. Airline operators totalled 454, with 1,852 PT6A-powered aircraft. The high-time engine had accumulated 64,410 hours. Demand shows no sign of slackening, and an additional market is being opened up by P&W (Rus).

An experimental PT6 engine ran for the first time in November 1959. It did not achieve its predicted performance, but after some redesign, a PT6A turboprop began flight testing in the nose of a Beech 18 in May 1961. The first production version was the PT6A-6. The same basic power section has since been used in many PT6A turboprop versions; in PT6B commercial and T74 military turboshaft engines, in PT6T and T400 coupled Twin Pacs, and in ST6 APUs and industrial, rail and marine engines. The PT6 also assisted design of the JT15D turbofan.

By 2000 the PT6A family had established a basic unscheduled removal rate of 1 per 77,000 hours,

and an IFSD rate of 1 per 250,000 hours. The latter figure has allowed for such aircraft as the Caravan, PC-12 and TBM700 to apply for single-engine IFR clearance in Canada. Australia and the USA are expected to grant similar approvals. The PT6D-114A, derived from the PT6A-114, forms the power section of the twin-engined Dual Pac (see under Soloy in USA section).

PT6A-6

Flat rated at 430 ekW; 410 kW (578 ehp; 550 shp) at 2,200 propeller rpm to 21ºC, this version received civil certification in December 1962. A total of 350 PT6A-6s were built between then and November 1965. Among aircraft powered by the PT6A-6 are the de Havilland Canada Turbo-Beaver and early DHC-6 Twin Otter Series 100.

PT6A-11

Flat rated at 394 ekW; 373 kW (528 ehp; 500 shp) at 2,200 propeller rpm to 42ºC. Piper Cheyenne I and IA, T-1040 and prototype PA-46T Malibu.

PT6A-11AG

Flat rated at 394 ekW; 410 kW (528 ehp; 550 shp) at 2,200 propeller rpm to 42ºC. Derived from the PT6A-21, cleared to use diesel fuel. Ayres Turbo-Thrush, Turbo Ag-Cat and Weatherly 620 TP.

PT6A-110

Flat rated at 374 ekW; 354 kW (501 ehp; 474.5 shp) at 1,900 propeller rpm to 38.3ºC. Dornier 123-6 Turbo-Skyservant.

PT6A-112

Flat rated at 394 ekW; 373 kW (528 ehp; 500 shp) at 1,900 propeller rpm to 56ºC. Cessna Conquest I, Reims-Cessna F 406 Caravan II.

PT6A-114

Flat rated at 471 ekW; 447 kW (632 ehp; 600 shp) at 1,900 propeller rpm to 57.7ºC. Cessna Caravan I, with single exhaust. See Soloy (USA) for Dual Pac (2 × PT6D-114A).

PT6A-114A

Flat rated at 529 ekW; 503 kW (709 ehp; 675 shp) at 1,900 propeller rpm to 46.1ºC. Cessna Caravan 208B.

PT6A-121

Flat rated at 481 ekW; 457 kW (647 ehp; 615 shp) at 1,900 propeller rpm to 32.8ºC. Piaggio P.166-DL3.

PT6A-15AG

Flat rated at 533 ekW; 507 kW (715 ehp; 680 shp) at 2,200 propeller rpm to 22ºC. Can use diesel fuel. Ayres Turbo-Thrush, Frakes Turbo-Cat, Turbo Ag-Cat D and Air Tractor AT-402/502.

PT6A-20

Flat rated at 432 ekW; 410 kW (579 ehp; 550 shp) at 2,200 propeller rpm to 21ºC, the -20 offered improved reliability and increases in maximum continuous, maximum climb and maximum cruise power ratings over the PT6A-6. The PT6A-20 was certificated in October 1965. Between then and 1974, approximately 2,400 were built to power the Beech King Air B90, Beech Model 99 Commuter Liner, prototypes of the Embraer EMB-110 Bandeirante, de Havilland Canada DHC-6 Twin Otter Series 100 and 200, James Aviation (Fletcher FU-24) conversion, Marshall of Cambridge (Grumman) Goose conversion, McKinnon G-21C and G-21D Turbo-Goose (Grumman Goose) conversions, Pilatus PC-6/B1-H2 Turbo-Porter, Pilatus PC-7 Turbo-Trainer and the Swearingen Merlin IIA (which can be re-engined with the PT6A-27).

PT6A-20A

Similar to A-20; fitted to early Beechcraft King Air C90.

PT6A-21

Flat rated at 432.5 ekW; 410 kW (580 ehp; 550 shp) at 2,200 propeller rpm to 33ºC. Mates A-27 power unit with A-20A gearbox. Beechcraft King Air C90.

PT6A-25

Flat rated at 432.5 ekW; 410 kW (580 ehp; 550 shp) at 2,200 propeller rpm to 33.8ºC. Oil system for sustained inverted flight and Beechcraft T-34C.

PT6A-25A

Some castings of magnesium alloy instead of aluminium alloy. Pilatus PC-7 and NAC Firecracker.

PT6A-25C

Flat rated at 584 ekW; 559 kW (783 ehp; 750 shp) at 2,200 propeller rpm to 30.5ºC. A-25 with A-34 hot end and A-27 first-stage reduction gearing. Embraer EMB-312 and Pilatus PC-7 Mk II.

PT6A-27

Flat rated at 553 ekW; 507 kW (715 ehp; 680 shp) at 2,200 propeller rpm to 22ºC, attained by increase in mass flow, at lower turbine temperatures than in PT6A-20. Hamilton Westwind II/III (Beech 18) conversions, Beechcraft 99 and 99A, and U-21A and U-21D (as **T74-CP-700/702**), DHC-6 Twin Otter 300, Pilatus/Fairchild Industries PC-6/B2-H2 Turbo-Porter, Frakes Aviation (Grumman) Mallard conversion, Let L-410A Turbolet, Saunders ST-27A conversion, Embraer EMB-110 Bandeirante (early)

and Harbin Y-12 II.

PT6A-28

Similar to PT6A-27, this has an additional cruise rating of 486 ekW (652 ehp) available to 21ºC and maximum cruise up to 33ºC. Beechcraft King Air E90 and A100, and 99A, Piper Cheyenne II and Embraer Xingu I.

PT6A-34

Flat rated at 584 ekW; 559 kW (783 ehp; 750 shp) at 2,200 propeller rpm to 30.5ºC, this version has air-cooled nozzle guide vanes. IAI 102/201 Arava, Saunders ST-28, Frakes Aviation (Grumman) Mallard conversion, Embraer EMB-110P1/P2 and EMB-111.

PT6A-34B

Aluminium alloy replaces magnesium in major castings. Beechcraft T-44A.

PT6A-34AG

Agricultural, certificated on diesel fuel, Frakes conversion of Ag-Cat and Ayres Turbo-Thrush. Turbo Ag-Cat, Air Tractor AT-402/502/503A and Croplease Fieldmaster.

PT6A-135

Flat rated at 587 ekW; 559 kW (787 ehp; 750 shp) at 1,900 rpm. Changed drive ratio to reduce noise; higher cycle temperatures. Beechcraft F90, Embraer 121A1 Xingu II, Piper Cheyenne IIXL, and Schafer Comanchero/Comanchero 750 conversions.

PT6A-135A

Higher thermodynamic ratings. Beechcraft F90-1, Israviation ST-50, Clark-Norman Triloader TA-3T and Dornier Composite Seastar. Selected with Hartzell reversing propeller for Khrunichev T-440 Mercury export versions.

PT6A-36

Flat rated at 584 ekW; 559 kW (783 ehp; 750 shp) at 2,200 rpm to 36ºC. Similar to -34 but higher rating. IAI 101B/202 Arava and Beechcraft C99.

PT6A-38

Derated A-41, flat rated at 597 ekW; 559 kW (801 ehp; 750 shp) to 39ºC. Beechcraft C-12A Huron.

PT6A-41

Higher mass flow, air-cooled nozzle guide vanes and two-stage free turbine. T-O rating of 673 ekW;

634 kW (903 ehp; 850 shp) at 2,000 propeller rpm, to 41ºC. Thermodynamic power 812 ekW (1,089 ehp). Beechcraft Super King Air 200 and C-12, and Piper Cheyenne III.

PT6A-41AG

For agricultural aviation. Frakes Turbo-Cat and Turbo Ag-Cat.

PT6A-42

A-41 with increase in cruise performance. Beechcraft Super King Air B200.

PT6A-42A

Thermodynamic power 814 ekW (1,090 ehp), T-O flat rating 298 kW (400 shp). Piper Malibu Meridian.

PT6A-45A

A-41 with gearbox to transmit higher powers at reduced speeds. Rated at 916 ekW; 875 kW (1,229 ehp; 1,173 shp) at 1,700 rpm to 8ºC, or to 20.5ºC with water injection. Shorts 330 and Mohawk 298.

PT6A-45B

A-45A with increased water injection. Flat rated to 11ºC dry or 28.3ºC wet.

PT6A-45R

A-45B with reserve power rating and deleted water system. Rated at 935 ekW; 892 kW (1,254 ehp; 1,197 shp) at 1,700 rpm to 22.8ºC.

T101-CP-100

A-45R for Shorts C-23A.

PT6A-50

A-41 with higher ratio reduction gear for quieter operation. T-O 875.5 ekW; 835 kW (1,174 ehp; 1,120 shp) with water at 1,210 propeller rpm up to 15ºC. DHC-7.

PT6A-60A

Uprated A-42 with new first-stage compressor, driven by turbine from PT6A-65, and gearbox from PT6A-45B, with jet flap intake and increased mass flow for high-altitude cruise. Rated at 830 ekW; 783 kW (1,113 ehp; 1,050 shp) at 1,700 rpm to 25ºC. Beechcraft Super King Air 300/350.

PT6A-60AG

Matches A-60A gas generator with power section for higher thrust at lower airspeeds, but with derated max continuous power. Intended for a new class of ag-aircraft with hoppers of 2,273-2,652 litre

(500-583 gal, 600-700 US gal) capacity.

PT6A-61

A-60 gas generator matched with A-41 power section with 2,000 rpm gearbox. T-O rating 672 ekW; 634 kW (902 ehp; 850 shp) to 46.1ºC. Cheyenne IIIA.

PT6A-62

Flat rated at 708 kW (949 shp). Pilatus PC-9 and PZL Orlik.

PT6A-64

A-67 gas generator with A-61 gearbox. Flat rated at 522 kW (700 shp) at 2,000 rpm to 63.5ºC. TBM 700.

PT6A-65B

A-65R without reserve rating. Flat rated at 931.8 ekW; 874.7 kW (1,249 ehp; 1,173 shp) at 1,700 rpm to 38.3ºC. Beechcraft 1900 and C-12J, PZL M-28 (An-28PT) and Beriev Be-32, and forward propulsion of AeroRIC Dingo.

PT6A-65R

A-45 with four-stage compressor with jet flap intake, fuel control and fuel dump. Improved hot end and exhaust duct. Reserve power 1,087 ekW; 1,026 kW (1,459 ehp; 1,376 shp) at 1,700 rpm to 27.8ºC. Alternative T-O at 975 ekW; 917 kW (1,308 ehp; 1,230 shp) at 1,700 rpm to 24ºC. Shorts 360.

PT6A-65AG

Agricultural and firefighting version of -65; T-O rating 1,030 ekW; 969 kW (1,381 ehp;1,300 shp) to 21ºC. Ayres Turbo Thrush, Air Tractor AT-802 and 802A and Croplease Firemaster.

PT6A-65AR

Reserve power 1,125 ekW; 1,062 kW (1,509 ehp; 1,424 shp) at 1,700 rpm to 27.7ºC. Shorts 360 and AMI DC-3.

PT6A-66

Flat rated at 674 ekW; 534 kW (905 ehp; 850 shp) at 2,000 rpm to 57.2ºC. Myasishchev M-102 and Piaggio Avanti with opposed-rotation gearbox.

PT6A-66A

Flat rated at 675 ekW; 634 kW (905 ehp; 850 shp) to 50.1ºC. Selected for AASI JetCruiser 500.

PT6A-67

Flat rated at 950 ekW; 895 kW (1,273 ehp; 1,200 shp) at 1,700 rpm to 46.1ºC. Beechcraft RC-12K/N/P/Q.

PT6A-67A

Flat rated at 950 ekW; 895 kW (1,273 ehp; 1,200 shp) at 1,700 rpm to 53ºC. Beechcraft Starship 2000.

PT6A-67AF

Flat rated at 1,125 ekW; 1,062 kW (1,509 ehp; 1,424 shp) at 1,700 rpm to 37.2ºC. Conair Turbo Firecat.

PT6A-67AG

Flat rated at 1,066 ekW; 1,006 kW (1,430 ehp; 1,350 shp) at 1,700 rpm to 33.3ºC. AirTractor AT-802A/AT-802AF.

PT6A-67B

A-67 modified for medium altitudes. Flat rated at 948.5 ekW; 895 kW (1,272 ehp; 1,200 shp) at 1,700 rpm to 45.0ºC. Pilatus PC-12.

PT6A-67D

A-67B with A-67R gearbox. Flat rated at 1,009 ekW; 953.8 kW (1,353 ehp; 1,279 shp) at 1,700 rpm to 46.1ºC. Beechcraft 1900D.

PT6A-67R

A-67 with reserve power rating for commuter aircraft. Flat rated at 1,125 ekW; 1,062 kW (1,509 ehp; 1,424 shp) at 1,700 rpm to 48.3ºC. Shorts 360-300 and Basler Turbo BT-67.

PT6A-68

A-67 core with 2,000-rpm gearbox, inverted-flight capability and full-authority power management; thermodynamic power 1,274 kW (1,708 shp), flat rated at 932 kW (1,250 shp) for Beech Pilatus Mk II (JPATS).

PT6A-68A

A-68 flat rated at 969 kW (1,300 shp) at 2,000 rpm to 44.9ºC. EMB-312H and 312HJ.

PT6A-68-1

Full-power 2,000-rpm gearbox rated at 1,193 kW (1,600 shp), driving Hartzell five-bladed propeller, for EMBRAER ALX.

The following data apply generally to the PT6A series:

Type

Free turbine axial-plus-centrifugal turboprop engine.

Intake

Annular air intake at rear of engine, with intake screen. Aircraft-supplied alcohol anti-icing system or inertial separation anti-icing system.

Compressor

Three axial flow stages, plus single centrifugal stage (-65 series, four axial stages). Single-sided centrifugal compressor, with 26 vanes, made from titanium forging. Axial rotor of disc-drum type, with stainless steel stator and rotor blades. The stator vanes (44 first-stage, 44 second-stage, 40 third-stage) are brazed to casing. The rotor blades (16 first-stage, 32 second-stage and 32 third-stage) are dovetailed to discs (-67 series has blisks). Discs through bolted, with centrifugal compressor, to shaft. Fabricated one-piece stainless steel casing and radial diffuser. PT6A-27: pressure ratio 6.7, mass flow 3.1 kg (6.8 lb)/s. PT6A-65: pressure ratio 10, mass flow 4.3 kg (9.5 lb)/s.

Combustion Chamber

Annular reverse-flow type of stainless steel construction, with 14 simplex burners around periphery of chamber. All versions up to A-34 have two glow plug igniters with option of two spark igniters; A-38 onwards, two spark igniters. PT6A-27 has one plug at 64º on starboard side of vertical centreline and one at 90º on port side.

Compressor Turbine

Single-stage; material varies with version, usually 58 rotor blades. Early engines, 29 NGVs; A-34 onward 14 air-cooled vanes.

Power Turbine

Models up to A-36 have single-stage axial; with 41 shrouded blades; models from A-38 onward have two-stage LP turbines. All blades have fir-tree root fixings.

Jetpipe

Single or twin pipes curved out from the front of the engine immediately behind the propeller gearbox, usually on horizontal centreline.

Mounting

Up to A-34, three-point ring suspension. A-38 onward, four-point mounting, except -50 has base mounting.

Output

(all models up to and including PT6A-41): Two-stage planetary gear train. Ratio 15:1. Rotation clockwise when viewed from rear. Drive from free turbine. Flanged propeller shaft. Plain bearings. Higher ratio reduction gears developed for PT6A-45R, -50, -60, -65 and -67.

Accessories

Mounting pads on accessory case (rear of engine) for starter/generator, hydraulic pump, aircraft

accessory drive, vacuum pump and tachometer generator. Mounting pad on the shaft-turbine reduction gear case for propeller overspeed governor, propeller constant speed control unit and tachometer generator.

Starting

Electric starter/generator on accessory case.

Control System

Bendix DP-F2 pneumatic automatic fuel control system. Pneumatic computing section, fuel metering and regulating section, gas generator governor and free turbine governor. Primary and secondary flow manifolds with seven nozzles per manifold. A-50 has DP-F3 with starting spill valve and motive flow systems; A-60 series (except -62) have Woodward 83212 hydromechanical system.

Fuel Specification

Commercial jet fuels JP-1, JP-4, JP-5, MIL-J-5624. Use of aviation gasolines (MIL-G-5572) grades 80/87, 91/98, 100/130 and 115/145 permitted for a period of up to 150 hours during any overhaul period.

Oil System

One pressure and four scavenge elements in the pump stacks. All are gear type and are driven by the gas generator rotor. Engine has an integral oil tank with a capacity of 8.75 litres (2.3 US gallons; 1.9 Imp gallons). Oil supply pressure is 5.5 bars (80 lb/sq in) on PT6A-11 to -28, 5.85 bars (85 lb/sq in) on -34 to -36, and 7.25 bars (105 lb/sq in) on -38 to -65.

Oil Specification

CPW202, PWA521 Type II (7.5 cs vis) (MIL-L-23699, MIL-L-7808 for military engines).

Dimensions

Max diameter	483 mm (19 in)
Length, excl accessories:	
PT6A-6 to -36 and -110 to -135A	1,575 mm (62 in)
PT6A-38, -41, -42, -61	1,701 mm (67 in)
PT6A-45, -60A, -60AG, -68, -68A	1,829 mm (72 in)
PT6A-50	2,133 mm (84 in)
PT6A-62, -66, -64, -66A	1,778 mm (70 in)
PT6A-65B, -67, -67A, -67B, -67D	1,880 mm (74 in)
PT6A-65R, -65AR, -65AG	1,905 mm (75 in)
PT6A-67R, -67AF, -67AG	1,930 mm (76 in)

Weight, Dry

PT6A-6	122.5 kg (270 lb)
PT6A-11, -15AG, -21, -27, -28	148.8 kg (328 lb)
PT6A-11AG	143.8 kg (317 lb)

PT6A-110, -112	151.5 kg (334 lb)
PT6A-114, -114A	158.8 kg (350 lb)
PT6A-121	151.0 kg (333 lb)
PT6A-135, -135A	153.5 kg (338 lb)
PT6A-20	130.0 kg (286 lb)
PT6A-25	160.1 kg (353 lb)
PT6A-25A	155.6 kg (343 lb)
PT6A-25C	156.9 kg (346 lb)
PT6A-34, -34AG, -36	150.1 kg (331 lb)
PT6A-34B, -135, -135A	156.0 kg (344 lb)
PT6A-38	168.3 kg (371 lb)
PT6A-41, -42	182.8 kg (403 lb)
PT6A-41AG	186.9 kg (412 lb)
PT6A-45A	196.8 kg (434 lb)
PT6A-45B	197.8 kg (436 lb)
PT6A-45R	203.2 kg (448 lb)
PT6A-50	275.3 kg (607 lb)
PT6A-60A, -60AG	215.5 kg (475 lb)
PT6A-61	194.6 kg (429 lb)
PT6A-62	205.9 kg (454 lb)
PT6A-64, -66A	207.0 kg (456 lb)
PT6A-65B, -65R	218.2 kg (481 lb)
PT6A-65AR, -65AG	220.4 kg (486 lb)
PT6A-66	213.2 kg (470 lb)
PT6A-67, -67A	229.5 kg (506 lb)
PT6A-67AF	241.0 kg (532 lb)
PT6A-67AG	235.9 kg (520 lb)
PT6A-67B	240.4 kg (530 lb)
PT6A-67R, -67D	233.5 kg (515 lb)
PT6A-68, -68A	259.5 kg (572 lb)

Performance Ratings

(S/L, static)

T-O: See under model listings

Max continuous:	
PT6A-6	392 ekW; 373 kW (525 ehp; 500 shp) at 2,200 rpm to 18ºC

PT6A-11	394 ekW; 373 kW (528 ehp; 500 shp) at 2,200 rpm to 42ºC
PT6A-11AG	432.5 ekW; 410 kW (580 ehp; 550 shp) at 2,200 rpm to 32.4ºC
PT6A-110	374 ekW; 354 kW (502 ehp; 475 shp) at 1,900 rpm to 38ºC
PT6A-112	394 ekW; 373 kW (528 ehp; 500 shp) at 1,900 rpm to 56ºC
PT6A-114	471 ekW; 447 kW (632 ehp; 600 shp) at 1,900 rpm to 57.7ºC
PT6A-114A	529 ekW; 503 kW (709 ehp; 675 shp) at 1,900 rpm to 46.1ºC
PT6A-121	481 ekW; 457 kW (647 ehp; 615 shp) at 1,900 rpm to 32.8ºC
PT6A-15AG, -27, -28	533 ekW; 507 kW (715 ehp; 680 shp) at 2,200 rpm to 22ºC
PT6A-20	432 ekW; 410 kW (579 ehp; 550 shp) at 2,200 rpm to 22ºC
PT6A-21	432.5 ekW; 410 kW (580 ehp; 550 shp) at 2,200 rpm to 33ºC
PT6A-25, -25A	432.5 ekW; 410 kW (580 ehp; 550 shp) at 2,200 rpm to 32.7ºC
PT6A-25C, -34, -34B	584 ekW; 559 kW (783 ehp; 750 shp) at 2,200 rpm to 30.5ºC
PT6A-135	587 ekW; 559 kW (787 ehp; 750 shp) at 1,900 rpm -135 to 29ºC, -135A to 34ºC
PT6A-36	586 ekW; 559 kW (786 ehp; 750 shp) at 2,200 rpm to 36ºC
PT6A-38	597 ekW; 559 kW (801 ehp; 750 shp) at 2,000 rpm to 39ºC
PT6A-41	673 ekW; 634 kW (903 ehp; 850 shp) at 2,000 rpm to 41ºC
PT6A-42, -42A	674 ekW; 634 kW (904 ehp; 850 shp) at 2,000 rpm to 41ºC
PT6A-45A, -45B, -45R	798 ekW; 761 kW (1,070 ehp; 1,020 shp) at 1,700 rpm to: -45A, 26.7ºC; -45B, 29ºC; -45R, 33ºC
PT6A-50	762 ekW; 725.5 kW (1,022 ehp; 973 shp) at 1,210 rpm to 32ºC
PT6A-60A	830 ekW; 783 kW (1,113 ehp; 1,050 shp) at 1,700 rpm to 25ºC

PT6A-60AG	830 ekW; 783 kW (1,113 ehp; 1,050 shp) at 1,700 rpm to 26.4ºC
PT6A-61	672 ekW; 634 kW (902 ehp; 850 shp) at 2,000 rpm to 46ºC
PT6A-62	751 ekW; 708 kW (1,008 ehp; 950 shp) at 2,000 rpm to 36.7ºC
PT6A-64	557 ekW; 522 kW (747 ehp; 700 shp) at 2,000 rpm to 63.5ºC
PT6A-65AG, -65AR	968 ekW; 910 kW (1,298 ehp; 1,220 shp) at 1,700 rpm to 38.3ºC
PT6A-65B	875.5 ekW; 820 kW (1,174 ehp; 1,100 shp) at 1,700 rpm to 45.6ºC
PT6A-65R	931 ekW; 875 kW (1,249 ehp; 1,173 shp) at 1,700 rpm to 38.3ºC
PT6A-66	675 ekW; 634 kW (905 ehp; 850 shp) at 2,000 rpm to 62ºC
PT6A-66A	675 ekW; 634 kW (905 ehp; 850 shp) at 2,000 rpm to 50.1ºC
PT6A-67	950 ekW; 895 kW (1,273 ehp; 1,200 shp) at 1,700 rpm to 34.6ºC
PT6A-67A	950 ekW; 895 kW (1,273 ehp; 1,200 shp) at 1,700 rpm to 53ºC
PT6A-67AF	965 ekW; 910 kW (1,294 ehp; 1,220 shp) at 1,700 rpm to 48.3ºC
PT6A-67AG	965 ekW; 910 kW (1,294 ehp; 1,220 shp) at 1,700 rpm to 33.6ºC
PT6A-67B	791 ekW; 746 kW (1,061 ehp; 1,000 shp) at 1,700 rpm to 52ºC
PT6A-67D	948 ekW; 895 kW (1,271 ehp; 1,200 shp) at 1,700 rpm to 39ºC
PT6A-67R	965 ekW; 910 kW (1,294 ehp; 1,220 shp) at 1,700 rpm to 48.3ºC
PT6A-68	988 ekW; 932 kW (1,325 ehp; 1,250 shp) at 2,000 rpm to 42.0ºC
PT6A-68A	1,027 ekW; 969 kW (1,378 ehp; 1,300 shp) at 2,000 rpm to 44.9ºC

Specific Fuel Consumption

At T-O rating:	
PT6A-11	109.4 µg/J (0.647 lb/h/ehp)
PT6A-11AG	106.3 µg/J (0.629 lb/h/ehp)
PT6A-110	111.1 µg/J (0.657 lb/h/ehp)

PT6A-112	107.6 µg/J (0.637 lb/h/ehp)
PT6A-114	108.2 µg/J (0.640 lb/h/ehp)
PT6A-15AG, -27, -28, -121	101.8 µg/J (0.602 lb/h/ehp)
PT6A-21, -25, -25A	106.5 µg/J (0.630 lb/h/ehp)
PT6A-25C, -34, -34B, -34AG	100.6 µg/J (0.595 lb/h/ehp)
PT6A-135, -135A	98.9 µg/J (0.585 lb/h/ehp)
PT6A-36	99.7 µg/J (0.590 lb/h/ehp)
PT6A-41, -61	99.9 µg/J (0.591 lb/h/ehp)
PT6A-42, -42A	101.5 µg/J (0.601 lb/h/ehp)
PT6A-45A, -45B	93.5 µg/J (0.554 lb/h/ehp)
PT6A-45R	93.4 µg/J (0.553 lb/h/ehp)
PT6A-50	94.6 µg/J (0.560 lb/h/ehp)
PT6A-60A, -60AG	92.6 µg/J (0.548 lb/h/ehp)
PT6A-62	95.8 µg/J (0.567 lb/h/ehp)
PT6A-64	118.8 µg/J (0.703 lb/h/ehp)
PT6A-65B	90.6 µg/J (0.536 lb/h/ehp)
PT6A-65R	86.5 µg/J (0.512 lb/h/ehp)
PT6A-65AR	86.0 µg/J (0.509 lb/h/ehp)
PT6A-65AG	87.2 µg/J (0.516 lb/h/ehp)
PT6A-66	104.8 µg/J (0.620 lb/h/ehp)
PT6A-67, 67A	92.4 µg/J (0.547 lb/h/ehp)
PT6A-67AF, -67R	87.9 µg/J (0.520 lb/h/ehp)
PT6A-67AG	89.5 µg/J (0.528 lb/h/ehp)
PT6A-67B, -67D	92.3 µg/J (0.546 lb/h/ehp)
PT6A-68	91.5 µg/J (0.540 lb/h/ehp)
PT6A-68A	91.6 µg/J (0.542 lb/h/ehp)

Oil Consumption

Max	0.091 kg (0.20 lb)/h

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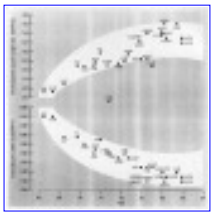
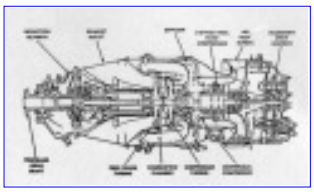
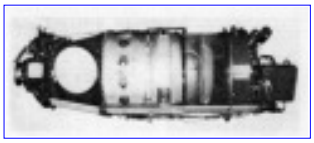


Evolution of the PT6A turboprop: power/weight ratio and specific fuel consumption. The PT6A-50 has a large double-reduction gearbox, making it heavy



Principal parts of a PT6A



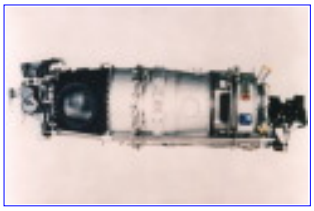
PT6A-27

PT6A-135
(2000)

PT6A-42
(2000)

PT6A-65
(2000)

Longitudinal section through PT6A-67



PT6A-68

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AERO-ENGINES - TURBOFAN, CANADA

Date Posted: 24 October 2001

Jane's Aero-Engines 11

PRATT & WHITNEY CANADA (Subsidiary of United Technologies Corporation)

P&WC PW500

By the 1990s, it was increasingly evident that there was a market for a state-of-the-art turbofan, using the overall technology P&WC understood so well, to occupy the thrust gap between the JT15D and the PW300. As before, MTU accepted responsibility for the LP turbine module, and other components, as a 25 per cent risk-sharing partner. Following discussions with Cessna, a schedule for certification was agreed, to meet which detailed engineering design began in November 1992.

Since then, two major versions of the PW500 family have been announced and more are likely to appear:

PW530A

Initial production version, with T-O rating of 13.34 kN (3,000 lb st). First engine run 29 October 1993, first flight in B-720 testbed 27 May 1994, certificated by Transport Canada 22 December 1995 and by FAA 17 April 1996 and JAA in April 1997. The certification schedule of 36 months is considered to be a record. Powers Cessna Citation Bravo. EIS June 1996 with TBO of 4,000 hours. By June 1999 over 190 engines had been delivered, with flight time over 50,000 hours, high-time engine over 1,200 hours.

PW535A

Rated at 14.94 kN (3,360 lb st). Selected for Citation Ultra Encore, first flown 9 July 1998. Certification June 1999, at which time ten engines had accumulated over 3,000 hours of testing. EIS was achieved in May 2000.

PW545A

Larger fan with added core booster and a third stage on the LP turbine. T-O rating 17.24 kN (3,876 lb st). Design started April 1994, first engine run 20 December 1994, first flight in Boeing 720 testbed 24 May 1995, and in prototype Citation Excel 29 February 1996. Certification by Transport Canada February 1997, followed by FAA on 18 July 1997. By June 1999 16 development engines had accumulated over 17,000 hours of testing, including over 800 on the B-720 testbed and more than 6,000 aboard Cessna's prototype aircraft.

By late July 2000 P&WC had delivered over 350 PW500 turbofan engines. These had logged over 105,000 hours in 109 aircraft of 93 operators in 15 countries. The high-time engine had accumulated over 2,000 hours.

Type

Two-shaft turbofan.

Fan

Single stage, IBR (Integrally Bladed Rotor) `blisk' design in titanium alloy, with 19 wide-chord snubberless blades. Abradable case with acoustic lining. Diameter (PW530A) 584 mm (23.0 in). PW545A has considerably larger blades, with diameter of 693 mm (27.3 in) with an integrally bladed booster in the extended inlet to the core. Bypass ratio (PW530A) 3.9, (PW545A) 4.0.

Compressor

Two integrally bladed axial stages followed by one centrifugal impeller, common to all currently planned versions.

Combustion Chamber

Annular, folded, reverse flow, fed by multiple curved pipes from the centrifugal-compressor diffuser. Fuel nozzles at rear surround the LP turbine case. The liner makes a second 180º bend to terminate in the cooled HP turbine nozzle ring.

HP Turbine

Single-stage, high rpm, with uncooled blades. PW545A blades are of single-crystal alloy.

LP Turbine

The PW530 has a two-stage rotor with uncooled blades, driving the fan shaft via the second-stage disc. The PW545A has a third stage of uncooled blades.

Jetpipe

Forced mixer upstream of a common nozzle from the fan duct and core.

Reverser

Both Citation Bravo and Citation Excel have NORDAM target-type reversers.

Accessories

Below the fan duct, driven by a bevel gear off the front of the HP spool. Provision for a starter/generator and hydraulic pump. The PW545A has an additional drive for an alternator.

Starting

Electric starter/generator, with dual high-energy igniters.

Control System

Advanced hydromechanical fuel control. An advanced EEC (electronic engine control) is available as an option.

Fuel Specification

JP-1, JP-4, JP-5 to CPW 204.

Oil Specification

CPW 202.

Dimensions

Length:	
PW530A	1,524 mm (60.0 in)
PW545A	1,727 mm (68.0 in)
Diameter:	
PW530A	701 mm (27.6 in)
PW545A	813 mm (32.0 in)

Weight, Dry

PW530A	278 kg (613 lb)
PW545A	347 kg (765 lb)

Performance Rating

(uninstalled)

T-O: See model listing	
Cruise (12,200 m; 40,000 ft at M0.8):	
PW530A	2.70 kN (607 lb)
PW545A	4.07 kN (915 lb)

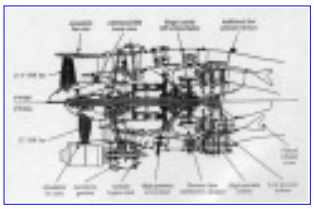
Specific Fuel Consumption

T-O rating as above:	

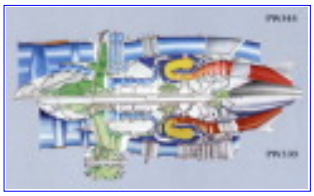
PW530A	13.45 mg/Ns (0.475 lb/h/lb st)
PW545A	12.35 mg/Ns (0.436 lb/h/lb st)

UPDATED

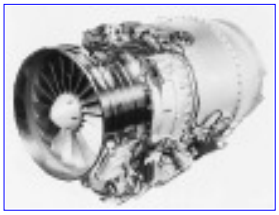
Part cutaway PW530A



Key features of the PW530A (lower half) and PW545



A later section drawing of the PW530A and PW545A



PW530A

PW545A

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AERO-ENGINES - TURBOFAN, CANADA

Date Posted: 24 October 2001

Jane's Aero-Engines 11

PRATT & WHITNEY CANADA (Subsidiary of United Technologies Corporation)

P&WC PW300

This all-new turbofan was developed to meet the requirements of mid-size long-range business jets. It was developed in partnership with MTU (DASA), a 25 per cent risk-sharing partner responsible for the LP turbine. Versions so far announced are:

PW305

T-O rating 23.24 kN (5,225 lb st). Received Canadian DoT certification in August 1990. Entered service in 1992 in Hawker 1000. Converted to 305B.

PW305A

T-O flat-rated at 20.81 kN (4,679 lb st) to 33.9ºC. Certificated December 1992. Entered service in Learjet 60 in January 1993. Flight time in June 2000 exceeded 600,000 hours on 371 engines.

PW305B

T-O rating 23.42 kN (5,266 lb st) to 23.5ºC. Certificated in January 1993, and entered service in same

month in Hawker 1000. Flight time in June 2000 exceeded 410,500 hours, excluding time as PW305. By mid-1999 TBO for all 305-family engines had been increased to 5,000 hours, with HSI at 2,500 hours.

PW306A

Major growth version, with larger fan, improved hot-end materials and lobed (forced mixer) nozzle. T-O rating 26.86 kN (6,040 lb st) to ISA + 16ºC. Certificated 22 November 1995, meeting 1996 emission standards. Selected for IAI Astra Galaxy, first flown 25 December 1997. In conjunction with Nordam, P&WC supplies the complete propulsion package including nacelle and reverser. EIS in 1999 was at TBO of 6,000 hours, with HSI at 3,000 hours.

PW306B

T-O rating 26.9 kN (6,050 lb st). Selected for Fairchild Dornier 328JET. Certificated December 1998 and to have 10,000-hour TII (threshold inspection interval) on entry to service.

PW306C

Flat rated at 26.33 kN (5,922 lb st) to ISA + 15ºC. Selected in October 1998 for Cessna Citation Sovereign. To enter service mid-2003.

PW308A

Further growth version, designed specifically for Raytheon Hawker Horizon. T-O rating 29.24 kN (6,575 lb st) to ISA + 22ºC. P&WC will supply the entire propulsion system, including Nordam nacelle and reverser. First engine run 24 June 1998. Horizon to fly in 1999, for certification in 2001 on 6,000/3,000-hour basis.

PW308B

T-O rating 32.9 kN (7,400 lb st). Selected for Fairchild Aerospace (Fairchild Dornier) 428JET. An uprated PW308 is a candidate engine for the Northrop Grumman X-47B UCAV-N (Unmanned Combat Air Vehicle-Navy), fabrication of which is expected to begin in January 2002 to enable flight testing to start in 2004.

The PW300 family are the company's most powerful jet engines, yet are claimed to be among the quietest ever built. By June 2000 more than 600 had accumulated over 1,050,000 hours with over 120 operators.

Type

Two-shaft turbofan.

Fan

Single stage, overhung ahead of front bearing without IGVs. Wide-chord titanium snubberless blades. PW305 fan diameter 778.5 mm (30.65 in), with conical spinner. PW306 fan diameter 803.9 mm (31.65 in), with short round spinner. PW308 fan diameter 843.3 mm (33.2 in). Bypass ratio (PW305/A/B) 4.3, (PW306) 4.5, (PW308B) 3.8. FPR (PW308B) 1.68.

Compressor

Four axial stages, each with an integrally bladed blisk, followed by a centrifugal stage, all made in advanced Ti alloy. Variable IGVs and first-stage stators. Core pressure ratio, (PW305/A/B) 12.9, (PW306) 12.7, (PW308B) 16.0. OPR (305A) 19.0.

Combustion Chamber

Annular, fed around diffuser periphery by ring of curved pipes. Radial fuel pipes feed 24 air-blast nozzles.

HP Turbine

Two axial stages (both stages air-cooled in the PW306, first stage only in the PW305/A/B) with rotor blades of monocrystal material highly resistant to oxidation.

LP Turbine

Three axial stages joined to fan shaft via centre disc. Two main LP shaft bearings.

Jetpipe

Full-length fan duct leading in PW306 to forced mixer, available as an option on the PW305 family. PW308, improved multilobe mixer.

Starting

Electric starter/generator and dual high-energy igniters.

Control System

DSIC dual-channel FADEC, with built-in diagnostics and auto start/relight.

Fuel Specification

JP-1, JP-4, JP-5 to CPW 204.

Oil System

Integral with gear-type pump. Capacity 8 litres (2.11 US gallons, 1.76 Imp gallons).

Oil Specification

CPW 202.

Dimensions

Diameter (all)	970.3 mm (38.2 in)
Length:	
PW305A	1,651 mm (65.0 in)
PW305B	2,070 mm (81.5 in)
PW306	1,920 mm (75.6 in)

Weight, Dry

PW305 family	450 kg (993 lb)

PW306	473 kg (1,043 lb)

Performance Ratings

T-O, see under model listing	
Cruise (12,200 m; 40,000 ft at M0.8):	
PW305A, 305B	5.04 kN (1,132 lb)
PW306A	5.87 kN (1,320 lb)

Specific Fuel Consumption

T-O:	
PW305A	10.99 mg/Ns (0.388 lb/h/lb st)
PW305B	11.07 mg/Ns (0.391 lb/h/lb st)
PW306A	11.16 mg/Ns (0.394 lb/h/lb st)
Cruise, as above:	
PW305A, 305B	19.29 mg/Ns (0.681 lb/h/lb)
PW306A	19.23 mg/Ns (0.679 lb/h/lb)

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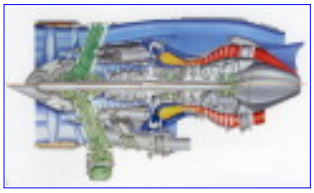
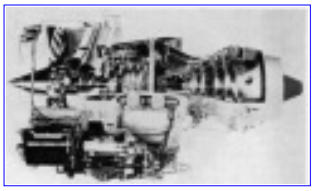
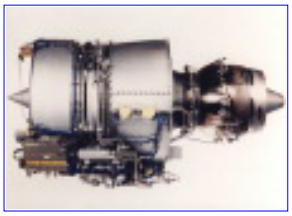


Longitudinal section through PW305 (lower half) and PW306 (upper half)



Cutaway PW305

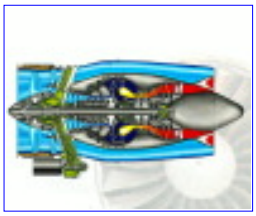
Section through PW305A (MTU contribution shaded)



PW305

PW306B

PW308B
(1998)



Longitudinal section through PW308B
(1998)

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

Date Posted: 01 May 2001

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LM - LIMING ENGINE MANUFACTURING CORPORATION

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General Manager: Yan Guangwei

With site area exceeding 100 ha (247.1 acres), more than 200,000 m^2 (2,152,850 sq ft) covered area, and a workforce of over 20,000, Liming (Daybreak) is one of the largest and most experienced aero-engine centres in China. Alternatively known as SEF (Shenyang Aero-Engine Factory), or just as The New Factory, it was set up on the basis of the old overhaul factory in 1954-57. The first product was the **WP5** turbojet, a licensed version of the Soviet Klimov VK-1F (Rolls-Royce Nene derivative). Production of the WP5 was achieved in June 1956. By 1957 the WP5A, based on the VK-1A, was in production, but in 1963, following increased demand to power the H-5, production was transferred to the Xian factory.

In 1956 SADO (Shenyang Aero-Engine Design Office) was established, later being restyled SARI. This undertook the design of the first Chinese turbojet, the **PF1A**, based on the WP5 but smaller and rated at 15.7 kN (3,527 lb st) to power the JJ-1 trainer. Design was complete in 1957 and the PF1A powered the prototype JJ-1 in July 1958, but later the requirement for the JJ-1 was dropped.

In January 2001 LM entered into a competition against CEC for the largest maintenance contract ever to be awarded by the People's Liberation Army/Air Force. This will be to maintain the AL-31F engines of Su-30MKK aircraft (see under Lyul'ka Saturn, though the engines may be manufactured by Salyut, Ufa or [according to one source] KMPA. The number of aircraft is expected to total 38, with a further batch of 40 in negotiation. To enable it to compete, LM is reported to have purchased (for US$10 million) the relevant technical manuals and software. Related engines are used by Su-27 fighters previously delivered, and also by the Chinese J-10 and J-11.

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

Date Posted: 01 May 2001

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CEC - CHENGDU ENGINE COMPANY

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Tel: (+86 28) 740 11 97
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Telex: 60132 CCDAC CN

Vice-Director: Prof Wang Zhong Quan
Project Manager: Li Xiao Tian

This company was formed in October 1958, and is also known as CEF (Chengdu Aero-Engine Factory). Most of the staff and resources came from the Shenyang Overhaul Factory, and the first task was to produce the **RD-500K** (RR Derwent derivative) turbojet for a cruise missile.

Today CEC has a site area of 137 ha (338.5 acres) and a workforce of almost 20,000. It produces the **WP6** turbojet (see LM), the **WP13** turbojet (see LMC) and components for the Pratt & Whitney **JT8D** turbofan, including combustion liners. In October 1988 SNECMA announced that it was assisting CATIC to develop the improved **WP13G** and **WP14** for later F-7 versions (see LMC). New annular combustion chambers will be produced.

In January 2001 CEC entered into a competition against LM for the largest maintenance contract ever to be awarded by the People's Liberation Army/Air Force. This will be to maintain the AL-31F engines of Su-30MKK aircraft (see under Lyul'ka Saturn, though the engines may be manufactured by Salyut, Ufa or [according to one source] KMPO). The number of aircraft is expected to total 38, with a further batch of 40 in negotiation. To enable it to compete, CEC is reported to have purchased (for US$10 million) the relevant technical manuals and software. Related engines are used by Su-27 fighters previously delivered, and also by the Chinese J-10 and J-11.

CEC is, through an agreement with AVIC, a partner in the design and manufacture of the Pratt & Whitney PW6000. With TPM of the USA it is co-producing the Pratt & Whitney FT8 industrial/marine gas turbine.

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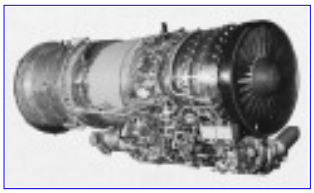
CEC WP6



CEC WP13G

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

Date Posted: 12 April 2001

Jane's Aero-Engines 10

XRA - XIAN XR AERO ENGINE COMPONENTS CO LTD

This joint-venture company was established in 1996 under the terms of an agreement between XAE and Rolls-Royce plc to expand the manufacture of engine parts in China. XRA will become the single-source manufacturer for certain aerofoils of the RR 535, Tay and Rolls-Royce GmbH BR710 and BR715.

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

Date Posted: 12 April 2001

Jane's Aero-Engines 10

XAE - XIAN AERO-ENGINE CORPORATION

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Telex: 70102 XIARO CN

President: Wang Xin Yan

This factory was built from August 1958, originally as the XEF (Xian Engine Factory). Its first major task was to produce the WP8 turbojet (Mikulin RD-3M-500A derived), deliveries of which began in January 1967 to power the H-6 (Tu-16 derived). Subsequently XEF produced various WP5 turbojets (VK-1 derived) and a small number of licensed Rolls-Royce Spey 202 augmented turbofans designated WS9. Since 1980 the Xian plant has produced large numbers of engine parts for Rolls-Royce (including over 30,000 Spey NGVs), GE, Pratt & Whitney, Allison (Rolls-Royce Corporation) and AlliedSignal (Honeywell). In June 1996 BMW Rolls-Royce (RR GmbH) became a new customer (see XRA entry).

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AERO-ENGINES - TURBOJET, CHINA, PEOPLE'S REPUBLIC

Date Posted: 12 April 2001

Jane's Aero-Engines 10

LIYANG MACHINERY CORPORATION

LMC WP7B

Derived from the WP7 (see under LM), this engine powers the J-7 fighter and JJ-7 trainer. The main change in the 7B concerned the structure and length of the afterburner. The cast air-cooled turbine blades tended to crack, with a high reject rate of castings, and burning of the rear fuselage was caused by excessive afterburner wall temperatures. The 7B was eventually certificated in 1978 and succeeded the WP7 in production in 1980. Further changes enabled TBO to be increased from 100 to 200 hours. Guizhou then eliminated the separate petrol (gasoline) starter and its tank and supply system. The engine is known as the **WP7B (M batch)** and entered production in 1982. A further modification, the **WP7B(BM)**, reduces weight by 17 kg (37.5 lb), enabling the F-7M to add drop tanks.

The following refers to the WP7B (BM):

Type

Two-shaft turbojet with afterburner.

Intake

No inlet guide vanes, first LP compressor stage overhung ahead of front roller bearing.

Compressor

Three-stage LP compressor with pressure ratio of 2.74. Five-stage HP compressor giving overall pressure ratio of 8.1. All blades inserted into discs carried on short tubular shafts.

Combustion Chamber

Can-annular, with 10 flame tubes, Nos. 1 and 6 being of a different pattern and incorporating torch igniters. Air-film liners coated on both sides with ceramic material.

HP Turbine

Single-stage with 96 inserted shrouded blades.

LP Turbine

Single-stage with shrouded blades. Outlet gas temperature 1,083ºK (810ºC).

Afterburner

Multiple gutters and double-wall liner. Multiflap nozzle driven by four hydraulic rams. Up to 40 hours operation permitted in each 200 hours overhaul period.

Performance Ratings

(S/L, static)

Max afterburner	59.82 kN (13,448 lb st)
Max dry	43.15 kN (9,700 lb st)

Specific Fuel Consumption

(as above)

Max afterburner	56.37 mg/Ns (1.99 lb/h/lb st)
Max dry	28.61 mg/Ns (1.01 lb/h/lb st)

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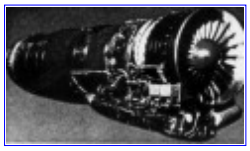

WP7B(BM)

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

Date Posted: 23 November 2000

Jane's Aero-Engines 09

SARI - SHENYANG AVIATION ENGINE RESEARCH INSTITUTE

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Fax: (+86 24) 82 06 73
Tx: 80055 SARI CN
Cables: 4391 (national), SARI (international)

Director: Hai Yide

Shenyang Aero-engine Research Institute (SARI), founded in 1961, has nearly 3,000 employees. It is responsible for research, design and development of large and intermediate size turbojet and turbofan engines and their components and systems. In the 1960s SARI modified the **WP7** engine into the **WP7A**, and transferred the engine to Liming Corporation in Shenyang and Liyang Corporation for serial production for F-7 and J-8 aircraft. From 1965 SARI developed the **WS6**, a high-thrust turbofan with afterburner. There were 10 WS6 demonstrator engines built. All performance goals were achieved, but the WS6 was not put into production (see under LM).

SARI is now developing new types of turbojet and turbofan engines. It is also working on various unspecified research programmes for Rolls-Royce Deutschland under agreements with CATIC.

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AERO-ENGINES - TURBOPROP, CHINA, PEOPLE'S REPUBLIC

Date Posted: 17 December 1999

Jane's Aero-Engines 07

DEMC - DONGAN ENGINE MANUFACTURING COMPANY

WJ5

This turboprop is based on the AI-24A (see under Ivchenko ZMKB Progress, Ukraine). Chinese development began in 1966 at ZEF and was transferred by the government to DEMC in 1968. Considerable difficulty was encountered in accurately welding the combustion chamber and broaching the turbine discs, but eventually the WJ5 was certificated in January 1977, test time then standing at 5,678 hours. All the following versions are constant-speed 15,100-rpm engines, though the WJ5A, AI and E can overspeed to 15,600 on T-O.

WJ5A

This engine was developed from 1969 to meet the needs of the SH-5 (PS-5) four-engined flying boat. Certificated after Cultural Revolution in January 1980 to power the Y7 (licensed An-24).

WJ5AI

Improved engine developed to power the Y7, Y7-100 and Y7H. Certificated in 1982.

WJ5E

Developed with the collaboration of GE Aircraft Engines. Aerodynamically redesigned and structurally simplified to give WJ5AI performance with reduced fuel burn and gas temperatures. Certificated by China Airworthy Management Bureau to FAR standards.Powers Y7, Y7-100, Y7-200B and Y7H, driving Baoding J16-G10A four-blade propeller.

WZ5

Turboshaft version for Z6 helicopter.Prototype developed by DEMC, but programme then transferred to ZARI and terminated.

WJ5AIG

Family of industrial/marine versions.

The basic description is the same as that of the AI-24 (see under Ivchenko ZMKB Progress, Ukraine).

FUEL SPECIFICATION: RP-1 (GB438-77), RP-2 (GB1788-88) OR RP-3 (GB6537-86) kerosenes.

OIL SPECIFICATION: Mixture of 75 per cent DB-45 transformer oil (GB2536-81) or HP-8 aviation oil (GB439-81) and 25 per cent HH-20 aviation oil (GB440-77) by volume.

DIMENSIONS (all versions):

Length	2,381 mm (93.74 in)
Height	1,080 mm (42.52 in)
Width, over mountings	770 mm (30.31 in)

WEIGHT, DRY (all versions):

Bare	600 kg (1,323 lb)
With all accessories	720 kg (1,587 lb)

PERFORMANCE RATINGS (S/L), static:

Max T-O:

WJ5	1,880 kW (2,521 shp), fuel flow 676 kg (1,490 lb)/h
WJ5A	2,320 kW (3,111 shp), fuel flow 812 kg (1,790 lb)/h
WJ5AI	2,130 kW (2,856 shp) to 2,500 m (8,200 ft),
	fuel flow 754 kg (1,662 lb)/h
WJ5E	2,130 kW (2,856 shp) to 3,000 m (9,843 ft),

	fuel flow 696 kg (1,534lb)/h

Rated power:

WJ5	1,545 kW (2,072 shp), fuel flow 591 kg (1,303 lb)/h
WJ5A	1,901 kW (2,549 shp), fuel flow 705 kg (1,554 lb)/h
WJ5AI	1,667 kW (2,235 shp), fuel flow 642 kg (1,415 lb)/h
WJ5E	1,670 kW (2,240 shp), fuel flow 586 kg (1,292 lb)/h

SPECIFIC FUEL CONSUMPTION (T-O, S/L):

WJ5	100 µg/J (0.591 lb/h/shp)
WJ5AI	98.7 µg/J (0.584 lb/h/shp)
WJ5E	89.4 µg/J (0.529 lb/h/shp)

WJ5E (1997)

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

Date Posted: 17 December 1999

Jane's Aero-Engines 07

DEMC - DONGAN ENGINE MANUFACTURING COMPANY

MANUFACTURER DETAILS

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Fax: (+86 451) 650 22 66
Tx: 87131 HDEC CN
Cables: 0021

GENERAL MANAGER: Song Jingang
CHIEF ENGINEER: Feng Yongchong

Also known as HEF (Harbin Engine Factory), this establishment was founded in 1948 and employs more than 12,000. Its first product was the 1,268 kW (1,700 hp) **HS7**, a 14-cylinder radial piston engine based on the Soviet Shvetsov ASh-82V. In parallel, in 1957-59, a few ASh-21 engines were made, but only the HS7 went into production, for the Z-5 helicopter. In the late 1950s there was a need for a better engine for the Il-12, Il-14, Tu-2 and Curtiss C-46, with better altitude performance. The result, produced from 1962 until 1980, was the **HS8**, which combined the main body and supercharger of the HS7 with the reduction gear of the ASh-82T. The HS8 is rated at 1,380 kW (1,850 hp).

For many years DEMC's biggest programme has concerned the WJ5 described below. A further

important task is production of the JS9 main gearbox, transmission shafting and tail gearbox of the Z9 (Dauphin) helicopter. Other products include car engines and rail wagon accelerator/retarders.

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AERO-ENGINES - TURBOJET, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 04

SPWAEC

MANUFACTURER DETAILS

In March 1998, SAEC (South Aero-Engine Co, see above) and Pratt & Whitney Canada announced the formation of Southern Pratt & Whitney Aero-Engine Company Ltd. The new company will manufacture gas-turbine engine components for P&WC. Total investment in the joint venture is over US $27 million. SAEC has a 51 per cent ownership share, and the new company is located in Zhuzhou. The partnership is the third joint venture between Pratt & Whitney and AVIC (Chinese) engine factories, and the 20th between United Technologies and Chinese partners

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AERO-ENGINES - TURBOJET, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 04

SMPMC - SOUTH MOTIVE POWER AND MACHINERY COMPLEX

MANUFACTURER DETAILS

PO Box 211, Zhuzhou, Hunan 412002
Tel: (+86 733) 211 51
Fax: (+86 733) 242 20
Tx: 995002 CHENF CN
Cables: 2820

SAEC (South Aero-Engine Company)

MANUFACTURER DETAILS

Address as SMPMC

GENERAL MANAGER: Wu Shenduo

With a covered area of nearly 300,000 m^2 (3,230,000 sq ft) and workforce of over 10,000, SMPMC is

one of the larger AVIC establishments. Until 1983 its aero-engine division, SAEC, was known as ZEF (Zhuzhou Aero-Engine Factory), and in 1951 was set up as the first aero-engine factory in China. Its first product was the Soviet Shvetsov **M-11FR** radial piston engine rated at 119 kW (160 hp), the first three being completed in July 1954. Mass production followed. To meet the needs of the Y-5 (licensed An-2) ZEF began in September 1956 to work on the **HS5** (licensed ASh-62IR). Over 2,600 of these 746 kW (1,000 hp) radial piston engines were produced by 1986, some being installed in CAAC Li-2s.

Lacking the chosen Praga Doris B engine to power the CJ-6 trainer, the Soviet Ivchenko (ZMKB Progress) AI-14R radial piston engine was produced as the **HS6**. Rated at 191 kW (260 hp), the HS-6 entered production in June 1962, about 700 being produced. To improve performance, especially at altitude, ZEF increased rpm, compression ratio and supercharger gear ratio. The result, in 1963, was the **HS6A**, with T-O power increased to 212.5 kW (285 hp). About 3,000 were made by 1986. In 1975, the engine was again modified to power the Y-11; rpm were increased and the reduction gear strengthened. The resulting **HS6D**, with power of 224 kW (300 hp), was certificated in August 1980. The **HS6E**, for the NAMC Haiyan, has a further increased compression ratio and modifed exhaust valves and reduction gear, raising output to 261 kW (345 hp). In 1990, the simplified **HS6K** was certificated at 298 kW (400 hp) and is intended as the future engine of the N-5A. Experimental models, in the 1963-70 period, were the turbocharged **HS6B** and the **HS6C** for helicopters, used in the 701 and Yan'an II helicopters.

Work on gas turbines began in January 1965 in support of the development by BIAA (Beijing Institute of Aeronautics and Astronautics) of the **WP11**. This simple turbojet, rated at 8.3 kN (1,874 lb st), powered the WZ-5 unmanned reconnaissance vehicle. The WP11 first ran in June 1971 and was certificated in 1980, manufacture then being transferred to ZEF. In September 1965, ZEF was selected to develop the **WJ5** turboprop, but this was transferred in 1968 to HEF (see DEMC).

In 1969, ZEF was ordered to develop the **WJ6** turboprop, based on the Soviet Ivchenko (now Progress, Ukraine) AI-20M, to power the Y-8. Testing started in 1970, but various problems delayed certification until January 1977. Further changes (for example to compressor vane angle, igniter and lubrication clearances) resulted in TBO being raised in stages from 300 to 3,000 hours. T-O power was 3,169 ekW (4,250 ehp) and weight 1,200 kg (2,645 lb). In 1977 work began on the **WJ6A** to power the Y-8C with a pressure cabin and greater payload. By using air-cooled blades and raising the rpm this engine was successfully run in 1983 at 3,393 ekW (4,550 ehp). SAEC still produces this engine.

In 1980, ZEF began the assembly and test of the **WZ8** (Turbomeca Arriel 1C) for the Z-9 helicopter. SAEC gives the output as 522 kW (700 shp) for a weight of 118 kg (260 lb). An all-Chinese WZ8 ran in 1985, and resulted in major technical upgrades at Zhuzhou (the high-voltage igniter box was the only imported part). As part of the offset, 40 accessory gearboxes were supplied to France.

SAEC also produces industrial and marine gas turbines and solid rocket motors for air-to-air missiles. It is a 51 per cent shareholder in SPWAEC, see next entry.

South Aero-Engines WJ6A (Günter Endres/Jane's) (1998)

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AERO-ENGINES - TURBOJET, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 04

SAMP - SHANGHAI AERO-ENGINE MANUFACTURING PLANT

MANUFACTURER DETAILS

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Fax: (+86 21) 665 14 82
Tx: 33136 SHAIR CN
Cables: 5834

DIRECTOR: Shen Huansheng

This factory was built in 1971-74 and was originally the SAF (Shanghai Aero-Engine Factory). It has a covered area of 56,191 m^2 (604,855 sq ft) and workforce of 2,000. Apart from various non-aero engines and components its main development effort was the **WS8** turbofan, to power the Y-10. Work proceeded quickly and the first engine went on test in June 1975. Eight engines were built, one running a 1,000 hour test, one a 150 hour certification test and one was flight tested, making eight flights totalling 22 hours. A front-fan engine with a short bypass duct, the WS8 was rated at 80 kN (18,000 lb st). About 17 per cent was titanium, and new techniques included anti-corrosion coating with cadmium and nickel; graphite varnish of titanium parts; and aluminised siliconising of turbine blades. The Y-10 was not put into production and the engine had no other application.

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AERO-ENGINES - TURBOJET, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 03

LIYANG MACHINERY CORPORATION - LMC

LMC WP13

The next major product of LMC was the **WP13**. Though this has some features in common with the Gavrilov R-13 (see Soyuz, Russia) it is a Chinese development, based on experience with the WP7. A two-spool afterburning turbojet, it was developed to power the J-7 III and J-8 II. Compared with the WP7 the air flow is increased, work per stage improved, pressure ratio raised (the HP spool having new blades) and surge margin doubled. New titanium alloys were used for the compressor discs and blades, and in a major development two more new titanium alloys were used for the cast compressor casings. WP13 development began in 1978, and it was decided to make the engine a 50/50 joint project with CEC. Both factories tested engines, and certification was gained in 1985. Further development introduced air-cooled HP turbine blades and modifications to the combustion chamber and afterburner, the afterburner of the **WP13A II** being longer.
DIMENSIONS:

Length overall:	
WP13	4,600 mm (181.1 in)
WP13A II	5,150 mm (202.75 in)
Diameter: both	907 mm (35.71 in)
Max height: WP13	1,085 mm (42.72 in)

WEIGHT, DRY:

WP13	1,211 kg (2,670 lb)
WP13A II	1,201 kg (2,648 lb)

PERFORMANCE RATINGS(S/L, static):

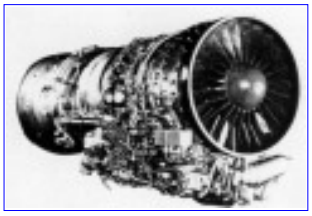
Max afterburner:	
WP13	64.73 kN (14,550 lb st) at 11,156 LP rpm
WP13A II	65.9 kN (14,815 lb st)
Max dry:	
WP13	40.21 kN (9,039 lb st) at 11,156 LP rpm
WP13A II	42.7 kN (9,590 lb st)

SPECIFIC FUEL CONSUMPTION (as above):

Max afterburner:	
WP13	63.73 mg/Ns (2.25 lb/h/lb st)
WP13A II	62.32 mg/Ns (2.20 lb/h/lb st)
Max dry:	
WP13	27.19 mg/Ns (0.96 lb/h/lb st)
WP13A II	28.04 mg/Ns (0.99 lb/h/lb st)

OVERHAUL LIFE:

WP13	500 h (total service life 1,500 h)
WP13A II	300 h (including up to 90 h in afterburner)



LMC WP13A II (afterburner not fitted)

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 03

LIYANG MACHINERY CORPORATION - LMC

MANUFACTURER DETAILS

PO Box 5, Pingba, Guizhou 561102
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Cables: 4099, 4101 PINGBA

GENERAL MANAGER: Hu Wenqin

With a covered area of 750,000 m^2 (8,073,200 sq ft), and a workforce of about 10,000, LMC is also known as GEF (Guizhou Aero-Engine Factory). The associated GADRI undertook the development of the **WP7B** afterburning turbojet, the programme being transferred to LMC because Shenyang (LM) was overloaded.

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AERO-ENGINES - TURBOFAN, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 03

LIMING ENGINE MANUFACTURING CORPORATION - LM

LM WS6

Also in 1964 a meeting was held to select an engine for a new indigenous fighter. The choice fell on an augmented turbofan, the **WS6**. In collaboration with SARI, drawings were produced in 1964-66, and by 1969 two prototype engines had been made. Then the cultural revolution delayed the programme by about 10 years, but after 1978 eight more engines were built. A two-spool engine with air-cooled HP blades, it reached design figures in 1980. Augmented thrust was increased to 122.1 kN (27,450 lb st) in 1982, but by then the associated fighter programme had been cancelled.
TYPE: Two-shaft augmented turbofan for supersonic applications.
FAN: Three stages, First stage transonic. No inlet guide vanes. Mass flow 155 kg (342 lb)/s. Bypass ratio 1.
COMPRESSOR: Eleven stages, with variable inlet vanes and fifth-stage bleed. Pressure ratio 6.78 at 9,400 rpm. Overall pressure ratio 14.44.
COMBUSTION CHAMBER: Can-annular.
TURBINES: Two-stage HP, inlet temperature 1,177ºC. Two-stage LP.
AFTERBURNER: Five circular gutters and six fuel-injection zones. Maximum temperature 1,527ºC.
NOZZLE: Outer nozzle only, multiflap type.
DIMENSIONS:

Length	4,654 mm (183.2 in)

Diameter (nozzle)	1,370 mm (53.94 in)
WEIGHT, DRY	2,100 kg (4,630 lb)

PERFORMANCE RATINGS (S/L):

Max T-O	122.1 kN (27,445 lb st)
Max dry	71.1 kN (15,991 lb st)

SPECIFIC FUEL CONSUMPTION:

Max T-O	64.01 mg/Ns (2.26 lb/h/lb st)
Max dry	17.56 mg/Ns (0.62 lb/h/lb st)

WS6 afterburning turbofan

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AERO-ENGINES - TURBOJET, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 03

LIMING ENGINE MANUFACTURING CORPORATION - LM

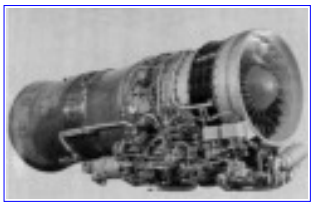
LM WP7

For the J-7 fighter the Shenyang factory produced the Tumanskii (Soyuz) R-11F-300 afterburning turbojet under licence. Designated **WP7**, this M2 engine was an even greater challenge as many key Soviet documents were not supplied, and 1,097 documents had errors or omissions. Eventually, using indigenous materials, the first WP7 went on test in October 1965. Dry and afterburning ratings were 38.2 kN (8,598 lb st) and 56.4 kN (12,676 lb st) respectively. By 1970 production of this engine was transferred to LMC. Shenyang continued to introduce improvements, and a stall flutter problem was solved by using 24 larger blades in the first stage of the compressor instead of 31, while other changes were made to the HP turbine disc, bearing lubrication and afterburner nozzle flap design.

In 1964 a decision had to be made on how to power the J-8 fighter. No engine in the 120 kN (12 tonne) class could be produced in time, but Rong Ke, deputy director of Beijing Aeronautical Materials Institute, undertook to produce air-cooled blades within a year to allow two uprated WP7 engines to be used. In May 1965 the resulting engine was authorised as the **WP7A**. After testing against forged blades with three large cooling holes the decision was taken to use nine-hole cast blades. Dry and afterburning ratings were established at 43.14 kN (9,698 lb st) and 59 kN (13,265 lb st) respectively. These engines powered the J-8 on its first flight in July 1969, and were certificated in June 1982. Subsequently LMC developed the **WP7B** (see LMC) and **WP7C**. The latter powers J-7 and J-7 II aircraft. Thrust ratings

are: maximum 60.6 kN (13,623 lb st), maximum dry 42.65 kN (9,588 lb st) and normal 33.54 kN (7,540 lb st). TBO is 300 hours. Latest known variant is the **WP7F**, which powers the J-7E; dry rating is 44.13 kN (9,921 lb st) and with afterburning 63.74 kN (14,330 lb st).

A description of the original R-11 family of engines appears under Soyuz (Russia).



WP7C (afterburner not fitted)

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AERO-ENGINES - TURBOJET, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 03

LIMING ENGINE MANUFACTURING CORPORATION - LM

LM WP6

In early 1958 Soviet documents arrived for licence production of the Tumanskii (Soyuz) RD-9BF-811. This single-shaft axial turbojet with afterburner proved a major challenge, but the Shenyang **WP6** was first tested at the end of 1958. Tests were not successful, but following improvements in quality control trial production restarted in 1961. Subsequently several thousand, with progressive upgrades, have been made for the J-6, JJ-6 and Q-5. The **WP6A** for the Q-5 I attack aircraft, has a variable inlet stator stage and increased turbine temperature. This engine was certificated in August 1973. A further variant was the **WP6B** for the J-12, not made in quantity.

The following is a description of the WP6A. This differs in detail from the original Tumanskii RD-9BF family, none of which is currently flying:

TYPE: Single-shaft turbojet with afterburner.

INTAKE: Aluminium-alloy casting with four long-chord radial struts supporting front stub shaft running in front bearings. Hot-air anti-icing of struts and fixed bullet fairing.

COMPRESSOR: Nine-stage axial. Variable IGVs, but remaining stators fixed. Steel rotor blades dovetailed into intersecting welded rings. Rear end carried by a cone running in main thrust ball bearing. Longitudinally split case. Mass flow 46.2 kg (101.85 lb)/s. Pressure ratio 7.44.

COMBUSTION CHAMBER: Minimum-diameter can-annular type with 12 flame tubes, each with a single spill-type fuel nozzle and terminating in a 30º segment of turbine inlet. Two igniters fed from

starting tank. Bolted connection to turbine case.
TURBINE: Two-stage, with rotor blades inserted into flat discs without central hole. Front disc bolted to tubular shaft running in cooled and oil-fed rear bearing and splined to rear end of compressor. TGT 902ºC.
AFTERBURNER: Long and untapered, with main starting burner in centre of turbine rear cone and single flameholder gutter ring around rear of cone. Ten adjustable nozzle flaps positioned by four hydraulic actuators. No separate jetpipe liner.
ACCESSORIES: Grouped on wheelcase above compressor, driven by bevel gear from front stub shaft and radial shaft in vertical intake strut. Starter/generator projects ahead of engine.
DIMENSIONS:

Length	5,483 mm (215.9 in)
Max height	950 mm (37.4 in)
Diameter	668 mm (26.3 in)
WEIGHT, DRY:	725 kg (1,598 lb)

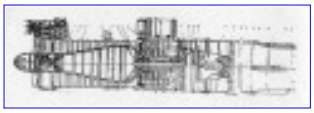
PERFORMANCE RATINGS (S/L, static):

Afterburner: WP6A	36.78 kN (8,267 lb st)
WP6B	39.72 kN (8,929 lb st)
Max dry: WP6A	29.42 kN (6,614 lb st)
Normal: WP6A	24.03 kN (5,401 lb st)
WP6B	24.51 kN (5,511 lb st)

SPECIFIC FUEL CONSUMPTION (WP6A):

Afterburner	0.163 kg/h/N (45.24 mg/Ns; 1.597 lb/h/lb st)
Max	0.1 kg/h/N (27.76 mg/Ns; 0.980 lb/h/lb st)
Normal	0.099 kg/h/N (27.48 mg/Ns; 0.970 lb/h/lb st)

WP6A afterburning turbojet (1996)



Longitudinal section through WP6A turbojet (afterburner shown shortened) 1: variable inlet stator, 2: stator vanes, 3: compressor case, 4: air bleed band, 5: bleed actuating cylinder, 6: rear load relief cavity, 7: centre bearing, 8: starting igniter, 9: stage 1 nozzle, 10: turbine rotor, 11: stage 2 nozzle, 12: quick-release ring, 13: diffuser, 14: quick-release ring, 15: front flange, 16: case, 17: shroud, 18: bracket, 19: actuating cylinder, 20: adjustable flap flange, 21: actuator and rod heat shield, 22: nozzle adjusting ring, 23: copper plate, 24: flap, 25: centring pin, 26: cylinder cowl, 27: clamp strip, 28: rear bearing, 29: oil jet, 30: flame tube, 31: compressor, 32: front load relief cavity, 33: front bearing, 34: front case

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 03

CHANGZHOU LAN XIANG MACHINERY WORKS - CLXMW

MANUFACTURER DETAILS

PO Box 37, Changzhou, Jiangsu 213022
Tel: +86 519 540 5131, 540 5392
Cables: 5046

DIRECTOR: Tian Taiwu

CLXMW has a payroll of over 5,000. Its main product is the **WZ6** turboshaft and the **WZ6G** industrial derivative. Rated at 1,106.25 kW (1,480 shp) and with a dry weight of 315 kg (694 lb), the WZ6 is derived from the Turbomeca Turmo IIIC. Work began in 1975, testing occupied 1980-82 and WZ6 engines first flew in a Z-8 helicopter in 1986.

Cutaway drawing of WZ6 (1996)

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 03

CHINA NATIONAL AERO-ENGINE CORPORATION - CAREC

MANUFACTURER DETAILS

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In addition to the engines described hereafter, the Beijing University of Aeronautics and Astronautics Propulsion Department produces the **WP11.** This small turbojet has an axial+centrifugal compressor (T-O mass flow 13.52 kg; 29.8 lb/s) and resembles an uprated Turbomeca Marboré. T-O thrust is 8.34 kN (1,873 lb st) at 22,000 rpm, dry weight 191 kg (421 lb), length 1,140 mm (44.9 in) and diameter 567 mm (22.3 in). It is intended for propulsion of unmanned vehicles at up to 18,000 m (59,055 ft).

In the past 30 years China has enormously expanded its aero-engine industry. Today there are eight major engine manufacturing centres, five factory-managed design institutes and four engine research and design institutes. Over 48,000 engines of 25 types have been manufactured for the air force and navy, 756 engines of 10 types manufactured for CAAC, and smaller numbers for export. Nine types of large solid and liquid rocket engines have also been manufactured. Most factories are seeking foreign orders or partners.

The following survey is by alphabetical order in English of the design or manufacturing organisations which are actually producing engines.

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

Jane's Aero-Engines 03

CHINA NATIONAL AERO-TECHNOLOGY IMPORT AND EXPORT CORPORATION - CATIC

MANUFACTURER DETAILS

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Tx: 22318 AEROT CN

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EXECUTIVE VICE-PRESIDENT: Tang Xiaping
DIRECTOR, PUBLIC RELATIONS: Bi Jianfa

China's former Ministry of Aero-Space Industry was abolished in 1993 and AVIC was created (on 26 June 1993) as an economic entity to develop a market economy and expand international collaboration in aviation programmes. The CATIC Group was formed 26 August 1993, with CATIC (founded January 1979) as its core company, to be responsible for import and export of aero and non-aero products, subcontract work and joint ventures.

The total workforce in Chinese aerospace is about 570,000, but most plants also produce non-aerospace items.

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AERO-ENGINES - MANUFACTURER, CHINA, PEOPLE'S REPUBLIC

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AVIATION INDUSTRIES OF CHINA - AVIC

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Fax: +86 10 401 3648

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VICE-PRESIDENTS:
Wang Ang
Zhang Hongbiao
Zhang Yanzhong
GENERAL MANAGER: Mao Dehua
CHIEF ENGINEER: Yan Huif

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AERO-ENGINES - TURBOPROP, CZECH REPUBLIC

Date Posted: 22 March 2002

Jane's Aero-Engines 02

WALTER AS

WALTER M601

This turboprop was originally designed to power the L-410 local-service transport. The first engine ran in 1967, at a rating of 410 ekW (550 ehp). Since then, the following versions have been developed:

M601A

Initial production version, developed from 1968. Powers L-410M, which entered Aeroflot service in early 1976. No longer supported.

M601B

First major series version, no longer in production. Powers L-410UVP and L-410MA.

M601D

Increased power and longer TBO. Entered production 1982. M-601D powered first L-410UVP and PZL-106BT-601 prototype. M601D-1 powers PZL-106 BT, TBO 1,500 h or 5,300 cycles. M-601D-2 powers Do 28-G92 and SMG-92 Turbo-Finist, TBO 1,500 h, 4,500 cycles. M601D-11 is specially equipped for ag-aviation; D-11NZ powers Fletcher FU-25, TBO 1,500 h, 5,300 cycles or up to 22,500

take-offs.

M601Z

Drive for auxiliary piston compressor and take-off shaft for spraying/dusting installation. Entered production 1983 to power Z-137T. All the above drive Avia-Hamilton V8.508 three-blade propeller.

M601E

Drives VJ8.510 five-blade propeller and alternator for anti-icing propeller and windscreen. Alternative propellers VJB.508E three-blade reversing or McCauley three-blade single-acting. TBO without HSI (engines produced before 1 January 1998) 2,000 h or 2,250 cycles or five calendar years, whichever is least; later engines 3,000 h, 3,300 cycles. Received Russian, Swedish, Czech and (M601E-11) FAA certification. M-601E powers L-410UVP-E, M-601E-11 powers Air Tractor, Grumman/Schweizer Ag-Cat and Ayres S2R Turbo Thrush; M-601E-11A powers King Air C90 (V8.510 five-blade propellers) and, undergoing certification, Piper Malibu.

M601T

Fully aerobatic version, TBO 1,000 h. V8.510 propeller. Powers PZL-130TM and -130TB Orlik.

M601F

Received Czech, Russian and FAA certification. TBO without HSI 3,000 h or 3,300 cycles or five years, whichever is least. Powers L-420 (F-21), Ae-270W (F-34), PZL-106 BT (F-33), M-101 Gzhel (F-22 and -32) and M-103 Oka (F-33). Selected for Russian (non-export) Khrunichev T-440 Mercury. In November 1998 an agreement was signed in Kuala Lumpur under the terms of which engines would be supplied by Walter for M-101T aircraft co-produced by Myasishchev and Malaysia.

By 2002 more than 4,500 M601 engines had been delivered, flying 14,500,000 hours. In 2002 different versions were serving more than 200 operators in over 50 countries.

Type

Free-turbine turboprop.

Intake

Annular, at rear (reverse flow engine).

Compressor

Two axial stages of stainless steel, plus single centrifugal stage of titanium. Pressure ratio (601 B) 6.4, (601 D) 6.55, (601 E, F, T) 6.65, at 36,660 rpm gas generator speed. Air mass flow (601 B) 3.25 kg (7.17 lb)/s, (601 D) 3.55 kg (7.83 lb)/s, (601 E, F, T) 3.6 kg (7.94 lb)/s.

Combustion Chamber

Annular combustor with rotary fuel injection and low-voltage ignition.

Compressor Turbine

Single stage with solid blades; inlet temperature 957~C.

Power Turbine

Single stage.

Output

Reduction gear at front of engine with drive from free turbine. Reduction ratio 14.9. Typical propeller speed 1,900 rpm.

Starting

LUN 2132-8 8 kW electric starter/generator. Starting cycle controlled automatically.

Control System

Low-pressure regulator. M601D-1, F-33, T and Z have emergency feathering, others have autofeather. Torque, shaft speeds and turbine inlet temperature controlled by limiters.

Fuel Specification

PL-6, PL-7, PSM-2, RT, TS-1 and Jet A and A-1 kerosene.

Oil System

Pressure gear-pump circulation. Integral oil tank.

Oil Specification

B3V synthetic oil or Aeroshell 500, 555, 560, Mobil Jet II, Exxon 2380, Castrol 599.

Contract Price

Varies, because most sales are of used engines with zero life and full warranty.

Dimensions

Length: 601D	1,658 mm (65.27 in)
601B, E, Z, F, T	1,675 mm (65.94 in)
Width	590 mm (23.23 in)
Height	650 mm (25.59 in)

Weight, Dry

601B, D (except D-11)	193 kg (425.5 lb)
601Z	197 kg (434.3 lb)
601D-11, E	200 kg (441 lb)
601F, T	202 kg (445 lb)

Performance Ratings

(T-O)

601B	515 kW (691 shp)

601D (except D-11)	540 kW (724 shp)
601D-11	450 kW (603 shp)
601E, T	560 kW (751 shp)
601Z	382 kW (512 shp)
601F	580 kW (778 shp)
Max continuous:	
601 D, E	490 kW (657 shp)
601F	500 kW (670 shp)

Specific Fuel Consumption

(T-O)

601B	110.8 Ág/J (0.656 lb/h/ehp)
601D (except D-11)	110.5 Ág/J (0.654 lb/h/ehp)
601D-11	114.9 Ág/J (0.68 lb/h/ehp)
601E, T	109.7 Ág/J (0.649 lb/h/ehp)
601F	106.9 Ág/J (0.633 lb/h/ehp)
601Z	135.8 Ág/J (0.804 lb/h/ehp)

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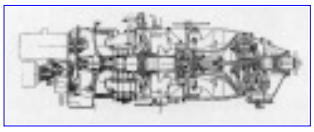
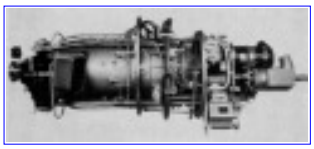
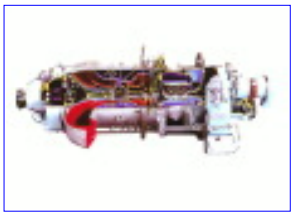


Longitudinal section through M601E



M601E



Cutaway M601F (1998)

M601F with VJ8.510 propeller (1998)

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AERO-ENGINES - MANUFACTURER, CZECH REPUBLIC

Date Posted: 22 March 2002

Jane's Aero-Engines 02

WALTER - WALTER AS

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General Manager: Dr-Ing Václav Vanek
Marketing Manager: Dr-Ing Michal Ptacnik

Walter operates the main aero-engine establishment in the Czech Republic. It has delivered over 17,000 piston engines, 16,000 turbojets and more than 4,500 turboprops. The Walter company was renamed Motorlet but reverted to its original title in March 1995, although the Walter name was used throughout as a trademark.

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AERO-ENGINES - TURBOJET, CZECH REPUBLIC

Date Posted: 22 March 2002

Jane's Aero-Engines 02

WALTER - WALTER AS

WALTER (MOTORLET) M701

This turbojet was the first gas turbine designed in Czechoslovakia. It was produced to power the C-29 (later L-29 Delfin) military trainer, the design team being led by Jirí Rada. Design started in 1955, the engine first ran in September 1958, flight testing was carried out with an Il-28 jet bomber testbed, and the engine was qualified for production in September 1961. Deliveries from the Motorlet plant at Jinonice reached 4,880 in 1974, 8,750 in 1986 and terminated at about 9,020 in 1989. Engines still in use are supported by Walter.

All models of the M701 have the same ratings and differ mainly with regard to the TBO, as indicated by their individual designations. The TBOs for the M701-b150, M701-c150, M701-c400 and M701-c500 are respectively 150, 250, 400 and 500 hours. The M701-c250 introduced flame tube and turbine improvements, and the M701-c400 and 500 have further improvements in turbine design.

Type

Single-shaft centrifugal turbojet.

Intake

Annular, with central bullet fairing. De-icing by engine-bleed air.

Compressor

Single-stage centrifugal type. Pressure ratio 4.3. Mass flow 16.9 kg (37.25 lb)/s at 15,400 rpm.

Combustion Chamber

Seven inclined straight-flow chambers, interconnected by flame channels. Two igniter plugs in Nos 2 and 7 chambers.

Turbine

Single-stage axial-flow type, with 47 stator and 61 rotor blades. Gas temperature at turbine entry 890ºC, after turbine 680-700ºC.

Jetpipe

Fixed-cone type.

Accessories

Drives on engine front casing to fuel pump, 28 V generator, hydraulic pump and tachometer. One spare drive.

Starting

LUN-2259 electric starter.

Control System

Fuel pump of the LUN 6201.03 multiplunger type. Barometric pressure control acts on servo-mechanism to vary fuel delivery according to altitude and speed. High-pressure shut-off cock. Max fuel pressure 85 kg/cm^2 (1,200 lb/sq in).

Fuel Specification

PL-4 to TPD-33.01960 standard, T-1 to GOST-4138-49 standard, or other similar fuels.

Oil System

Wet sump type. Sump at bottom of front case. One three-stage gear-type pump. Sump capacity 3.5 litres (6 Imp pints). Normal oil supply pressure 2.5 kg/cm^2 (35.5 lb/sq in).

Oil Specification

OLE-TO to TP 200/074-59 standard, or GOST 982-53, later MS-8P or MK-8P.

Dimensions

Width	896 mm (35.28 in)
Height	928 mm (36.53 in)
Length overall	2,067 mm (81.38 in)

Weight, Dry

	330 kg (728 lb)

Performance Ratings

Max T-O	8.79 kN (1,962 lb st) at 15,400 rpm
Normal	7.90 kN (1,764 lb st) at 14,950 rpm
Max cruise	7.11 kN (1,587 lb st) at 14,500 rpm

Specific Fuel Consumption

At normal power	32.28 mg/Ns (1.14 lb/h/lb st)

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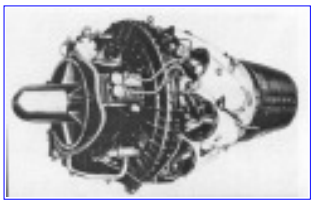
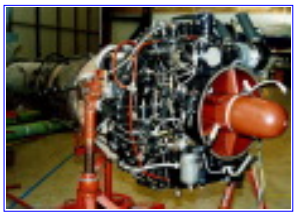


M701



M701 (2002)

Cutaway M701 (2002)

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AERO-ENGINES - TURBOPROP, CZECH REPUBLIC

Date Posted: 22 March 2002

Jane's Aero-Engines 02

WALTER AS

WALTER M602

This engine was developed to power the L-610, which first flew on 28 December 1988. Full production launch awaited, though 34 engines had run by 1993. The L-610 was expected to be certificated with this engine in 1990, but the only certificated version is powered by the imported General Electric CT7-9.

M602A

Basic version, on test for Czech Air Force.

M602B

Growth version for advanced L-610. On-condition maintenance.

Type

Three-shaft turboprop.

Intake

At front, S duct from chin inlet passing up behind reduction gear.

Compressors

LP centrifugal, pressure ratio 4.15 at 25,000 rpm. HP centrifugal, pressure ratio 3.133 at 31,489 rpm. Overall pressure ratio 13. Mass flow 7.33 kg (16.16 lb)/s.

Combustion Chamber

Short annular reverse flow with 14 simplex nozzles and low-voltage semiconductor ignition.

Compressor Turbines

Single-stage HP, single-stage LP.

Power Turbine

Two-stage, 16,600 rpm.

Control System

LP electrohydraulic regulator and electronic limiter.

Output

Double spur reduction, ratio 12.58.

Starting

LUN 5363-8 pneumatic.

Fuel Specification

T-1, TS-1, RT, Jet A-1.

Oil System

Pressure gear-pump circulation, integral oil tank and cooler.

Oil Specification

B3V, AeroShell 500, 550.

Dimensions

Length:	
M602A	2,869 mm (105.08 in)
M602B	2,285 mm (89.96 in)
Width: M602A, B	753 mm (29.65 in)
Height:	
M602A	872 mm (34.33 in)
M602B	852 mm (33.54 in)

Weight, Dry

M602A	570 kg (1,257 lb)
M602B	490 kg (1,080 lb)

Performance Ratings

(S/L)

M602A:	
T-O	1,360 kW (1,824 shp) at 1,320 propeller rpm
Max continuous	1,200 kW (1,608 shp)
Cruise	700 kW (938 shp)
M602B:	
T-O	1,500 kW (2,012 shp) to 40ºC at 1,200 propeller rpm
Max continuous	1,500 kW (2,012 shp) ISA

Specific Fuel Consumption

(T-O)

M602A	97.2 µg/J (0.575 lb/h/shp)
M602B	84.1 µ/J (0.498 lb/h/shp)

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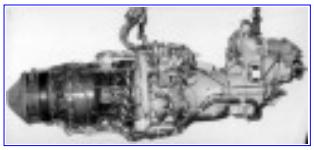

M602A

M602 with Avia V518 five-blade propeller (L-610 in background)

Cutaway drawing of the M602

Cutaway M602B

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AERO-ENGINES - TURBOSHAFT, FRANCE

Date Posted: 18 April 2002

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SOCIETE TURBOMECA

TURBOMECA ARRIEL

This turboshaft has modular construction. The first complete engine ran on 7 August 1974. The following are current versions.

Arriel 1B

For Eurocopter AS 350B/BA. T-O rating 478 kW (641 shp), max continuous 440 kW (590 shp); sfc at 351 kW (470 shp) rating 106.4 Ág/Ns (0.630 lb/h/shp).

Arriel 1C

For twin-engined AS 365N. T-O rating 492 kW (660 shp).

Arriel 1C1, 1K, 1M, 1S

Respectively power twin-engined AS 365F, Agusta A 109K, AS 365 N1 and Sikorsky S-76A+. T-O rating 522 kW (700 shp). Arriel 1C1 is produced as WZ8 by SAEC in China.

Arriel 1C2

Powers twin-engined AS 365N2. OEI 2.5 min 575 kW (771 shp), unlimited OEI and T-O 550 kW (738 shp), max continuous 471 kW (632 shp); sfc at 350 kW (470 shp) 106.9 Ág/J (0.633 lb/h/shp).

Arriel 1D

Powers AS 350B and AS 350L1. T-O rating 510 kW (684 shp); sfc 106.4 Ág/J (0.630 lb/h/shp).

Arriel 1D1

Powers AS 350B2 and AS 550 Fennec. T-O rating 546 kW (732 shp), max continuous 466 kW (625 shp); sfc at 350 kW (470 shp) 106.9 Ág/J (0.633 lb/h/shp); at max continuous 98.12 Ág/J (0.581 lb/h/shp).

Arriel 1E

Powers twin-engined Eurocopter BK 117C1. T-O rating 528 kW (708 shp).

Arriel 1E2

Powers twin-engined BK 117C1. OEI 2.5 min 574 kW (770 shp), unlimited. OEI and T-O 550 kW (738 shp), max continuous 516 kW (692 shp); sfc at 350 kW (470 shp) 106.9 Ág/J (0.633 lb/h/shp); at max continuous 96.94 Ág/J (0.574 lb/h/shp).

Arriel 1K1

Powers twin-engined A 109K2. OEI 2.5 min 575 kW (771 shp), unlimited OEI and T-O 550 kW (738 shp), max continuous 471 kW (632 shp); sfc at 350 kW (470 shp) 106.9 Ág/J (0.633 lb/h/shp).

Arriel 1M1

Powers twin-engined AS 565MA and 565UA. T-O rating 558 kW (723 shp).

Arriel 1S1

Powers twin-engined S-76A+ and S-76C. OEI 2.5 min 598 kW (802 shp), unlimited OEI 588 kW (789 shp), T-O and max continuous 541 kW (725 shp); sfc at 350 kW (470 shp) 106.4 Ág/J (0.630 lb/h/shp), at max continuous 95.25 Ág/J (0.564 lb/h/shp).

Arriel 2

Growth version with increased mass flow and single-stage compressor (gas-generator) turbine. For variants, see below.

Arriel 2B

Powers AS 350B3. T-O rating 632 kW (848 shp), max continuous 544 kW (730 shp); sfc at 400 kW

(536 shp) 104.4 Ág/J (0.618 lb/h/shp), at max continuous 93.76 Ág/J (0.555 lb/h/shp).

Arriel 2C

Powers twin-engined AS 365N3. OEI 30 s 718 kW (963 shp), OEI 2 min 646 kW (866 shp), continuous OEI and T-O 626 kW (839 shp), max continuous 581 kW (779 shp); sfc at 400 kW (536 shp) 104.4 Ág/J (0.618 lb/h/shp), at max continuous 92.74 Ág/J (0.549 lb/h/shp).

Arriel 2C1

Powers twin-engined EC 155B (former AS 365N4). Performance as Arriel 2C but equipped with dual-channel FADEC with manual back-up.

Arriel 2S1

Powers twin-engined S-76A++ and S-76C+. FADEC control. OEI 30 s 735 kW (986 shp), OEI 2 min 663 kW (889 shp), continuous OEI and T-O 638 kW (856 shp), max continuous 587 kW (787 shp); sfc at 400 kW (536 shp) 104.4 Ág/J (0.618 lb/h/shp).

The Arriel is Turbomeca's current best-seller. By February 2000 deliveries totalled 4,500. These had then flown 11,080,000 hours with over 1,300 operators in 110 countries. These totals do not include engines produced in the People's Republic of China (see under SMPMC). Most `1' versions are certificated in the Russian Federation and Associated States (CIS)(RFAS).

The following relates to the Arriel 1B:

Type

Single-shaft free turbine turboshaft.

Compressor

Single-stage axial and supersonic centrifugal. Pressure ratio 9. Mass flow not disclosed.

Combustion Chamber

Annular, with flow radially outwards and then inwards. Centrifugal fuel injection.

Compressor Turbine

Arriel 1, two stages with solid inserted blades; Arriel 2, single stage with single-crystal blades.

Power Turbine

Single axial stage with inserted blades.

Output

Light alloy gearbox, containing two stages of helical gears, giving drive at 6,000 rpm to front and rear. Hydraulic torquemeter.

Accessories

Main pad provides for optional 12,000 rpm alternator; other drives for oil pumps, tachometer generator, governor and starter.

Starting

Electric starter or starter/generator.

Oil System

Independent circuit through gear pump and metallic cartridge filter.

Oil Specification

AIR 3512 or 3513A.

Dimensions

Length, excl accessories	1,090 mm (42.91 in)
Height overall	569 mm (22.40 in)
Width	430 mm (16.93 in)

Weight, Dry

With all engine accessories	120 kg (265 lb)

Performance Ratings

See variants list.

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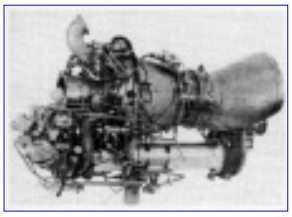
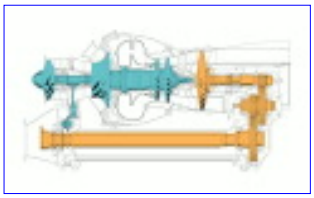
Arriel 1C1

Arriel 1D
(2000)

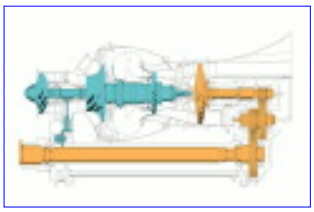


Arriel 1S



Longitudinal section through Arriel 1
(2000)

Arriel 2 (typical)
(2000)



Longitudinal section through Arriel 2
(2000)

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3 Images

AERO-ENGINES - TURBOSHAFT, FRANCE

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOCIETE TURBOMECA

TURBOMECA ARRIUS

Previously known as the TM 319, this turboshaft is compact. It has been selected for important new helicopters, and has also been developed in a turboprop version.

Arrius 1A

Powers AS 355N Twin Squirrel.

Arrius 1M

Powers AS 555 Twin Fennec.

Arrius 2B1

Option for EC135 and EC635; JAA certificated May 1996, FAA 31 July 1996, followed by many other countries. The first time Turbomeca has had to compete for a helicopter also offered with a rival engine; excellently received, ``70 per cent of customers who expressed a choice have chosen the Arrius''. Now warranted 3 years or 2,500 hours, with usual Power by the Hour option.

Arrius 2F

Previously designated 1F, powers EC120 Colibri. Outstanding prototype flight programme, JAA certification December 1996.

Arrius 2G

Being developed for application to Russian helicopters. Initial agreement signed in January 2002 provides for Arrius 2G engines to be made under licence by an affiliate of NPO Saturn (which see, under Russia) to supplement or replace the Rolls-Royce 250-C20R/2 as the powerplant of the twin-engined Ka-226A. The same engine is also intended to power the single-engined Ka-115 Moskvich, at present fitted with the Pratt & Whitney Canada PW206D.

Arrius 2K1

FADEC, option for A109 Power.

Arrius 2K2

Selected to power 30 (+10 option) A109 Power for South Africa.

Deliveries, mainly for the AS355/555, began in 1987 and passed the 850 mark at the end of February 2000, by which time flight time exceeded 510,000 hours with 150 customers in 40 countries. TBO is 3,000 hours, except Arrius 1A which is on 2,000. The same gas generator is used in the Arrius 1D turboprop.

Compressor

Single-stage axial with titanium rotor. Mass flow not disclosed. Pressure ratio 9.0.

Combustion Chamber

Annular reverse-flow, with fuel injection from 10 evenly spaced burners.

Compressor Turbine

Single axial stage with single-crystal blades cast as one-piece blisk.

Power Turbine

Single axial stage.

Control System

Automatic constant-speed control by Elecma. FADEC, except for Arrius 2F which is hydromechanical.

Specific Fuel Consumption

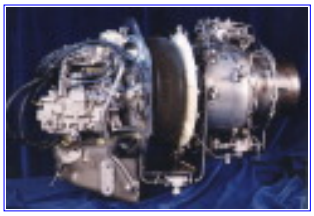
1A 108.3 Ág/J (0.641 lb/h/shp) at 300 shp, 97.32 Ág/J (0.576 lb/h/shp); 2B, 2C, 102.8 Ág/J (0.608 lb/h/shp) at 402 shp; 2B1, 2K1, 104.0 Ág/J (0.615 lb/h/shp) at 402 shp; 92.50 Ág/J (0.547 lb/h/shp) at 570 shp; 2F, 114.0 Ág/J (0.674 lb/h/shp) at 268 shp, 95.90 Ág/J (0.567 lb/h/shp) at 450 shp.

	Dimensions	Weight, dry	Performance ratings (S/L, ISA)

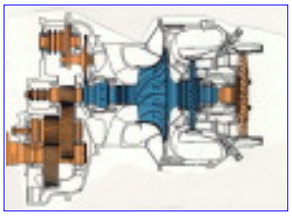
	Length	Width	Height	(Equipped)	OEI 2.5 min	OEI unlimited	T-O	Max continuous
Arrius 1A/1M	793 mm	367 mm	568 mm	101 kg	408 kW	380 kW	357 kW	303 kW
	(31.2 in)	(14.45 in)	(22.36 in)	(223 lb)	(547 shp)	(509.6 shp)	(479 shp)	(406 shp)
Arrius 2B1	947 mm	404 mm	692 mm	111 kg	560 kW	500 kW	500 kW	425 kW
	(37.3 in)	(15.91 in)	(27.24 in)	(245 lb)	(750 shp)	(670 shp)	(670 shp)	(570 shp)
Arrius 2F	945 mm	459 mm	696 mm	103 kg	-	-	376 kW	335 kW
	(37.2 in)	(18.07 in)	(27.40 in)	(227 lb)			(504 shp)	(449 shp)
Arrius 2K1	968 mm	470 mm	670 mm	115 kg	560 kW	500 kW	500 kW	425 kW
	(38.1 in)	(18.50 in)	(26.38 in)	(253.5 lb)	(750 shp)	(670 shp)	(670 shp)	(570 shp)

UPDATED

Arrius 1A



Arrius 2B1



Longitudinal section through Arrius (except 2B1)

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AERO-ENGINES - TURBOSHAFT, FRANCE

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOCIETE TURBOMECA

ROLLS-ROYCE TURBOMECA RTM 322

See the International section.

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AERO-ENGINES - TURBOFAN, FRANCE

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOCIETE TURBOMECA

ROLLS-ROYCE TURBOMECA ADOUR

See the International section.

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SOCIETE TURBOMECA

MTU-TURBOMECA-RR MTR 390

See the International section.

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AERO-ENGINES - MANUFACTURER, FRANCE

Date Posted: 22 March 2002

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Turbomeca was founded by J R Szydlowski in 1938. In 1987, it became part of the Labinal group. M Szydlowski died in 1988, and his daughter succeeded him until her own death in 1996. The Szydlowski family still have a substantial interest, but, in 1999, Turbomeca was purchased by the Groupe Labinal, of which it became an operating division. In its moves to rationalise European industry into stronger groups, the larger French *motoriste* SNECMA (which see) on 15 June 2000 purchased 100 per cent of Sopartech, the company which controls Labinal with 49.4 per cent of the shares and 64.9 per cent of the voting rights. SNECMA (Sopartech) was then offering to buy out the Labinal minority shareholders. SNECMA then planned to dispose of three Labinal divisions, but is retaining the Turbomeca-Microturbo division. (Microturbo, originally an independent company at Toulouse, produces small jet engines for missiles, targets and other UAVs.)

From the outset Turbomeca has specialised in small gas turbines for aircraft. By 1999, nearly 50,000 small turbine engines had been produced, including well over 30,000 for aircraft propulsion, used by over 1,200 customers in 120 countries. Of these, over 17,000 are currently in use, flying more than 3.5 million hours per year. In addition, over 14,000 aero-engines have been produced under licence by what are today Rolls-Royce plc in the UK, Teledyne CAE in the USA, ENMASA in Spain, Hindustan Aeronautics in India, Bet-Shemesh in Israel, SMPMC in China and factories in Romania and former Yugoslavia.

In January 2002 Turbomeca signed a Memorandum of Understanding with two Russian companies. This arrangement is described under the Arrius engine.

A European Small Engines Co-operation Agreement signed in April 1985 joined Turbomeca, MTU of Germany and Rolls-Royce of the UK in promoting what were then three complementary new engines: the Turbomeca TM 333, MTU-Turbomeca-RR MTR 390 and Rolls-Royce Turbomeca RTM 322. Other European small engine makers may join the collaboration, in which each partner may share in engines sold to its own government. Since then the TM333 has been a wholly Turbomeca engine and the RTM 322 has been joined by Rolls-Royce Deutschland, with a share by Netherlands companies.

In March 1989, a UK marketing company was formed, Turbomeca Ltd (see above).

Total covered floor area for Turbomeca's three plants at Bordes, Mézières and Tarnos is 140,500 m^2 (1,512,200 sq ft). The company devotes some 13 per cent of turnover to research and development at its main facility at Bordes. Flight testing is conducted by a subsidiary, CGTM, at Pau-Pyrénées airport. Though a 50 per cent partner in the Adour (see Rolls-Royce Turbomeca in the International section) and Larzac (see Turbomeca-SNECMA) turbofans, and in small turboprops, the company's main business is turboshaft engines for helicopters.

Turbomeca has built up a substantial business in aero-derived engines for surface applications, and for this purpose has formed Eurodyn in partnership with Volvo and Ulstein. Sales in the USA are managed and supported by Turbomeca Engine Corporation, of Grand Prairie, Texas, and Turbomeca has additional subsidiaries in Australia, Brazil, Germany, Japan, Singapore, UK and Uruguay,

From a peak of 4,700 the workforce (excluding Microturbo) was slimmed to 3,410. Sales in 1998 were FFr2,911 million (US$568 million), 735 engines being sold and 1,446 repaired. In mid-2000, employment of the Turbomeca-Microturbo Division was 4,150. Sales amounted to FFr 3,350 million (US$654 million). The division is calculated to have 24 per cent of the world market for helicopter engines.

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AERO-ENGINES - MANUFACTURER, FRANCE

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SNECMA - SOCIETE NATIONALE D'ETUDE ET DE CONSTRUCTION DE MOTEURS D'AVIATION

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Snecma was formed by government decree in 1945 by nationalising all French aircraft-engine companies. From the outset its dwindling piston engine business was partnered by growing sales of the German-designed Atar military turbojet, which proved so amenable to development that it remained in

production for 40 years, over 5,200 being delivered. It provided a foundation for the later M53 and M88. Other military work included the Larzac turbofan and Tyne turboprop, both in partnership with others, and today Snecma has become a major partner in FLA propulsion.

Partnership with GE to produce a turbofan in the 10-ton thrust class led to the CFM56. Sales could hardly have got off to a slower start, but today this family of engines has overtaken the JT8D as the best-selling civil engine of all time. Snecma shares in the GE90 and CF6 (see the collaborative programmes graphical insert).

This has established Snecma as a player on the airline market, enabling an agreement to be signed in April 1996 with Pratt & Whitney Canada for the development of a completely new turbofan for regional aircraft. See JV in International section, where it is explained that pressure from partner GE resulted in Snecma (at least for the present) abandoning this joint venture.

From 1962 Snecma collaborated with Bristol Siddeley (later Rolls-Royce) on the engine for Concorde (which see, in International section) and collaboration with the British company has continued. Since 1990, the two companies have been working together on AMET (Advanced Military Engine Technology), with the objective of doubling the thrust/weight ratio of combat-aircraft engines by 2010 and reducing operating costs by 30 per cent. Thus, it could be a collaborative engine that will follow the M88 and EJ200.

Other projects include the M138 turboprop in the 6,000-7,500 kW (8,000-10,000 shp) class which has been merged with the rival BR700TP to produce the TP400 engine in the 9,694 kW (13,000 shp) class (see under AMC in the International section), and various projects for propulsion of a second-generation SST (see feature article `Civil engines').

For many years - in particular in 1992 - Snecma made serious trading losses. The previous chairman was dismissed by the French government in June 1996 for failing to turn the company around, in particular by selling subsidiaries (listed below). A major problem was that, with the main development of the CFM56 and M88 completed, there was little work for some 800 highly-qualified engineers, and especially those who specialise in hot-section development. The then-chairman urged the board of CFM International (qv) to assign the hot section of the proposed CFMXX to the French partner.

This was not accepted by GE, which itself entered into a unilateral MoU with Airbus for an engine pitched just above the upper limit of the GE-Snecma thrust bracket for CFM engines. This is reported to have infuriated the Snecma management, triggering the agreement with Pratt & Whitney Canada, as explained above. This clearly would have competed with GE's growth versions of CF34, resulting in the US partner urging a new chairman to be appointed in Paris.

In 2002 all Snecma's subsidiaries were still in place. These include:

Hispano-Suiza (reversers, nacelles, gearboxes and power transmissions on such engines as the Trent, CFM56, CTS800 and CFE738); Sochata (engine repair and maintenance for many air forces and airlines); SEP (engines for large space launchers and a wide range of missiles); Messier-Dowty (landing gears); Messier-Bugatti (braking systems and hydraulics); and Techspace Aero (see under Belgium). FAMAT, jointly owned with GE, produces large castings for the CFM56, CF6-80 and GE90 at St Nazaire. CFAN, also jointly owned with GE, produces the fan blades for the GE90 at San Marcos, Texas.

In 1999 the Snecma group employed about 21,000, just under half of whom were on Snecma's own payroll. Sales totalled FFr31.9 billion (US$4.72 billion), up 23 per cent on 1998. Exports accounted for 71 per cent of the total. Far and away the biggest factor was Snecma's 50 per cent share in CFMI. Military busiiness, which once accounted for 92 per cent of Snecma sales, fell to 15 per cent. Group profit was up to a new high of FFr1.7 billion.

In 2000 Snecma expanded by acquiring the powerful Labinal group, one of the subsidiaries of which is Turbomeca (see later). Another acquisition was Hurel-Dubois, whose reversers are featured in `Civil engines'. In December 2000 Finance Minister Laurent Fabius praised Snecma for these and other acquisitions, and said he would like to see the group - by this time the Western world's fourth-largest aero-engine manufacturer - ``play a key role in coming changes in the Continent's aircraft-engine branch".

This clearly indicated that he expected the French *motoriste* to play the central role in organising the creation of a mighty European aero-engine group. Such a move would be difficult were Snecma to remain 97 per cent state-owned. In January 2001 Chairman Bechat said ``The other European engine manufacturers don't want to find themselves falling partly under French government ownership. I'm not ashamed of our status as a public company, but that status isn't compatible with forming a big European alliance".

Despite this, by September 2001 everything was in place for Snecma to offer 25 per cent of its equity to the public. This was expected to raise Euro1.5 billion, but in that same month the company announced that it was postponing the offering ``because of poor market conditions" (this was three days before the terrorist attack on New York). Three months later the European Commission authorized the French Government to lend Snecma Euro120 million to help fund its participation in two large US engine programmes, the GP7000 and GE90-115. It said the advance would be repaid ``if the programmes are successful, in the form of a tax on the delivery of engines and a tax on maintenance activities".

In spring 2001 sales for 2000 were predicted to total FFr38.5 billion (US$5.7 billion). In fact the actual figures were published as revenues of Euro5.6 billion and net earnings of Euro318 million. Of these totals, an even greater proportion than ever before was accounted for by the CFM56.

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AERO-ENGINES - TURBOSHAFT, FRANCE

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOCIETE TURBOMECA

TURBOMECA ASTAZOU TURBOSHAFT

This turboshaft series is derived from the Astazou II turboprop. Variants are:

Astazou IIA

Rated at 390 kW (523 shp) for SA 318C Alouette II. Total 732, completed 1977.

Astazou IIIA

Derived from IIA but with revised turbine to match power needs of SA 341G Gazelle. Rated at 441 kW (591 shp). Produced as IIIA2, B2, C2 and N2 jointly by Turbomeca and Rolls-Royce, with 1,008 delivered.

Astazou IIIB

For SA 316B Alouette III. Derated to 425 kW (570 shp).

Astazou XIVB and XIVF

For SA 319B and 319C; XIVB is civil and XIVF military. Flat rated to 441 kW (591 shp) (1 hour) up to 4,000 m (13,125 ft) or +55~C.

Astazou XIVH

For SA 342J/L, rated at 649 kW (870 shp) to remove altitude and temperature limitations; 1,146 delivered.

Astazou XVIIIA

Higher gas temperature. Powers AS 360C.

Astazou XX

Fourth axial compressor stage added. Designed for operation in hot and high countries. Powered prototype SA 361.

By 1999 a total of 2,754 Astazous had been delivered by Turbomeca (excluding Rolls-Royce production); the engine is no longer in production.

The following description relates to the Astazou III, except where indicated:

Type

Single-shaft axial-plus-centrifugal turboshaft.

Compressor

Single-stage axial (IIA, III), two-stage axial (XIV, XVIII) or three-stage axial (XX) followed by single-stage centrifugal. Mass flow (II, III) 2.5 kg (5.5 lb)/s, (XIV, XVIII) 3.3 kg (7.3 lb)/s, (XX) 4.2 kg (9.25 lb)/s. Pressure ratio (II, III) 6; (XIV, XVIII) 7.5, (XX) 9.4.

Combustion Chamber

Reverse flow annular with centrifugal injector using rotary atomiser. Two ventilated torch igniters.

Turbine

Three-stage axial with blades integral with discs.

Output

Two-stage epicyclic having helical primary and straight secondary gears. Ratio 7.039:1 (XIVB/F, 7.345; XVIIIA, 7.375).

Accessories

Five drive pads on casing forming rear of air intake.

Starting

Electrical, automatic.

Control System

Automatic constant-speed control.

Oil System

Pressure type with gear-type pumps. Oil tank of 8 litres (17 US pints; 14 Imp pints) capacity.

Dimensions

Length overall: Astazou IIA	1,272 mm (50.0 in)
Astazou III, XIVB/F	1,433 mm (56.3 in)
Astazou XIVH	1,470 mm (57.9 in)
Astazou XVIIIA	1,327 mm (52.2 in)
Astazou XX	1,529 mm (60.22 in)
Height: Astazou IIA	458 mm (18.0 in)
Astazou III, XIVH	460 mm (18.1 in)
Astazou XVIIIA	698 mm (27.48 in)
Astazou XX	721 mm (28.4 in)
Width: Astazou IIA	480 mm (18.8 in)
Astazou III, XIVH	460 mm (18.1 in)

Weight, Dry

Equipped: Astazou III	147 kg (324 lb)
Astazou III_2	150 kg (330 lb)
Astazou XIVB/F	166 kg (366 lb)
Astazou XIVH	160 kg (353 lb)
Astazou XVIIIA	155 kg (341 lb)
Astazou XX	195 kg (430 lb)

Performance Ratings

Max power: Astazou IIA	390 kW (523 shp)
Astazou III	441 kW (591 shp)
Astazou III_2	481 kW (645 shp)
Astazou XIVH	649 kW (870 shp)
Astazou XX	749 kW (1,005 shp)
One hour: Astazou XIVB/F	441 kW (591 shp)
Astazou XVIIIA	651 kW (873 shp) maintained at sea level to 40~C
Max continuous: Astazou IIA	353 kW (473 shp)
Astazou III	390 kW (523 shp)
Astazou III_2	441 kW (592 shp)
Astazou XIVB/F	405 kW (543 shp)

Astazou XIVH	flat rated in SA 341 at 440.7 kW (591 shp) to 55~C or 4,000 m (13,125 ft)
Astazou XVIIIA	600 kW (805 shp)
Astazou XX	675 kW (905 shp)

Specific Fuel Consumption

At max power rating:	
Astazou IIA	105.3 Ág/J (0.623 lb/h/shp)
Astazou III	108.7 Ág/J (0.643 lb/h/shp)
Astazou III_2	109.9 Ág/J (0.650 lb/h/shp)
Astazou XIVB/F	105.5 Ág/J (0.624 lb/h/shp)
Astazou XVIIIA	91.3 Ág/J (0.540 lb/h/shp)
Astazou XX	85.9 Ág/J (0.508 lb/h/shp)

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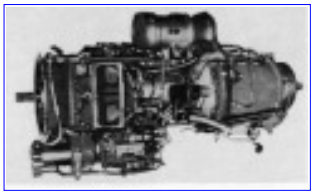
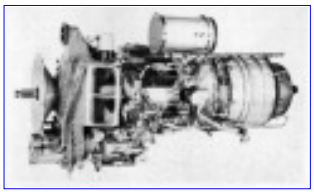
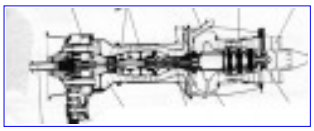


Astazou XVIIIA



Astazou XX



Longitudinal section through typical Astazou

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AERO-ENGINES - TURBOPROP, FRANCE

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOCIETE TURBOMECA

TURBOMECA ASTAZOU TURBOPROP

Though no longer in production, the Astazou is the major turboprop in the Turbomeca range. The Astazou XIV was certificated by the French airworthiness authorities in October 1968, followed by ARB/FAA certification of the Astazou XIVC and C1 in March 1969.

Current versions of the Astazou are:

Astazou XII

Powered Shorts Skyvan Srs 2 at 515 kW (690 shp) and Pilatus Turbo-Porter PC-6/A1-H2 at 522 kW (700 shp).

Astazou XIV (AZ14)

Developed from Astazou XII. Powered early Jetstream aircraft.

Astazou XVI (AZ16)

Higher-rated version of Astazou XIV and first engine in production with Turbomeca air-cooled turbine. The XVID, without starter/generator, powers the former production versions of the Jetstream, including aircraft originally delivered as the Jetstream T.1 of the RAF and T.2 and T.3 of the Royal Navy. The

XVIZ powers the Nord 260A. The Astazou XVIG, equipped for sustained inverted flight, powers the Argentine IA 58 Pucará. Deliveries of all XVI versions totalled 367, completed in 1984.

The basic core is similar to that of the Astazou turboshaft (which see). The only significant difference is the front drive to a reduction gear housed in the centre of a cast aluminium inlet which carries the accessories on its rear face. Like all Astazou engines the main shaft turns at a constant 43,000 rpm, power being controlled by a SFERMA system which can operate in either of two methods. Power and jetpipe temperature can be controlled at preselected levels by varying fuel flow and pitch of the Ratier-Figeac propeller. Alternatively, propeller pitch is maintained at a pilot-selected value, and the engine is controlled by varying fuel flow, a mechanical governor holding rpm steady at any airspeed.

Dimensions

Diameter over intake cowl	546 mm (21.5 in)
Overall length, incl propeller	2,047 mm (80.6 in)

Weights

With accessories:	
Astazou XIV	approx 206 kg (454 lb)
Astazou XVID	205 kg (452 lb)
Astazou XVIG	228 kg (502 lb)
Astazou XVIZ	213 kg (468 lb)

Performance Ratings

T-O:	
Astazou XIV	636 ekW; 596.5 kW (853 ehp; 800 shp) at 43,000 rpm
Astazou XVID	723 ekW; 681 kW (969 ehp; 913 shp) at 43,089 rpm
Astazou XVIG, XVIZ	761 ekW; 720 kW (1,020 ehp; 965 shp) at 43,000 rpm
Max continuous:	
Astazou XIV	574 ekW; 537 kW (770 ehp; 720 shp) at 43,000 rpm
Astazou XVID	626 ekW; 586 kW (840 ehp; 786 shp) at 43,089 rpm
Astazou XVIG, XVIZ	696 ekW; 654 kW (934 ehp; 877 shp) at 43,000 rpm

Specific Fuel Consumption

(at T-O rating)

Astazou XIV	92.4 µg/J (0.547 lb/h/shp)
Astazou XVI (all versions)	88.7 µg/J (0.525 lb/h/shp)

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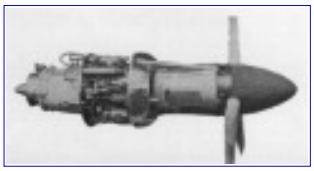

Astazou XIVC

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AERO-ENGINES - TURBOSHAFT, FRANCE

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOCIETE TURBOMECA

TURBOMECA ARTOUSTE

The Artouste was the world's most important pioneer small turboshaft engine. It made possible the Sud-Aviation (later Aérospatiale) Alouette, the first production turbine-engined helicopter.

Artouste IIC

Powers the SE 313B Alouette II, first flown on 12 March 1955 and certificated on 2 May 1957. Total produced 1,445, ending 1964, not including engines made under licence by HAL (India) and Blackburn/Bristol Siddeley, now Rolls-Royce (UK).

Artouste III

Uprated version to power all variants of the SA 315B Lama and 316B Alouette III. The IIIB, with an output-shaft speed of 5,864 rpm, was certificated in May 1961, and the further-uprated IIID, with an output-shaft speed of 5,773 rpm, in April 1971. About 2,550 of both versions were produced, ending in 1988, not including engines made under licence by HAL (India).

Type

Single-shaft turboshaft engine.

Intake

Cast aluminium with rectangular air intake on each side and accessory wheelcase and output shaft on the front.

Compressor

(II) Single-stage centrifugal, followed by radial and then axial diffusers. Mass flow 3.2 kg (7.05 lb)/s. Pressure ratio 3.88. (III) Single-stage axial followed by single-stage centrifugal. Mass flow 4.3 kg (9.5 lb)/s. Pressure ratio 5.2.

Combustion Chamber

Annular reverse-flow, with centrifugal fuel injection from rotary disc atomiser mounted on main shaft. Two Turbomeca igniters.

Turbine

(II) Two-stage axial with blades integral with discs. (III) Three-stage axial.

Accessories

Drives for oil pump, fuel control unit, Labavia electric starter and tachogenerator.

Jetpipe

Single pipe pointing to rear, in helicopter curved slightly upwards. Maximum gas temperature 500~C.

Fuel Specification

AIR 3405; (II only) gasoline (petrol) used for starting.

Oil Specification

AIR 5212 or 3155A mineral type.

Dimensions

Length:	
II	1,440 mm (56.7 in)
III	1,815 mm (71.46 in)
Width:	
II	390 mm (15.35 in)
III	507 mm (19.96 in)
Height:	
II	545 mm (21.5 in)
III	627 mm (24.68 in)

Weight, Dry

II	115 kg (254 lb)
IIIB	182 kg (400 lb)

IIID	178 kg (392 lb)

Performance Ratings

(S/L, ISA)

T-O:	
II	395 kW (530 shp) at 34,000 rpm
IIIB	420 kW (563 shp) at 33,300 rpm, flat-rated to 55~C or 4,000 m (13,125 ft)
Max continuous:	
II	358 kW (480 shp) at 34,000 rpm
IIIB, IIID	405 kW (543 shp) at 33,300 rpm

Specific Fuel Consumption (T-O):

II	139.0 Ág/J (0.823 lb/h/shp)
IIIB	128.7 Ág/J (0.762 lb/h/shp)
IIID	126.2 Ág/J (0.747 lb/h/shp)

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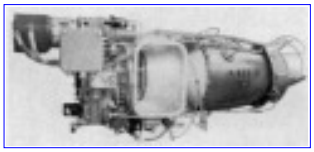

Artouste IIIB

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AERO-ENGINES - TURBOSHAFT, FRANCE

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOCIETE TURBOMECA

TURBOMECA ARRIUS 1D

The turboprop version is fully aerobatic. The gas generator and power turbine modules are identical to those of the Arrius 1.

Arrius 1D

This engine first ran on 11 September 1985. Flight testing in an Epsilon began in November 1985, followed by a Valmet L-90 TP in December 1987. The Arrius 1D is flying in the Socata Oméga. Performance figures below.

Arrius 2D

Uprated version under study to re-engine Epsilon trainers of the Armée de l'Air.

Dimensions

Length	826 mm (32.52 in)
Width	476 mm (18.74 in)
Height	590 mm (23.22 in)

Weight, Dry

Bare	111 kg (245 lb)

Performance Ratings

(ISA, S/L)

T-O	313 kW (420 shp)
Cruise (6,100 m; 20,000 ft)	179 kW (240 shp)

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Arrius 1D

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AERO-ENGINES - TURBOJET, FRANCE

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOCIETE NATIONALE D'ETUDE ET DE CONSTRUCTION DE MOTEURS D'AVIATION

SNECMA ATAR

Remarkably, this engine, still in worldwide use, was originally developed (mainly by German engineers, led by Dr H Oestrich), in 1945-47, and the first engine was on test in April 1948. All examples running today are of course of much later versions.

Atar 8

This was designed in 1954-56 and introduced a two-stage turbine driving an improved compressor with a zero stage handling greater airflow. The Etendard IVM and IVP are powered by the Atar 8C. The Super Etendard is powered by the Atar 8K-50, which is essentially a 9K50 without the afterburner.

Atar 9C

Compared with the earlier 9B this introduced a new compressor, a self-contained starter and an improved overspeed, which comes into operation automatically when the aircraft reaches M1.4, giving power equivalent to a sea level thrust of 62.7 kN (14,110 lb). Equips most Mirage III and 5.

Atar 9K10

Improved combustion chamber, turbine cooling and afterburner. Powers the Mirage IVP.

Atar 9K50

Derived from the Atar 9K10. Designed to offer improved subsonic specific fuel consumption, increased thrust for supersonic acceleration and improved overhaul life. The main improvements are in an entirely redesigned turbine with blades not forged but cast and coated with refractory metal from the vapour phase. Stages 1 and 8 of the compressor have been redesigned, resulting in pressure ratio raised, coupled with slightly augmented mass flow. The control and electronic equipment have been revised and extended to improve the security of single-engined aircraft. The 9K50 is the power plant of all production Mirage F1 versions and the Mirage 50. Total deliveries were 1,092, for 28 air forces. Total flight time 1.8 million hours.

Atar 8K50

This is essentially the 9K50, the latest variant in production, re-engineered to have a simple unaugmented jetpipe and fixed nozzle, for the Super Etendard. Parts are protected against sea corrosion. The 8K50 completed certification in May 1975. Production deliveries began in May 1977.

Atar Plus

In June 1995 Snecma, Denel of South Africa and ITP of Spain signed an agreement covering modifications to the core of the Atar 8K50 and 9K50. Objectives included reduction in maintenance cost of 10 to 15 per cent, extension of HSI from 300 to 400 hours, TBO to 1,200 hours and a reduction in time to accelerate from idle to maximum afterburner `by 3 to 5 seconds'. Modifications included a new compressor outlet guide vane assembly, turbine nozzle and control system.

Production of Atar engines was completed in 1995 at a total of 5,250. Of these engines, about 1,600 are still in service. Different versions were in production for 45 years.

Type

Single-shaft turbojet.

Intake

Annular type surrounding starter bullet. Six radial struts house the accessories bevel gear driveshafts, starter feed ducts and lubricating oil outlet ducts. De-icing by circulation of hot air in casing, struts and hollow nozzle guide vanes.

Compressor

Nine-stage axial flow. Rotor drum made up of individual discs bolted to shaft. Shaft carried in ball bearing at front, roller bearings at centre and rear. Compressor casing is of light alloy in two halves. Forged and machined solid rotor and stator blades. Stator blades: rows 1, 2, 7-9 of steel, rows 3-6 of light alloy. Rotor blades: rows 1, 7-9 of steel, rows 2-6 of light alloy. All blades have prismatic roots which slide into corresponding slots in periphery of discs. Mass flow (8B, 9C) 68 kg (150 lb)/s, (9K, 8K) 72 kg (158 lb)/s. Pressure ratio (8B, 9C) 5.2, (9K, 8K) 6.5.

Combustion Chamber

Annular type with 20 direct-flow burners. Steel construction. Two starting pre-chambers and two igniters to ensure easy starting and relighting.

Turbine

Two-stage axial flow type. Steel wheels splined to shaft, which is carried in roller bearings. Steel blades with fir-tree roots. First stage cooled by air carried through annular duct around shaft. Second stage cooled by air ducted through interior of shaft. Hollow air-cooled steel NGVs.

Jetpipe

(8B, 8K) Outer sheet steel casing with central cone. (9B) Variable-area nozzle with two clam-shell shutters, operated hydraulically. (9K) Variable-area nozzle with multiple petals.

Accessories

Two angled drive shafts, above and to port side of engine. Total available power 90 hp.

Starting

(9B) Compressed air starter inside intake central bullet, (other versions) autonomous pneumatic starter.

Control System

Twin-delivery type. Two-stage fuel pump: first stage of low-pressure centrifugal type, second stage of high-pressure gear type. Atar single-lever control regulating automatically rpm and temperature. Max fuel pressure 80 kg/cm^2 (1,140 lb/sq in).

Oil System

Non-return system. Normal oil supply pressure 3.5 kg/cm^2 (50 lb/sq in).

Mounting

Suspension by means of one attachment at the front of the engine, above the casing, and two points on each side of the central casing.

Dimensions

Diameter	1,020 mm (40.2 in)
Length overall:	
Atar 8B	3,914 mm (154.1 in)
Atar 8K50	3,936 mm (155 in)
Atar 9C, 9K50	5,944 mm (234 in)

Weight, Dry

Complete with all accessories:	
Atar 8B	1,040 kg (2,293 lb)
Atar 8K50	1,155 kg (2,546 lb)
Atar 9C	1,456 kg (3,209 lb)
Atar 9K50	1,582 kg (3,487 lb)

Performance Ratings

Max with afterburner:	
Atar 9C	58.9 kN (13,320 lb st) at 8,400 rpm
Atar 9K50	70.6 kN (15,870 lb st) at 8,400 rpm
Max without afterburner:	
Atar 8B	43.1 kN (9,700 lb st) at 8,400 rpm
Atar 8K50	49 kN (11,025 lb st) at 8,550 rpm
Atar 9C	42 kN (9,430 lb st) at 8,400 rpm
Atar 9K50	49.2 kN (11,055 lb st)

Specific Fuel Consumption

At max rating with afterburner:	
Atar 9C	57.5 mg/Ns (2.03 lb/h/lb st)
Atar 9K50	55.5 mg/Ns (1.96 lb/h/lb st)
At max rating without afterburner:	
Atar 8B	27.8 mg/Ns (0.98 lb/h/lb st)
Atar 8K50	27.5 mg/Ns (0.97 lb/h/lb st)
Atar 9C	28.6 mg/Ns (1.01 lb/h/lb st)
Atar 9K50	27.5 mg/Ns (0.97 lb/h/lb st)

Oil Consumption

	max 1.5 litres (2.64 Imp pints)/h

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Atar 9K50

Cutaway drawing of Atar 9K50

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AERO-ENGINES - TURBOFAN, FRANCE

Date Posted: 01 May 2001

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SOCIETE NATIONALE D'ETUDE ET DE CONSTRUCTION DE MOTEURS D'AVIATION

SNECMA M88

Snecma began to develop a totally new and smaller fighter engine in the late 1970s, supporting it with a programme called Dextre which concentrated on highly loaded turbines with aircooled blades. By 1980 a demonstrator was running, with the principal objectives an engine with 50 per cent better thrust/weight ratio than the Atar 9K50 and thrust of 75 to 85 kN (16,872 to 19,100 lb st). Today the M88 is planned to cover thrusts from 50 to 105 kN (11,250 to 23,600 lb st) using an essentially common core, and the same core has been studied as the basis for the CFMXX commercial turbofan.

In 1983 Dassault committed to build the Rafale fighter in both air force and naval versions, the prototypes being initially powered (for reasons of timing) by the GE F404. Snecma obtained full development funding for the M88 in 1987, since when the M88-2 has been developed on time and 15 per cent under budget.

M88-2

Baseline engine for all current versions of Rafale, as described below. T-O rating 75.0 kN (16,872 lb st) with maximum afterburner, 50.0 kN (11,250 lb st) maximum dry. Core tested at 1700K 1984-86 and at 1850K 1986-89. FETT February 1989, first flight (Rafale A, at first replacing one F404 only) February 1990. Total time on 22 development engines (October 1999) 16,700 hours, including 6,400 hours in Rafale flight test. First production order (including spares, equivalent to 33 engines) December 1992. Qualification 22 March 1996. First delivery December 1996. In October 1999 the order book stood at 160 engines, plus modules and spares.

At the beginning of 2001 it was admitted that the M88-2 was experiencing problems. Total hot-section life was then 300 hours (compared with 800 hours mandated in the specification), with an inspection requiring engine removal every 150 hours. This was considered to be serious for the Aéronavale, whose Rafale M is scheduled to enter service in 2001, embarked aboard *Charles de Gaulle*. According to Snecma ``Within a one-hour flight the Rafale pilot, because his aircraft has vastly superior performance, would go through seven or eight cycles (reheat, deceleration, reheat again. . .) compared with two or three in the F-8 or Super Etendard. Consequently, [maintainability of] the M88 has suffered".

M88-2K

Version for single-engined aircraft, initially proposed for South Korean KTX-2. T-O rating 75 kN (16,872 lb st).

M88-2 Step 4 One solution being considered is to load new FADEC software which prevents maximum acceleration (spool-up) or the use of full reheat. For the longer term, in late 2000 Snecma and the Defence Ministry launched a programme costed at £80 million to develop the Step 4 engine. Chronologically preceding the M88-3 (see next version), this will incorporate a new HP compressor with the first three stages made as blisks, with slightly increased disc diameter and blades aerodynamically reprofiled according to 3-D computer modelling. This will enable cooling airflow to be reduced, at the expense of a 50ºC increase in TET, thus increasing combustion airflow. Other changes include a change to a cheaper alloy in the LP turbine (the N18 at present used is very costly) and a careful revision of titanium alloy parts because of the discovery of ``certain metallurgical defects". The first Step 4 engine went on test in June 2000, began flight test at Istres later that year, and is now to be qualified in October 2001. The first production Step 4 engines will be Nos 31 and 32, ready for installation in the `Rafale Mk 2' prototype in January 2002. It is planned to introduce the improved engine in 2004, and to remanufacture the 30 Step 1 engines to Step 4 standard. However, the Aéronavale has expressed concern that the Step 4 engine does not address the life problems, mentioned earlier, which surfaced after Step 4 was defined.

M88-3

First growth stage, developed for `20 per cent thrust increase', with a new LP compressor with increased diameter and higher performance (developed within CENTOR programme). T-O rating 90 kN (20,250 lb st) class. OPR 26. Evaluated for JAS 39 upgrade.

It was originally intended to begin testing the M88-3 in December 1996. For several years work on this engine has caused it to diverge to the point where it can be considered a second-generation design, commonality with previous M88 engines being no more than 50 per cent. The list of modifications, too numerous to enumerate here, and manufacture of three prototype engines, are estimated to result in a development bill of not less than £250 million. Snecma has indicated its willingness to pay half of this, and in February 2001 it was expected that an agreement to go ahead was imminent. Snecma now hopes to begin testing the first M88-3 in September 2003, to begin flight testing in a Rafale in October 2003, and to deliver the first two production engines to Dassault in January 2006.

Type

Two-shaft turbofan (bypass turbojet) with afterburner.

Inlet

Titanium ring with thermal anti-icing and 15 fixed radial struts, each with a variable-incidence trailing fan.

LP Compressor

Three stages, with solid titanium-alloy blades with part-span shrouds on first stage. Designed with large surge

margin for violent manoeuvres. Fan duct of PMR15 composite. Mass flow 65 kg (143.3 lb)/s. BPR 0.3.

HP Compressor

Six stages, with three variable stators. Discs of N18 powder-metallurgy. OPR 24.0.

Combustion Chamber

Annular, with inverse taper (turbine diameter greater than HP compressor delivery), multiple airblast fuel nozzles, advanced cooling and ceramic coatings.

HP Turbine

Single stage, with air-cooled AM1 single-crystal rotor blades inserted in N18 powder-metallurgy disc. TGT 1,577ºC.

LP Turbine

Single stage, with air-cooled rotor blades.

Aferburner

High-intensity combustion from radial flameholder gutters `designed to provide outstanding time to max A/B thrust and low IR signature`.

Nozzle

Variable profile and area with 10 primary and secondary flaps, outer ring of SEP-developed C/SiC material for reduced weight.

Accessories

Mounted on remote gearbox curved around underside of compressor case, driven by radial bevel-gear shaft from front of HP compressor.

Control System

Fully redundant FADEC which also performs safety and maintenance functions.

Dimensions

Length	3,538 mm (139.3 in)
Inlet diameter	669 mm (26.3 in)
Overall diameter	820 mm (32.3 in)

Weight

Dry	897 kg (1,970 lb)

Performance Ratings

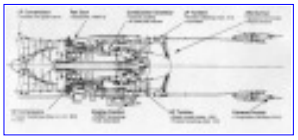
See model listing

Specific Fuel Consumption

Maximum a/b	51.0 mg/Ns (1.80 lb/h/lb st)
MIL (max dry)	22.1 mg/Ns (0.78 lb/h/lb st)

UPDATED

M88 first demonstrator



Longitudinal section through Snecma M88-2

M88-2 development engine

M88-2 production engine

M88-2 cutaway drawing

Installing M88-2 in Rafale CO1

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AERO-ENGINES - TURBOSHAFT, FRANCE

Date Posted: 30 August 2000

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SOCIETE TURBOMECA

TURBOMECA MAKILA

This turboshaft engine was designed to power the AS 332 Super Puma. It was derived partly from the Turmo, but has a higher pressure ratio for better fuel economy, and the design incorporates rapid-strip modular construction.

Makila 1A

Certificated 1980. Powers AS 332 Super Puma.

Makila 1A1

OEI 2.5 min 1,400 kW (1,877 shp), unlimited OEI 1,330 kW (1,784 shp), T-O 1,357 kW (1,820 shp), max continuous 1,185 kW (1,589 shp). Certificated 1984. Powers Super Puma I and Cougar I.

Makila 1A2

OEI 30 s 1,573 kW (2,109 shp), OEI 2 min 1,467 kW (1,967 shp), unlimited OEI 1,420 kW (1,904 shp), T-O 1,376 kW (1,845 shp), max continuous 1,236 kW (1,657 shp). Certificated 1991. Powers Puma II and Cougar II.

Makila 1K2

Performance as 1A2. Powers CSH-2 Rooivalk.

By 2000, about 1,400 Makila engines had been delivered. They had then flown just over 4 million hours. On 29 May 1996, a Puma began flight testing after being re-engined, and the Makila is now being marketed for this retrofit. TBO (all) is 3,000 hours.

Type

Free-turbine turboshaft.

Intake
Cast aluminium intake has an axial inlet and incorporates integral oil tank and an accessory drive train.

Compressor
Three axial stages followed by one centrifugal. Mass flow 5.5 kg (12.1 lb)/s. Pressure ratio 10.4.

Combustion Chamber
Annular, with centrifugal fuel injection from rotary atomiser on main shaft. Two ventilated torch igniters.

Compressor Turbine
Two-stage axial with single-crystal blades. Inlet gas temperature 1,180~C.

Power Turbine
Two-stage axial. Output speed (1A, 1A1) 22,850 rpm, (1A2, 1K2) 22,962 rpm.

Jetpipe
Single, handed to right or left to discharge outboard.

Accessories
Fuel-controller, starter, oil pumps and tachogenerator.

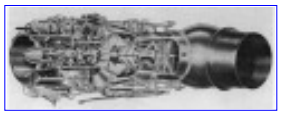
Starting
Electric starter and ignition unit controlled by ECB or DECU.

Control System
Hydromechanical unit governing fuel flow. On 1A and 1A1, an electronic control box. On 1A2 and 1K2, a DECU.

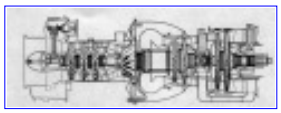
	Dimensions			**Weight, dry**		**Specific fuel consumption**
	Length	**Width**	**Height**	**Basic**	**Equipped**	**Cruise (S/L)**
Makila 1A	2,103 mm	528 mm	680 mm	176 kg	243 kg	97.7 Ág/J
	(82.795 in)	(20.787 in)	(26.771 in)	(388 lb)	(535 lb)	(0.578 lb/h/shp)
Makila 1A1	2,103 mm	528 mm	680 mm	174 kg	241 kg	94.9 Ág/J
	(82.795 in)	(20.787 in)	(26.771 in)	(383 lb)	(531 lb)	(0.562 lb/h/shp)
Makila 1A2	2,117 mm	498 mm	673 mm	180 kg	247 kg	93.1 Ág/J
	(83.346 in)	(19.606 in)	(26.496 in)	(396 lb)	(544 lb)	(0.551 lb/h/shp)
Makila 1K2	2,117 mm	498 mm	673 mm	180 kg	247 kg	94.5 Ág/J
	(83.346 in)	(19.606 in)	(26.496 in)	(396 lb)	(544 lb)	(0.559 lb/h/shp)

UPDATED

Makila 1A2



Cutaway drawing of Makila 1A



Longitudinal section through Makila 1A

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AERO-ENGINES - TURBOFAN, FRANCE

Date Posted: 30 August 2000

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SOCIETE TURBOMECA

TURBOMECA AUBISQUE

This small turbofan found only one application, but a substantial number of this aircraft have flown intensively for 30 years. The aircraft is the Saab-105 light twin-jet, first flown in 1963 and used as the Sk60A trainer and Sk60C reconnaissance aircraft, 150 being built. From 1998 they have been re-engined by the Williams Rolls FJ44-1C.

Type

Light turbofan engine.

Intake

Annular intake and central bullet fairing of light alloy, with two support webs in vertical plane. Starter/generator and accessory gear trains in bullet fairing. Front casing of light alloy, comprising an outer casing and an inner wall which forms the air duct. Lower part of outer casing is extended to provide a mounting for the accessories.

Fan

Single-stage fan is driven through spur reduction gearing with a ratio of 1:2.1318, so that it turns at 15,245 rpm at T-O rating. The front of the casing supports the fan-stage front bearing and carries a row of variable-incidence inlet guide vanes which are provided with thermal de-icing. At the rear of the casing are two rows of straightener vanes and the housing for the rear fan-stage bearing.

Compressor

Single axial stage followed by a single centrifugal stage. Two rows of diffuser vanes between the stages and two more aft of the centrifugal stage, of which the first is radial and the second axial. Pressure ratio 6.9. Air mass flow 22.25 kg (49 lb).

Combustion Chamber
Annular type, with usual Turbomeca rotary atomiser fuel injection system.

Turbine
Two-stage turbine with separate discs and inserted blades. Discs coupled together and to fore and aft shafts by special bolts and curvic couplings. Front shaft is coupled directly to compressor. Rear shaft is carried in rear rotor bearing.

Jetpipe
Inner and outer sheet metal casings, latter supported by three hollow struts, surrounded by annual bypass air duct.

Accessories
Provision for tachometer drive, oil pumps (including one for inverted flight), guide-vane controls, fuel pump and regulator and, eventually, a 20 kVA alternator.

Mounting
Lateral attachment points on each side of entry casing. Main mounting points on lower part of centre casing.

Dimensions

Length	2,048 mm (80.63 in)
Width	650 mm (25.59 in)
Height	750 mm (29.53 in)

Weight, Dry

With full equipment	290 kg (640 lb)

Performance Ratings (S/L, ISA)

T-O	7.30 kN (1,642 lb st)
Max continuous	5.84 kN (1,312 lb st)

Specific Fuel Consumption

T-O, max continuous	16.64 mg/Ns (0.60 lb/h/lb st)

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AERO-ENGINES - TURBOSHAFT, FRANCE

Date Posted: 30 August 2000

Jane's Aero-Engines 08

SOCIETE TURBOMECA

TURBOMECA TURMO

The Turmo is a free-turbine engine for helicopters. It was designed to power the Super Frelon, but most were produced to power the Puma. Production was completed in 1990 at 2,020, not including engines made under licence by Turbomecanica of Romania. The replacement engine for later helicopters is the Makila.

Turmo IIIC$_3$

This was the original engine of the triple-engined SA 321 Super Frelon helicopter. Maximum contingency rating is 1,480 shp.

Turmo IIIC$_4$

Developed from Turmo IIIC$_3$ and with a maximum contingency rating of 1,384 shp, this all-weather version was manufactured jointly by Turbomeca and Rolls-Royce to power SA 330 Puma twin-engined helicopters as part of the Franco-British helicopter agreement of October 1967. Certificated by the Services Officiels Français on 9 October 1970.

Turmo IIIC$_5$, IIIC$_6$, IIIC$_7$

Similar to Turmo IIIC$_3$, but with different ratings. Total of 549 to power SA 321F, G, H and Ja, not including the WZ6 and industrial WZ6G made under licence by CLXMW in China.

Turmo $IIIE_6$

Similar except for material of gas generator turbine, which is improved to allow higher gas temperatures.

Turmo IV

The Turmo IVA is a civil engine derived from the $IIIC_4$, with a maximum contingency rating of 1,417 shp. The IVB is a military version having the same ratings as the $IIIC_4$. The IVC is the only version still in production; it is made under licence by Turbomecanica to power the IAR-built Puma.

Type
Free-turbine turboshaft.

Intake
Direct forward-facing circular intake, with accessory drive pads above and below.

Compressor
Single-stage axial followed by single-stage centrifugal with single-sided impeller. Two rows of light alloy stator blades aft of axial stage. Centrifugal stage has steel radial and axial diffusers; impeller located by lugs on turbine shaft. Axial rotor blades, titanium in Turmo $IIIC_3$, C_5 and E_3 and steel in Turmo $IIIC_4$, pin-mounted in steel disc with integral shaft. Pressure ratio 5.9 on Turmo $IIIC_3$. Air mass flow 5.9 kg (13 lb)/s. Axial rotor carried on ball bearing ahead of disc and roller bearing aft of disc. Also, ball bearing ahead of impeller.

Combustion Chamber
Reverse-flow annular type with centrifugal fuel injector using rotary atomiser disc. Ignition by two ventilated torch igniters.

Compressor Turbine
Two-stage axial unit with integral rotor blades. Discs with curvic couplings through-bolted to compressor shaft. Carried on roller bearing at rear of second-stage disc.

Power Turbine
Two-stage axial unit in Turmo $IIIC_3$, C_5 and E_3, and single-stage in Turmo $IIIC_4$. Blades carried in discs by fir-tree roots. Rotor overhung from rear on through-bolted output shaft. Shaft carried on roller bearing at front (at rear of turbine disc) and ball bearing at rear (at input to reduction gear). In all advanced production engines of $IIIC_4$ derivation the power turbine speed is 22,840 rpm under all high-power conditions.

Jetpipe
Fixed type with lateral bifurcated exhaust duct in Turmo $IIIC_3$, C_5 and E_3, and single lateral duct on Turmo $IIIC_4$.

Mounting
Two lateral supports fitted to lower part of turbine casing at rear flange output shaft protection tube. On Turmo $IIIC_4$, also on reduction gear case.

Output
$IIIC_3$, C_5 and E_3 fitted with rear-mounted reduction gear; $IIIC_4$ direct drive.

Accessories
Mounted above and below intake casing with drive pads for oil pump, fuel control unit, electric starter, tachogenerator and, on Turmo $IIIC_4$, oil cooler fan. Control unit remove drive also provided on Turmo $IIIC_4$ from bevel-gear drive on power turbine output shaft.

Starting
Automatic system with electric starter motor.

Control System
Fuel control unit for gas generator on Turmo IIIC$_3$, C$_4$ and E$_3$, with seed limiter for power turbine also fitted on E$_3$. Constant-speed system fitted on Turmo IIIC$_4$ power turbine, with speed limiter also fitted on gas generator.

Fuel Specification
AIR 3405 for Turmo IIIC$_4$.

Oil System
Pressure type with oil cooler and 13 litre (3.43 US gallon; 2.86 Imp gallon) tank at front of engine on Turmo IIIC$_4$, with oil tank only around intake casing on Turmo IIIC$_3$, C$_5$ and E$_3$, and by intake accessory drive gear on Turmo IIIC$_4$.

Oil Specification
AIR 3155A, or synthetic AIR 3513, for Turmo IIIC$_4$.

Dimensions

Length:	
Turmo IIIC$_3$, C$_5$ and E$_3$	1,975.7 mm (78.0 in)
Turmo IIIC$_4$	2,184 mm (85.5 in)
Width:	
Turmo IIIC$_3$, C$_5$ and E$_3$	693 mm (27.3 in)
Turmo IIIC$_4$	637 mm (25.1 in)
Height:	
Turmo IIIC$_3$, C$_5$ and E$_3$	716 mm (28.2 in)
Turmo IIIC$_4$	719 mm (28.3 in)

Weight, Dry

Turmo IIIC$_3$ and E$_3$ fully equipped	297 kg (655 lb)
Turmo IIIC$_5$, IIIC$_6$ and IIIC$_7$	325 kg (716 lb)
Turmo IIIC$_4$, equipped engine	225 kg (496 lb)

Performance Ratings

T-O:	
Turmo IIIC$_3$ and E$_3$	1,104 kW (1,480 shp)
Turmo IIIE$_6$	1,181 kW (1,584 shp)
Max contingency:	
Turmo IIIC$_4$ at 33,800 gas generator rpm	1,032 kW (1,384 shp)
Turmo IIIC$_6$ at 33,550 gas generator rpm	1,156 kW (1,550 shp)
Turmo IIIC$_7$ at 33,800 gas generator rpm	1,200 kW (1,610 shp)

Turmo IVA at 33,950 gas generator rpm	1,057 kW (1,417 shp)
Turmo IVC at 33,800 gas generator rpm	1,163 kW (1,560 shp)
T-O and intermediate contingency:	
Turmo $IIIC_5$	1,050 kW (1,408 shp)

Specific Fuel Consumption

At T-O rating:	
Turmo $IIIC_3$ and E_3	101.9 µg/J (0.603 lb/h/shp)
At max contingency rating:	
Turmo $IIIC_4$, C_5, C_6, C_7 and IV	106.8 µg/J (0.632 lb/h/shp)
Turmo IVA	106.3 µg/J (0.629 lb/h/shp)

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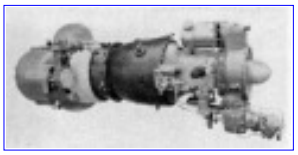
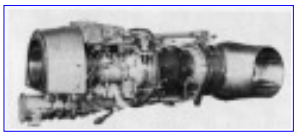


Turmo $IIIC_3$



Turmo $IIIC_4$

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AERO-ENGINES - TURBOSHAFT, FRANCE

Date Posted: 30 August 2000

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SOCIETE TURBOMECA

TURBOMECA TM 333

This turboshaft was launched in July 1979 to power the AS 365 and other helicopters in the 4,000 kg (8,800 lb) class, including the Indian ALH. French certification of the 1A version was obtained on 11 July 1986.

TM 333 1A

Basic version, comprising three modules: gas generator, free power turbine and reduction gear. Max contingency 788 kW (1,057 shp), T-O 747 kW (1,001 shp), max continuous 663 kW (889 shp). Powered AS 365/565 Dauphin.

TM 333 1M

Same ratings. Powered AS 565 Panther.

TM 333 2B

Growth version with single-crystal HP turbine. OEI 30 s 861 kW (1,155 shp), OEI 2 min 784 kW (1,051 shp), T-O/continuous OEI 741 kW (994 shp), max continuous 666 kW (893 shp). Powers HAL (India) ALH, which first flew on 20 August 1992. Certificated December 1993.

TM 333 2B2

Further uprated. OEI 30 s 931 kW (1,248 shp), OEI 2.5 min 839 kW (1,125 shp), T-O/continuous OEI 801 kW (1,074 shp), max continuous 711 kW (954 shp). In July 1999, HAL placed an order for 30 TM 333 2B2 engines

for the first batch of production helicopters, comprising 12 ALHs of five versions.

TM 333 2E

Dual-channel DECU. Described as `about 9 per cent more powerful than TM 333 2B'.

The TM 333 is one of three new engines included in the European Small Engines Co-operation Agreement. Another partner is Techspace Aero of Belgium.

Type
Free turbine turboshaft.

Compressor
Variable inlet guide vanes, two-stage axial compressor, single-stage centrifugal. Mass flow and pressure ratio not disclosed.

Combustion Chamber
Annular, reverse flow.

Compressor Turbine
Single-stage with uncooled inserted blades.

Power Turbine
Single-stage axial with uncooled inserted blades.

Output
Two stages to give drive at 6,000 rpm to front output shaft.

Control System
Microprocessor numerical control (essentially means FADEC).

Oil System
Independent system. Oil passes through gear pump and metallic cartridge filter.

Dimensions

Length, including accessories	1,045 mm (41.1 in)
Height overall	712 mm (28.0 in)
Width	454 mm (17.9 in)

Weight, Dry

TM 333 1A/1M	156 kg (345 lb)
TM 333 2B	167 kg (367 lb)

Performance Ratings
see list of variants

Specific Fuel Consumption

TM 333 1A/1M:	
Max contingency	88 Ág/J (0.523 lb/h/shp)

T-O	89.4 Ág/J (0.529 lb/h/shp)
Max continuous	91.7 Ág/J (0.543 lb/h/shp)
TM 333 2B:	
60 per cent T-O power	99.1 Á/g (0.587 lb/h/shp)

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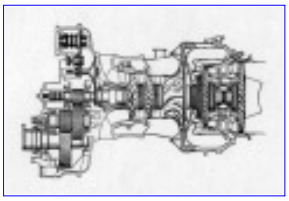
TM 333 2B



Rotating assemblies of the TM 333

Twin TM 333 1A engines in Dauphin 363C (cowlings removed)

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AERO-ENGINES - TURBOJET, FRANCE

Date Posted: 30 August 2000

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SOCIETE TURBOMECA

TURBOMECA MARBORÉ

The Marboré turbojet is still the most widely used of Turbomeca's range of gas turbines. Designed in 1949-1950, it was a direct scale of the company's first engine for aircraft propulsion, the 0.98 kN (220.5 lb st) Piméné of 1948. It first ran in 1950, and the 2.94 kN (661 lb st) Marboré I powered the Gémeaux II on 16 June 1951.

Marboré II

Rated at 3.92 kN (880 lb st) at 22,600 rpm. Fitted to many aircraft, notably including the Fouga Magister twin-jet trainer. When production of the Marboré II ceased in 1979, a total of 4,353 of this 3.91 kN (880 lb st) version had been delivered by Turbomeca and a further 10,000 by Continental Aviation and Teledyne CAE (see US section) as the J69. Production of the Marboré IID continued for the Aérospatiale CT.20 target drone, and a version designated WP11 is still being produced by the propulsion department of Beijing University of Aeronautics and Astronautics.

Marboré VI

This version received type approval in June 1962. Four versions, each with differing accessory arrangements, have been delivered; the Marboré VIC for the Morane-Saulnier Paris II, the Marboré VID for the Aérospatiale M.20 drone, the Marboré VIF for the CM.170 Super Magister, and the Marboré VIJ for the Morane-Saulnier Paris IA. During 1968, the TBO for the Marboré VIF2 was increased to 1,000 hours. Production of the Marboré VI by Turbomeca was completed at 1,194 engines in 1979. The Marboré VI was also built under licence in Spain by ENMASA as the Marboré M21.

The following particulars relate primarily to the Marboré VI series:

Type
Single-shaft centrifugal-flow turbojet.

Intake
Annular sheet metal nose intake bolted to front of light alloy compressor casing.

Compressor
Single-sided impeller machined from two alloy forgings, shrunk on steel shaft and locked and dowelled to maintain alignment. Externally finned light alloy compressor casing supports front ball-bearing for rotating assembly in a central housing supported by three streamlined struts. This housing also contains gears for accessory drives. Pressure ratio 3.84:1 Air mass flow 9.8 kg (21.6 lb)/s (Marboré II, 8.0 kg; 17.6 lb)/s.

Combustion Chamber
Composed of inner and outer sheet metal casings, forming annular flame tube. Air from compressor passes through both radial and axial diffuser vanes and divides into three main flows, two primary for combustion and one secondary. Two primary flows enter combustion zone from opposite ends of chamber, the rear stream through turbine nozzle guide vanes which it cools. Secondary flow enters through outer casing for dilution and cooling of combustion gases. Two torch igniters.

Turbine
Single-stage turbine with 37 blades with fir-tree root fittings in steel disc. Bolted to main shaft and tail shaft, latter supported by rear roller bearing for rotating assembly. 25 hollow sheet steel guide vanes cooled by part of primary combustion air. Gas temperature 613ºC at 21,500 rpm.

Jetpipe
Inner and outer sheet metal casings, latter supported by three hollow struts. Inner tapered casing extends beyond end of outer casing to induce airflow through struts to cool rear main bearing and inner casing.

Mounting
Four points, with Silentbloc rubber mountings, two at front and two at rear.

Accessories
Gear casing in central compressor housing with drives for fuel and oil pumps. Connecting shaft to underside of accessories gear case above compressor casing. Accessories include tachometer generator and electric starter. Take-off (4 hp continuous) for remotely driven accessory box.

Starting
Air Equipement 24 V electric starter or compressed air starter. Two Turbomeca igniter plugs.

Control System
Fuel, pumped through hollow impeller shaft, is fed to combustion zone by rotating injector disc around periphery of which are number of vents which act as nozzles. Fuel is vented by centrifugal force, being atomised in the process. Fuel delivery at low thrust settings regulated by bypass valve.

Fuel Specification
AIR 3405 (JP-1).

Oil System
Pressure type. Single gear-type pump serves front gear casing, two main bearings and rpm governor. Three scavenge pumps return bearing oil to tank via cooler. Normal oil pressure 2.8 kg/cm^2 (40 lb/sq in).

Oil Specification
AIR 3512 (mineral) or AIR 3513A (synthetic).

Dimensions

Length with exhaust cone but without tailpipe	1,416 mm (55.74 in)

Width	593 mm (23.35 in)
Height	631 mm (24.82 in)

Weight, Dry

Equipped	140 kg (309 lb)

Performance Ratings (ISA, S/L)

T-O, 21,500 rpm	4.71 kN (1,058 lb)
Cruising, 20,500 rpm	4.11 kN (925 lb)

Specific Fuel Consumption

At cruising rating	30.3 mg/Ns (1.07 lb/h/lb)

UPDATED

Marboré VI

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AERO-ENGINES - TURBOPROP, FRANCE

Date Posted: 30 August 2000

Jane's Aero-Engines 08

SOCIETE TURBOMECA

TURBOMECA BASTAN

The Bastan turboprop is one of the second-generation of Turbomeca engines, which are characterised by their two-stage axial-centrifugal compressor.

Bastan VIC

Powers the Aérospatiale N 262 and was certificated by the Services Officiels Français and the FAA in 1964. The **Bastan VID** powers the Argentine GII.

Bastan VIII

Powers the Aérospatiale Frégate. The Bastan VII was certificated by the Services Officiels Français on 3 August 1970.

Over 600 Bastan engines were delivered, production being completed in 1978. By 1975 most Bastan engines had a TBO of 3,000 hours.

Type
Single-shaft turboprop.

Intake
Annular intake at rear of reduction gear casing. Outer wall of intake, of triangular cross-section, provides mounting for accessories. Front ball bearing for compressor shaft carried by air intake assembly.

Compressor
Single axial stage for Bastan VIC, and two axial stages for Bastan VII, followed by single centrifugal stage. Two rows of diffuser vanes between axial stages and two more aft of the centrifugal stage, of which the first is radial

and the second axial. On Bastan VII first axial rotor blades are titanium and pin-mounted in disc, and second axial rotor blades are light alloy integral with disc. Central portion of casing carries rear ball bearing for compressor shaft. Bastan VIC pressure ratio 5.83 and mass flow 4.5 kg (10 lb)/s. Bastan VII pressure ratio 6.68 and mass flow 5.9 kg (13.1 lb)/s. Water-methanol injection in Bastan VIC.

Combustion Chamber
Direct-flow annular type. Usual Turbomeca rotary atomiser fuel injection system. Two torch igniters. Gas temperature before turbine 870~C.

Turbine
Three-stage axial-flow turbine with separate discs. Each turbine preceded by axial-flow nozzle guide vane assembly. Turbine casing houses combustion chamber and turbine nozzle assembly. Supports engine rear roller bearing at rear end.

Jetpipe
Annular welded sheet assembly comprising cylindrical outer casing and central bullet fairing.

Control System
By two governors. One adjusts fuel flow entering engine so that it is maintained at the value set by the power control lever, as a function of the variations of pressure and temperature at the engine air intake. The second governor maintains the propeller rpm at the value set by the rpm control lever, by varying propeller pitch.

Output
Two-stage epicyclic type, inside tapered cylindrical casing at front of engine. Ratio 1:21.0957. Propeller shaft carried in ball bearing at front.

Accessories
Upper pinion train drives dynamo starter, propeller governor and fuel pump with fuel metering device. Lower gear drives electric tachometer transmitter, fuel pump, alternator and 20 kVA alternator and hydraulic pump. All accessories mounted on intake casing.

Starting
Electric; automatic starter/generator on Bastan VII.

Mounting
Three attachment points, two lateral, one at bottom of engine.

Dimensions

Length:	
VIC	1,549 mm (60.95 in)
VII	1,911 mm (75.2 in)
Diameter (VII)	550 mm (21.7 in)
Height (VIC)	775.5 mm (30.53 in)
Width (VIC)	685 mm (26.97 in)

Weight, Dry

Fully equipped:	
VIC	322 kg (710 lb)
VII	370 kg (816 lb)

Performance Ratings (S/L, ISA)

VIC T-O and max continuous:	595 kW (798 shp), 790 ekW (1,060 ehp) at 33,500 rpm
VII T-O	780 kW (1,046 shp), 1,089 ekW (1,460 ehp) at 32,000 rpm, maintained to 40~C or 3,650 m (11,975 ft)

Specific Fuel Consumption (T-O)

VIC	98.07 Ág/J (0.582 lb/h/ehp)
VII	88.46 Ág/J (0.525 lb/h/ehp)

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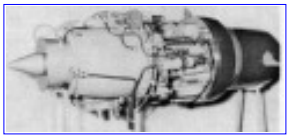
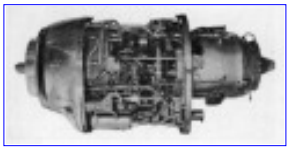


Bastan VI



Bastan VII

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AERO-ENGINES - TURBOFAN, FRANCE

Date Posted: 17 August 2000

Jane's Aero-Engines 08

SOCIETE TURBOMECA

TURBOMECA-SNECMA LARZAC

See under Turbomeca-SNECMA (GRTS).

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AERO-ENGINES - TURBOFAN, FRANCE

Date Posted: 17 December 1999

Jane's Aero-Engines 07

SOCIETE TURBOMECA

TURBOMECA-SNECMA LARZAC

Originally this small turbofan was planned for a wide range of applications, and the first prototype was a 1,000 kg (2,200 lb st) engine aimed at the commercial market. This type of engine ran in May 1969 and began flight development in a pod carried by a Constellation in March 1971. By this time the main immediate market had shifted to military trainers and GRTS designed the Larzac 04 for this purpose. A commercial version, the Larzac 03, was intended for the Aérospatiale Corvette.

In February 1972 the Larzac 04 turbofan was selected for a joint Franco-German programme to provide propulsion for the Alpha Jet trainer. In addition to the two French partners in GRTS, two German companies, MTU and KHD, were added to the programme. Both played a part in the manufacture of prototype engines and the achievement of endurance tests. All four companies shared in production, engines being assembled and tested in both countries. When Belgium bought Alpha Jets FN (now Techspace-Aero) assembled and tested the required engines, giving an overall workshare of: Turbomeca 29.4 per cent, SNECMA 23.0, MTU 22.6, KHD 22.0 and FN 4.0. All versions are of modular design, and intended to minimise noise and harmful emissions.

Larzac 04-C6

Baseline engine for Alpha Jet. Bench testing from May 1972, flight test on Constellation from March 1973 and on Falcon 10 from July 1973, first flight of Alpha Jet 26 October 1973, engine qualified May 1975. Subject of agreement with Teledyne CAE of USA which sought customers for a version designated Model 490-04, but none found.

Larzac 04-C20

Growth version with increased mass flow and higher TET. First run March 1982, first flight December 1982. A small number were delivered from December 1984 for retrofit to Alpha Jets of the Luftwaffe and for prototypes of later versions.

Total production of the Larzac was 1,264 engines, completed in 1988. In late 1996 flight time was 2,300,000 hours. Since then two further versions have been announced:

Larzac 04-R20

Derived from 04-C20 to power MiG-AT advanced trainer, first flown in March 1996. It was the intention that production engines should be made under licence by Chernyshov, Moscow Machine-Building Production Association.Chernyshov has now developed its own RD-1700 engine for this application.

Larzac 04-V3

Intended for Polish M-95, cancelled 1997.
TYPE: Two-shaft turbofan, or bypass turbojet.
INTAKE: Simple circular aluminium inlet, without radial struts or inlet guide vanes.
FAN: Two-stage, with short blades carried in rings together and overhung ahead of the front bearing. Mass flow 28 kg (62 lb)/s. Bypass ratio 1:15.
COMPRESSOR: Four-stage, with blades mounted in rings carried on conical drive shaft. Overall pressure ratio 10.6. Engine mounted on each side of main frame carrying HP shaft front bearing.
COMBUSTION CHAMBER: Annular, with downstream vaporising burners.
TURBINE: Single-stage HP, with cooled blades, and single-stage LP. TET (04-C6) 1,130ºC.
JETPIPE: Plain fixed-area, handling fan and core flows but without mixer.
CONTROL SYSTEM: Hydromechanical, with computer assistance.
ACCESSORIES: Tower shaft from front of HP spool drives gearbox under fan case.
DIMENSIONS:

Length overall	1,179 mm (46.4 in)
Diameter	602 mm (23.7 in)

WEIGHT, DRY:

04-C6	290 kg (640 lb)
04-C20, R20, V3	302 kg (666 lb)

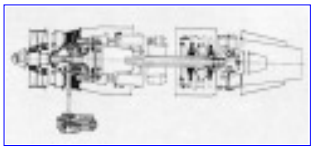
PERFORMANCE RATINGS (T-O, S/L):

04-C4	13.19 kN (2,966 lb st)

04-C20, R20, V3	14.12 kN (3,175 lb st)

SPECIFIC FUEL CONSUMPTION (T-O, as above):

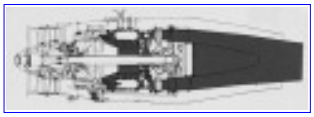
04-C6	20.1 mg/Ns (0.71 lb/h/lb st)
04-C20, R20, V3	20.95 mg/Ns (0.74 lb/h/lb st)



`Exploded' longitudinal section of Larzac 04 (1996)

Larzac 04-R20 (1996)

Larzac 04-C6 (1996)



Longitudinal section through Larzac 04 (1997)

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AERO-ENGINES - TURBOJET, FRANCE

Date Posted: 17 December 1999

Jane's Aero-Engines 07

SNECMA - SOCIETE NATIONALE D'ETUDE ET DE CONSTRUCTION DE MOTEURS D'AVIATION

SNECMA M53

The M53 was designed in 1967-69. Originally called the Super Atar, it represented a fresh attempt to provide a superior engine for highly supersonic aircraft. Among design objectives were modular construction, and a simpler and less costly engine than the SNECMA TF306, derived from the Pratt & Whitney TF30. Accordingly it was made a single-shaft engine, though with an LP compressor handling a slightly larger mass flow than the HP compressor to give a modest bypass ratio. It was expected to enter Armée de l'Air service in 1975 in the Mirage F1 and swing-wing Mirage G4, followed by the twin-engined Mirage 4000. In the event, the only aircraft it powers is the Mirage 2000 in all its versions. The M53 was developed through four stages:

M53

The first of 20 prototype engines began testing in February 1970. The second began in August 1970 and quickly achieved design rpm, and MILitary (maximum dry) rating of 50.96 kN (11,446 lb st) in October 1970. Design maximum afterburning thrust of 83.43 kN (18,740 lb st) was reached in September 1971.

Flight testing began in the right-hand pod of a Caravelle in July 1973, followed in December 1974 by the start of high-Mach testing in the Mirage F1-M53, the aircraft which competed against the F-16 for NATO orders.

M53-2

Initial production version. Solid titanium LP compressor without IGVs with first stage having 21 blades without snubbers. Mass flow 84 kg (185 lb)/s at 10,200 rpm. BPR 0.32. OPR 8.5. T-O rating 83.43 kN (18,740 lb st). Powered Mirage F1-M53, Mirage 4000 and Mirage 2000 prototypes.

M53-5

LP spool unchanged but shaft speed increased. Mass flow 85 kg (187.4 lb)/s at 10,500 rpm. OPR 9.3. T-O rating 88.2 kN (19,830 lb st) with afterburner, maximum dry (MIL) rating 54.4 kN (12,230 lb st). Produced 1980-85 for Mirage 2000.

M53-P2

Originally to have been the M53-7. Growth version under development from 1980 for heavier Mirage 2000 versions. Numerous changes, including a redesigned LP spool with different bearings, discs and aerodynamics, first stage with 23 blades with part-span snubbers. Mass flow 86 kg (189.6 lb)/s at 10,600 rpm. BPR 0.4. OPR 9.8. T-O rating 95.0 kN (21,230 lb st), MIL 64.3 kN (14,455 lb st). In production from 1984, initially for Mirage 2000N.

By January 1999 SNECMA had delivered 675 M53 engines. At that time M53-P2 passed the 700,000 hour mark. Today SNECMA is studying ways of reducing sfc to extend aircraft range, and reducing TET to increase in hot-section life.

TYPE: Low-BPR turbofan (continuous-bleed turbojet).

LP COMPRESSOR: Three stages, with no IGVs or variable stators. See model descriptions for airflow.

HP COMPRESSOR: Five stages, without variable stators. See model descriptions for OPR. Between the compressors is a mid-frame incorporating a front roller bearing and a ball thrust bearing. Construction visible from the two section drawings.

COMBUSTION CHAMBER: Annular, untapered drum type, with several P&W features. Designed for smoke-free operation.

TURBINE: Two stages. The delivery casing incorporates the third bearing.

AFTERBURNER: Three main flameholder gutters with fuel spray rings, the outermost being in the bypass flow. Corrugated perforated liner. Variable 14-flap nozzle with hydraulic actuation.

ACCESSORIES: Mainly grouped on the front of a `banana' external gearbox around the underside of the compressor, driven by a tower shaft passing through the mid-frame.

CONTROL SYSTEM: Dual-digital hydromechanical, monitored by an ELECMA computer.

DIMENSIONS:

Length:

M53-2, -5	4,853 mm (191.0 in)
M53-P2	5,070 mm (199.6 in)

Diameter	1,055 mm (42.0 in)

WEIGHT, DRY:

M53-2	1,420 kg (3,130 lb)
M53-5	1,470 kg (3,240 lb)
M53-P2	1,500 kg (3,307 lb)

PERFORMANCE RATINGS: See model listing

SPECIFIC FUEL CONSUMPTION:

Max dry (MIL) rating, S/L:

M53-2, -5	24.64 mg/Ns (0.87 lb/h/lb st)
M53-P2	25.55 mg/Ns (0.90 lb/h/lb st)

Cutaway drawing of M53-P2 (1997)

Longitudinal section through M53-5 (1996)

Longitudinal section through M53-P2 (upper half shows nozzle in afterburner) (1996)

Two views of M53-P2 (1996)

Two views of M53-P2 (1996)

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AERO-ENGINES - MANUFACTURER, FRANCE

Jane's Aero-Engines 03

GROUPEMENT TURBOMECA-SNECMA (GRTS)

MANUFACTURER DETAILS

2 boulevard du Général Martial Valin, F-75725 Paris Cedex 15
Tel: +33 1 40 60 80 80
Fax: +33 1 40 60 81 02

Announced in March 1969, Groupement Turbomeca-SNECMA is a company formed jointly by Société Turbomeca and SNECMA to be responsible for the design, development, manufacture, sales and service support of the Larzac all-axial small turbofan launched in 1968 as a joint venture by the two companies. Groupement Turbomeca-SNECMA has no capital and primarily comprises a joint management organisation to produce the Larzac engine.

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AERO-ENGINES - TURBOFAN, GERMANY

Date Posted: 22 March 2002

Jane's Aero-Engines 02

MTU AERO ENGINES GmbH

ENGINE 3E

This is the name of a technology programme under which MTU Aero Engines has for many years been exploring an advanced jetliner propulsion concept. As the illustration shows, the LP turbine drives the fan (which might have variable-pitch blades) through a reduction gearbox. There is a sharp kink at the inner diameter of the fan propulsive nozzle.

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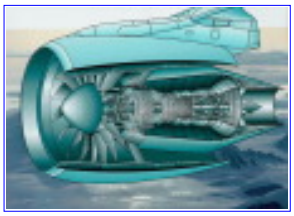

Artwork showing an Engine 3E concept (1998)

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AERO-ENGINES - TURBOFAN, GERMANY

Date Posted: 22 March 2002

Jane's Aero-Engines 02

ROLLS-ROYCE DEUTSCHLAND

BR715

The BR715 is the largest current member of the BR700 engine family. It is designed for 75.6 to 102.3 kN (17,000 to 23,000 lb st).

BR715-58, BR700-715C1-30

Selected for the Boeing 717-200 (previously designated MD-95-30), for which Valujet (now called AirTran) placed a launch order for 50, with 110 engines, in October 1995. By July 2000 firm and option orders for the 717 had reached 300.

An engine of this type first ran on 28 April 1997, soon reaching 110 kN (25,745 lb st), and a 150-hour test was first completed in October 1997. First engine shipped to Boeing Long Beach 26 November 1997. At mid-1998, nine engines had run 1,607 hours and 5,314 cycles at Dahlewitz and at Derby. The first Boeing 717 was rolled out at Long Beach on 10 June 1998, at which time there was a problem with cracking of third HP stage rotor blades. Following strengthening of the containment shroud, the fan-blade-off test was successfully passed at Rolls-Royce Hucknall on 8 July 1998.

One month ahead of schedule, the BR715-58 received JAA certification on 28 August 1998, and FAA certification on 1 September. The first 717-200 began flight testing on 2 September. In 1999, it was decided to offer the BR715-58 at three thrust levels, all for the 717 (see below). EIS October 1999, on AirTran routes from Orlando to the US Midwest and other East Coast cities. European operations began with Olympic in January 2000. In service, fuel burn has been `8 per cent below prediction'. The

SFC figure given below is the original prediction. According to RRD, the BR715 is ``the most efficient, cleanest and most silent in its class, giving the 717 the lowest fuel costs in the 100-seat market''. In summer 2000, Boeing was ``in response to market interest, evaluating smaller 717 versions seating 70 to 86 passengers''.

In 2001 Rolls-Royce held discussions with Boeing, at Long Beach and Seattle (then the company headquarters), and with Boeing Airplane Services at Wichita, regarding the possible market for re-engining the large surviving fleet of MD-80 aircraft. This aircraft could have an installation almost identical to that of the 717. No launch customer had been announced by late September 2001.

In June 1999, BMW Rolls-Royce signed a 10-year agreement with Rolls-Royce Canada under which the latter's facility at Montreal will provide complete overhaul and repair of BR715 engines in North America. The Canadian company is investing up to C$10 million in the necessary tooling and test-cell modifications. The German company will provide full technical support including training. As was done with the first customer, AirTran, operators will be offered Fleet Hour Agreements. Of course Dahlewitz has a 24-hour AOG (aircraft on ground) service made possible by Rolls-Royce's global network of field service engineers.

The core is essentially the same as that of the BR710. Main differences of the BR715 are:

Fan

Single stage with 24 solid titanium blades. Mass flow 267.6 kg (590 lb)/s to 288.5 kg (636 lb)/s, depending on rating. BPR 4.5.

LP Compressor

Two-stage booster rotating with fan. Design and manufacture of the intermediate case, which supports the two front bearings and the fan case, is the responsibility of Volvo Flygmotor (which see).

HP Compressor

Increased efficiency. Overall engine pressure ratio 37.6.

HP Turbine

Single-crystal blades.

LP Turbine

Three stages, with shrouded second stage.

Fan Duct

In production engines, made of light alloy.

Dimensions

Length overall	3,599 mm (142 in)
Inlet diameter	1,564 mm (62 in)
Fan diameter	1,473 mm (58 in)

Weight, Dry

Dressed engine	2,114 kg (4,660 lb)
Complete nacelle	2,792 kg (6,155 lb)

Performance Ratings

(installed)

T-O, ISA+15ºC:	
BR715-58A1	82.3 kN (18,500 lb st)
BR715-58B1	88.97 kN (20,000 lb st)
BR715-58C1	93.42 kN (21,000 lb st)
Cruise 10,668 m (35,000 ft) M0.76	16.01 kN (3,600 lb)

Specific Fuel Consumption

Cruise, (see note above)	17.24 mg/Ns (0.61 lb/h/lb)

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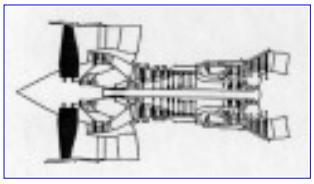
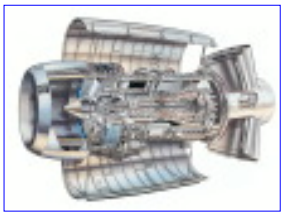


Longitudinal section through BR715

BR715 assembly area (1998)

Cutaway drawing of BR715



Cutaway drawing of BR715 propulsion system

BR715

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AERO-ENGINES - TURBOFAN, GERMANY

Date Posted: 22 March 2002

Jane's Aero-Engines 02

ROLLS-ROYCE DEUTSCHLAND

BR710

This is the baseline engine of the BR700 family, and is designed for 62.27 to 75.6 kN (14,000 to 17,000 lb st). The first complete engine ran on 1 September 1994, and a 150-hour endurance test was successfully completed on 28 February 1995. So far it has gained three applications, on each of which it is the sole power plant:

BR700-710A1-10

T-O rating 65.6 kN (14,750 lb st). Powers Gulfstream V. Programme launched by initial order for 200 engines placed on 8 September 1992. First GV flew 28 November 1995. Engine completed JAA certification in record time on 14 August 1996, with FAA certification following on 18 September. Sir Ralph Robins, Chairman of Rolls-Royce plc, said, ``Certificated one day early, with sfc better than prediction and weight spot-on'', though efforts are being made to reduce the weight of the engine dressing and nacelle. The GV entered service in April 1997 and has demonstrated a dispatch reliability over 99 per cent. In November 1998, Gulfstream ordered an additional 200 engines, bringing total BR710 orders at that time to over 650. By April 2000, 100 aircraft had been rolled out. By July 2000 the 83 GVs in service had logged 105,000 engine hours.

BR700-710A2-20

T-O rating 65.3 kN (14,690 lb st). Powers Bombardier Global Express, the first of which opened its flight test programme on 13 October 1996. JAA/FAA certification 28 January 1997, first aircraft delivered 8 July 1999, 54 delivered by July 2000.

BR700-710B3-40

British Service designation BR710 Mk 101. T-O rating 69.0 kN (15,500 lb st). Powers BAE Nimrod MRA.4. Initial order for 87 engines for 21 existing Spey-powered aircraft being rebuilt for continued RAF service. Marinised engine (for example, magnesium thrust-bearing ring and gearbox replaced by aluminium) with various special features including drives for high-power alternators, minor changes to the FADEC control to interface with the MRA.4 systems, and a new mixer and tailcone to match the long jetpipe. Produced in partnership with prime contractor RR Military Aircraft Engines Ltd. The first engine began testing at Dahlewitz in July 1998. In June 1999 a Mk 101 engine successfully completed a 150-hour endurance test. This paved the way for a 1,200-hour corrosion test in salty atmospheres at the Defence Evaluation Research Agency site at Pyestock, UK, which was completed in December 1999. Altitude testing at Pyestock was completed in late 1998, while calibration and crosswind testing of the inlet took place at Rolls-Royce Hucknall. The Mk 101 was due to complete certification in third quarter 2000, ready for MRA.4 flight test to begin later in the same year. Numerous problems have caused this date to be postponed by two years. Engines will be maintained by RR East Kilbride, Scotland.

In addition to the above applications, different versions of BR710 are candidate engines for several forthcoming or projected aircraft. In June 1997, BMW Rolls-Royce signed an agreement with ANTK Tupolev for engines to power the Tu-354 (previously called Tu-334-200). Russian state help is needed for this application.

Between autumn 1996 and October 1999 Dahlewitz had produced 220 BR710 engines. Of these, 20 are fully instrumented for development and testing, while 200 had at that time been delivered to customers, logging over 50,000 hours by October 1999. In July 2000 the BR710 order book had reached 1,066. The 232 engines then in service had logged almost 160,000 hours.

In November 1998, testing began on the low-emission staged combustor, a central project in the German 3E (environment, economy, efficiency) programme. Results showed NO_x 50 per cent of the current limit, NO 20 per cent and UHC less than 3 per cent. In October 1999, Rolls-Royce launched a programme called Corporate Care to maintain and overhaul all A1-10 and A2-20 engines. This programme is in place at Rolls-Royce Canada's Montreal facilities.

In early 1996, studies went ahead on a possible turboprop version. This has now led to the TP400, described under APA in the International section.

The following data refer generally to A versions. The B3-40 is slightly heavier.

Type

Two-shaft turbofan.

Fan

Light alloy and composite inlet and case. Single-stage fan with 22 solid titanium blades. Mass flow 197 kg (435 lb)/s. BPR 4.2.

Compressor

Derived from V2500, 10 stages with first four having variable stators. Customer/handling bleeds at fifth

and eighth stages. Overall engine pressure ratio 30.

Combustion Chamber

Annular, derived from Trent technology, thermal barrier coating, Z-ring cooling, 20 airspray burners.

HP Turbine

Two stages, with DS shrouded air-cooled blades.

LP Turbine

Two stages, derived from Trent design, optimised for low noise.

Jetpipe

Multilobe core exhaust for rapid mixing with fan flow, integral reverser and single nozzle.

Reverser

Two-door pivot (target) type integrated into nacelle surface.

Accessories

Magnesium gearbox supplied by APT (partnership of Hispano-Suiza and ZFL) under centreline of intermediate case, with integral oil tank. Radial drive from HP spool. Drives for engine dedicated generator, starter, one or two hydraulic pumps, engine oil pump, fuel pumps with FMV, and either one or two IDGs.

Starting

Pneumatic, air motor supplied from APU, ground supply or cross-bleed. Electric power from 28 V DC bus.

Control System

Dual-lane FADEC supplied by RoSEC with independent processors and integrated fault monitoring (BITE).

Emissions

Noise Significantly Below FAR 36 Stage 3, NO_x 30 per cent below ICAO, with advanced combustor designed for further 30 per cent reduction.

Dimensions

(dressed engine)

Length overall	3,409 mm (134 in)
Inlet diameter	1,311 mm (52 in)
Fan diameter	1,219 mm (48 in)
Max height	1,572 mm (62 in)
Nacelle length	5,105 mm (201 in)

Weight, Dry

Dressed engine	1,633 kg (3,600 lb)
Complete nacelle	2,105 kg (4,640 lb)

Performance Ratings

(installed)

T-O, see model listing

Cruise 12,500 m (41,000 ft) M 0.8	10.23 kN (2,300 lb)

Specific Fuel Consumption

Cruise, as above	17.81 mg/Ns (0.63 lb/h/lb)

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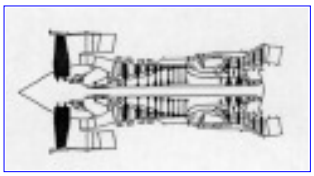
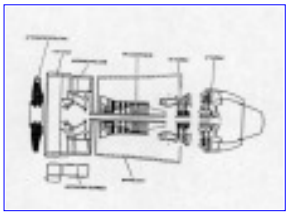
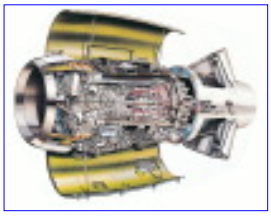


Simplified longitudinal section of BR710

BR710 showing core mixer

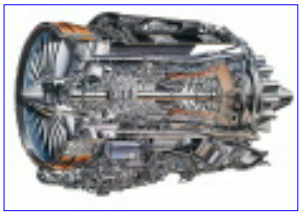


Modules forming BR710



Cutaway drawing of BR700-710A series propulsion system

Cutaway drawing of BR700-710B3-40 propulsion system



Cutaway drawing of BR710

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AERO-ENGINES - MANUFACTURER, GERMANY

Date Posted: 22 March 2002

Jane's Aero-Engines 02

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In October 1999, Rolls-Royce took complete control of the former BMW Rolls-Royce GmbH, in which the British company previously had a 49.5 per cent share. The other 50.5 per cent was exchanged for 33,300,000 RR shares and German car maker BMW AG of Munich said it would purchase a further 90 million RR shares in the open market in order to bring its total shareholding in Rolls-Royce to 10 per cent. RR Deutschland (RRD) has since been run as a German subsidiary of the British aero-engine firm.

The original joint-venture company was established in July 1990. Until late 1998, the head office was located in Oberursel/Taunus, at the facilities of the former KHD company. It had capital of DM250 million, and was an independent German company, created primarily to produce turbofan engines in the thrust bracket 62.27 to 102.3 kN (14,000 to 23,000 lb st). It also took over activities of the former KHD, including the Tornado secondary power system and T117 propulsion for the CL-289 UAV. It had a 20 per cent share in the Rolls-Royce Tay programme, 5 per cent in the Rolls-Royce Trent and 5 per cent in the CFM56-5.

Today RRD has more than 2,000 employees and is the RR `centre of excellence for all two-shaft engines in the thrust bracket 13 to 23 K' [thousands of pounds], though the Tay continues to be manufactured at Derby.

Both partners developed the BR700 series, a new family of turbofans. These entered production at a purpose-built factory at Dahlewitz, 20 miles south of Berlin, in June 1995. As engines come off production they are trucked to the modernised testbeds of MTU (qv) at nearby Ludwigsfelde for pass-off testing. In May 1999, it was announced that a US$8 million expansion was being made to the Dahlewitz plant. Completed in first quarter 2000, it enables capacity to be increased from 220 to 300 engines per year. Actual deliveries were 97 engines in 1998, 200 in 1999, 220 in 2000 and over 250 in 2001.

In early 1996, BMW RR began an urgent project study into a turboprop suitable for the Future Large Aircraft. This engine will now be developed jointly with the former rival team led by Snecma: see A400M under APA in the International section.

RRD is making a major contribution to the Engine 3E 2010 initiative which is targeting major advances in environmental performance and operating efficiency. In July 2000 the company said that the first phase of this programme had produced ``encouraging results. A BR700 core with a staged combustion chamber has delivered a 25 per cent reduction in NO_x compared with the current BR700, which itself has industry-leading levels. The use of advanced blisk compressor stages has resulted in weight reductions exceeding 25 per cent, as well as savings in fuel. Initial elements of 3E 2010 have already proved themselves in the BR715".

In June 1996 the company signed a contract with AVIC of the People's Republic of China for BR parts to be made by XAE (which see, in Chinese section). In November 1996 it signed a further agreement with AVIC providing for much broader long-term co-operation, involving a selection of AVIC research institutes and universities. Later in that month, a second agreement, involving four RRD engineering projects, was signed with Beijing University of Aeronautics and Astronautics. In November 2001 a further agreement was signed with AVIC to support the selection of the BR710 as the engine of the Chinese ARJ21 regional jet.

In early 1998 the company signed an agreement with Rolls-Royce Turbomeca for the future development, production, marketing and customer support of the RTM 322-01/9 turboshaft engine for the NH 90 helicopter family. Engines for NH 90 helicopters purchased by the German government are being assembled and tested at Oberursel. In May 1998, the company announced that `in the coming months' it would relocate its headquarters at Dahlewitz. This took place progressively through 1998. In addition to the RTM 322, Oberursel remains the centre for the small engines listed above, and is also supplying components for the BR700 family.

Rolls-Royce Deutschland is building a global support operation. In October 1999, it announced that RR Canada, which was already maintaining New Jersey-based Executive Jet's Tay engines, had signed a US$45 million contract to maintain the same customer's BR710A1 engines.

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AERO-ENGINES - TURBOSHAFT, GERMANY

Date Posted: 23 October 2001

Jane's Aero-Engines 11

MTU AERO ENGINES GmbH

TURBO-UNION RB199

MTU has a 40 per cent share in this engine, described in the International section.

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AERO-ENGINES - TURBOSHAFT, GERMANY

Date Posted: 23 October 2001

Jane's Aero-Engines 11

MTU AERO ENGINES GmbH

TURBOMECA-SNECMA LARZAC

MTU has a 23 per cent share (see under France). MTU is responsible for the combustion-chamber casing, HP and LP turbine nozzles, HP turbine disc, turbine exhaust casing, exhaust cone and jetpipe.

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Date Posted: 23 October 2001

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MTU AERO ENGINES GmbH

ROLLS-ROYCE TYNE

MTU had a 28 per cent share in about 170 Tyne turboprops for the Transall. MTU supports all Tyne 21 engines (Atlantique) and Tyne 22 (Transall), as well as Tynes used by civil operators.

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AERO-ENGINES - TURBOSHAFT, GERMANY

Date Posted: 23 October 2001

Jane's Aero-Engines 11

MTU AERO ENGINES GmbH

ROLLS-ROYCE 250-C20B

MTU licence-built more than 700 engines, designated 250-MTU-C20B, for German military variants of the Eurocopter BO 105 helicopter. MTU is supporting all 250 Series II engines used by civil operators.

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AERO-ENGINES - TURBOFAN, GERMANY

Date Posted: 23 October 2001

Jane's Aero-Engines 11

MTU AERO ENGINES GmbH

PRATT & WHITNEY PW4084

MTU is a partner with a 12.5 per cent share. It is responsible for the LP turbine. The engine powers the Boeing 777 and is suitable for the A330 Advanced.

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Date Posted: 23 October 2001

Jane's Aero-Engines 11

MTU AERO ENGINES GmbH

PRATT & WHITNEY PW2000

MTU is a partner, with FiatAvio, in the PW2037 and 2040. It is responsible for the LP turbine, under a 21.2 per cent share.

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AERO-ENGINES - TURBOFAN, GERMANY

Date Posted: 23 October 2001

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MTU AERO ENGINES GmbH

P&WC PW500

MTU has a 25 per cent share in this Canadian turbofan. It is responsible for the LP turbine.

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AERO-ENGINES - TURBOFAN, GERMANY

Date Posted: 23 October 2001

Jane's Aero-Engines 11

MTU AERO ENGINES GmbH

P&WC PW300

MTU has a 25 per cent share in this Canadian turbofan. One of its responsibilities is the LP turbine.

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Date Posted: 23 October 2001

Jane's Aero-Engines 11

MTU AERO ENGINES GmbH

PRATT & WHITNEY JT8D-200

MTU has a 12.5 per cent share, being largely responsible for the manufacture of the HP compressor and both HP and LP turbines.

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Date Posted: 23 October 2001

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MTU AERO ENGINES GmbH

MTU-TURBOMECA-RR MTR390

Details of this three-nation helicopter engine are given in the International section.

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Date Posted: 23 October 2001

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MTU AERO ENGINES GmbH

MTFE

This high-bypass turbofan (Mid-Thrust Fan Engine) was announced in February 1994. To be rated in the 53.3 to 99.79 kN (12,000 to 22,000 lb st) class, it was stated to be a co-operative venture with ` SNECMA, GE, Pratt & Whitney and other participants'. This has been overtaken by recent developments, see PW6000 (US section) and JV (International).

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AERO-ENGINES - TURBOPROP, GERMANY

Date Posted: 23 October 2001

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MTU AERO ENGINES GmbH

M138

MTU is a partner in this proposed turboprop for the A400M (described under AMC in the International section).

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AERO-ENGINES - TURBOFAN, GERMANY

Date Posted: 23 October 2001

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MTU AERO ENGINES GmbH

IAE V 2500

MTU has a 12.0 per cent share in IAE (described in the International section).

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Date Posted: 23 October 2001

Jane's Aero-Engines 11

MTU AERO ENGINES GmbH

GENERAL ELECTRIC CF6

MTU has a 10 per cent share in the manufacture of the CF6-50 for the A300, a 12 per cent share of the CF6-80A/A1 for the A310 and 767 and a 9 per cent share of the CF6-80C2 for the A300-600, 747 and 767 and of the CF6-80E1 for the A330. MTU makes HP turbine parts.

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AERO-ENGINES - TURBOFAN, GERMANY

Date Posted: 23 October 2001

Jane's Aero-Engines 11

MTU AERO ENGINES GmbH

EUROJET TURBO EJ200

MTU has a 33 per cent share in this engine, described in the International section.

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AERO-ENGINES - MANUFACTURER, GERMANY

Date Posted: 23 October 2001

Jane's Aero-Engines 11

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MTU Aero Engines, until 2001 MTU München, is a subsidiary of Deutsche Aerospace AG. It states that it is Germany's largest aero-engine company and the world's leading independent aero-engine maintenance company. It produces engines for most classes of aircraft. In March 1990, it signed an agreement which gives Pratt & Whitney a firm foothold in Europe and makes MTU United Technologies `preferred partner worldwide'. Despite this, the refurbished MTU engine testbeds at Ludwigsfelde are being used for production testing of BR710 engines.

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AERO-ENGINES - TURBOFAN, INDIA

Date Posted: 01 May 2001

Jane's Aero-Engines 10

GAS-TURBINE RESEARCH ESTABLISHMENT

GTRE Kaveri

This engine is planned as the power plant of the production LCA (Light Combat Aircraft), replacing the General Electric F404-F2J3 used in the first LCA demonstrator. Although influenced by existing engines, the Kaveri is a completely Indian project. It was preceded by the following development engines:

GTX37-14U: This afterburning turbojet was the first designed in India. First run in 1977, it had a three-stage LP compressor and seven-stage HP compressor, both driven by single-stage turbines. It was flat rated to ISA +30ºC at 44.5 kN (10,000 lb st) dry and 64.3 kN (14,450 lb st) with full reheat. A few engines will continue running to support later variants.

GTX37-14UB: Turbofan version with bypass ratio of 0.215. Maximum thrust 88.9 kN (19,990 lb st) with a larger frontal area.

GTX-35: Advanced turbojet with five-stage HP compressor, new annular combustor and increased turbine temperature. Offered required thrust for LCA, but higher fuel consumption due to higher thrust levels.

Kaveri: Improved turbofan planned as engine for LCA. Earlier designation of GTX-35VS changed to Kaveri with a redesigned core compressor of six stages, updated full-authority digital engine control of Indian design and advanced nozzle. The core engine first ran in March 1995, and the first complete engine in September 1995.

In the third quarter of 1998 an engine to the latest standard underwent various tests at Cima, near

Moscow.

By 2001 several of a planned total of 17 development engines were on test in Russia and India, and an engine installed in a pod under a Tu-16 was to begin flight testing in January 2000, but will now begin in October 2001. In February 2001 Defence Minister George Fernandes recommended that India ``should establish links with other Asian countries and South Africa to further the LCA``. Previously, BAE Systems of the UK and Sukhoi of Russia had expressed an interest in creating LCA partnership agreements.

Type

Low-BPR turbofan with afterburner.

LP Compressor

Three stages with transonic blading. Pressure ratio 3.4.

HP Compressor

Six stages with variable IGVs and first two stators. Pressure ratio 6.4, giving OPR 21.5.

Combustion Chamber

Annular, with dump diffuser and air-blast fuel atomisers.

HP Turbine

Highly loaded single stage with cooled blades of DS material. Entry gas temperature 1,214 to 1,427ºC (1,487 to 1,700K).

LP Turbine

Single-stage, cooled.

Control System

FADEC developed at GTRE in collaboration with HAL.

Starter

HAL-manufactured jet-fuel starter.

Performance Ratings

Flat rated to S/L ISA+20ºC:	
Max dry (MIL)	52.0 kN (11,687 lb st)
Max afterburner	81.0 kN (18,210 lb st)

Specific Fuel Consumption

Max dry	22.09 mg/Ns (0.78 lb/h/lb st)
Max afterburner	57.50 mg/Ns (2.03 lb/h/lb st)

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GTRE Kaveri on testbed (1998)

GTRE Kaveri on display (1999)

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AERO-ENGINES - MANUFACTURER, INDIA

Date Posted: 01 May 2001

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GTRE - GAS-TURBINE RESEARCH ESTABLISHMENT

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Project Director, Kaveri: T K Sampathkumaran

Established in 1959, the GTRE is one of the major R&D establishments administered by the DRDO (Defence Research & Development Organisation). By far its biggest challenge is the design and development of a new engine for India's indigenous fighter aircraft, the Light Combat Aircraft.

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AERO-ENGINES - MANUFACTURER, INDIA

Jane's Aero-Engines 03

HINDUSTAN AERONAUTICS LTD - HAL

MANUFACTURER DETAILS

PO Box 5150, 15/1 Cubbon Road, Bangalore 560 001
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CHAIRMAN: R N Sharma

The Bangalore Engine and Koraput Divisions of HAL constitute the main aero-engine manufacturing elements of the Indian aircraft industry.

BANGALORE COMPLEX (Engine Division)

This division manufactures under licence the Adour 811 (RRTI), TPE331-5-252D (AlliedSignal) and Artouste IIIB (Turbomeca). These engines, as well as Avon, Dart, Orpheus and Gnome engines, are repaired and overhauled. This division has started production of the PTAE-7 UAV (target) engine of 3.43 kN (772 lb st).

KORAPUT DIVISION

This Division was established to manufacture under Soviet government licence the Tumanskii R-11 afterburning turbojet. With help from the Soviet Union, the first engine was run in early 1969. In 1977 production switched to the R-25 for the MiG-21bis, followed in 1984 by the R-29B for the MiG-27M. R-11 and R-25 engines are repaired and overhauled, joined in 1996 by the RD-33.

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AERO-ENGINES - TURBOSHAFT, INTERNATIONAL

Date Posted: 22 March 2002

Jane's Aero-Engines 02

ROLLS-ROYCE TURBOMECA LIMITED

ROLLS-ROYCE TURBOMECA RTM 322

Rolls-Royce and Turbomeca combined their extensive experience in helicopter gas turbines to produce the RTM 322 family of engines. The launch engine was the RTM 322-01 turboshaft, which is conservatively rated at 1,566 kW (2,100 shp) with easy growth potential to well beyond 2,237 kW (3,000 shp).

The engine family, which will include turboprop and turbofan derivatives, is configured to combine simple design, reliability, low fuel consumption, light weight and low cost of ownership. A turboprop using the RTM 322-01 core would produce 1,193 to 1,491 kW (1,600 to 2,000 shp), with potential for growth to 2,088 kW (2,800 shp). It is therefore suitable for aircraft in the 35 to 70 seat range.

The turboshaft itself has full authority digital electronic control, availability of different output drive configurations, a choice of three starting systems and options for an inlet particle separator and infra-red suppressor. Combined with engine mounts configured for compatibility with a number of existing airframes, these features give the unit a wide range of potential civil and military applications in the 7 to 15 tonne class. Examples are: the Merlin (all versions), EH 101, Sikorsky S-92, Black Hawk and Seahawk series, NH Industries NH90, Westland WS-70 and AH-64 Apache. The engine has been studied by the US Army as a potential growth power plant for the Black Hawk and Apache, and during 1987 the US Navy carried out an operability study in an SH-60B.

RTM 322-01

Baseline development version for helicopters. First run as a complete power plant on 4 February 1985. Ratings: max contingency 1,724 kW (2,312 shp), max T-O 1,566 kW (2,100 shp), cruise 940 kW (1,260 shp). First flown 14 June 1986 in S-70C, which logged over 1,200 hours, followed by SH-60B. UK military certification completed October 1988, followed by civil certification May 1992.

RTM 322-01/8

In 1988, a competition was held to select an engine for all UK EH101 helicopters. In September of that year, the Minister of Defence Procurement announced that the RTM 322 had won this hard-fought evaluation, as it ``provided the best value for money''. The first flight of the RTM 322 on the three-engined EH101 took place on 6 July 1993. Orders have since been placed for this version to power 44 Merlin HM.1 ASW/multirole helicopters for the Royal Navy. This engine has flown over 3,300 hours in EH101 versions. Production engines have been delivered since 1995 to Merlin prime contractor Lockheed Martin (125 by June 2000), the first production Merlin was flown in January 1997 and EIS was achieved in December 1998.

RTM 322-02/8

Physically in many respects identical to the 01/8, the -02/8 is cleared to higher powers. It powers the EH101 Support Helicopter, an initial 22 of which are in production for the RAF as the Merlin HC.3. First rolled out November 1998 and flown 24 December. EIS achieved in December 2000. In April 1999, RRTM and Finavitec signed an agreement under which, should Merlin Mk 3 be selected for the Finnish Army, the Finnish company expects to make components and overhaul and repair engines in Finnish service (as it will do with the 01/9 engines for NH90s, see below).

RTM 322-01/12

Following a second hard-fought evaluation the RTM 322 was selected in June 1995 as the engine of 67 twin-engined WAH-64D Longbow Apache helicopters for the British Army. RRTM claims that the European engine will provide significant advantages in performance and cost of ownership to Apache operators. The engine contract was signed with GKN Westland in April 1996. Prototype engines were delivered from September 1997, and the WAH-64D made its first flight at Boeing's Mesa facility on 29 May 1998. Engine qualification was achieved in March 1999. Hot/high testing was completed in September 1999 without incident and RRTL state ``flight testing has confirmed improved engine handling and performance over the AH-64D''. Flight testing showed ``full operational payload and performance can be maintained in hotter climates and at higher altitudes. Over its full service life the RTM322 promises to be cheaper, more reliable and to retain full performance levels for longer.'' Delivery of Apache helicopters to the British Army began on 15 March 2000. Chuck Velow, Boeing's V-P Apache Programs, said ``We have confidence in offering the RTM 322 as an option to new customers.''

RTM 322-01/9

The RTM322-01/9 is the most powerful of all current versions of the engine and was selected as the baseline for NH Industries' twin-engine NH90 Tactical Transport Helo (TTH) and NATO Frigate Helo

(NFH). This engine has powered all five of the flying prototypes since first flight of PT1 on 18 December 1995 and had completed 720 flying hours by June 2000. A Collaboration Agreement and MoU dealing with engine markets based on German and Italian requirements have been signed with BMW RR (now Rolls-Royce Deutschland GmbH) and Piaggio respectively, though Italy selected the General Electric T700-T6E for its NH90s. Engines for the three German armed forces are being assembled and tested by Rolls-Royce at Dahlewitz. The NH90 performance requirement calls for an OEI rating of 2,106 kW (2,825 shp). During testing at the Bordes facility of Turbomeca in early 1999 an 01/9 recorded a power of 2,331 kW (3,125 shp). RRTI then stated ``The RTM 322 has demonstrated that it is the only engine it its class with the hot and high performance margin and power growth necessary". At the beginning of June 2000 RRTM signed a contract valued at approximately US$1 billion for 01/9 engines for 399 NH90s for France, Germany and Netherlands. Engines for the French armed forces are being assembled and tested by Turbomeca at Bordes. Those for the Netherlands are being assembled and tested by Standard Aero. By 2002 additional 01/9 engines had been selected by Sweden and Finland, as part of the Nordic Standard Helicopter programme.

By June 2000, the RTM 322-01 engine had logged 20,000 hours of bench running and over 8,000 hours of flight test experience. By late 2000, the total had risen to almost 40,000 hours, and was rising sharply. Nearly 1,300 engines were then on order. Engines in UK service are supported by RR Ansty, DARA Fleetlands and Turbomeca at Tarnos.

The following applies generally to all versions.

Type

Free-turbine turboshaft.

Intake

Anodised aluminium casting with circular forward-facing inlet. Central bearing for output shaft carried on four radial struts (01/8, three). RTM 322-01 has inlet particle separator made from cast aluminium, stainless steel and glass-carbon/carbon epoxy composite.

Compressor

Three axial blisk stages followed by single centrifugal impeller, all made from one-piece titanium forgings. Mass flow 5.75 kg (12.69 lb)/s. Pressure ratio 14.7.

Combustion Chamber

Annular reverse-flow. Outer case of Inconel, flame tube welded from Nimonic sheet. Lucas ignition exciter.

Compressor Turbine

Two-stage. Discs made from Inconel forgings. First-stage nozzles and blades air-cooled. Second-stage nozzles cooled, uncooled blades. All rotor blades single crystal.

Power Turbine

Two-stage. Discs made from Inconel forgings. Blades cast Inconel with integral tip shrouds. Drive to front or rear.

Output

No integral gearbox. Output shaft from the intake centre runs at power-turbine speed.

Accessories

On intake case, driven by gas generator shaft. Provides drives for LP and HP fuel pumps, oil pump, alternator and tachogenerator.

Starting

Electric, with high-energy ignition unit serving two igniters with long and short high-tension leads from Lucas Aerospace exciter.

Control System

FADEC sends signals to the hydraulic and metering components of the hydromechanical fuel system to operate electric actuators and solenoids controlling fuel flow.

Dimensions

Length	1,171 mm (46.0 in)
Diameter (envelope)	736 mm (28.98 in)
Diameter (over particle separator)	604 mm (23.8 in)

Weight, Dry

01/8, 02/8	248 kg (546.7 lb)
01/12	249 kg (549 lb)
01/9	227 kg (500.4 lb)

Performance Ratings

T-O (intermediate contingency):	
01/8, 01/12	1,566 kW (2,101 shp)
02/8	1,670 kW (2,241 shp)
01/9	1,735 kW (2,327 shp)

Specific Fuel Consumption

T-O, as above:	
01/8	74.4 Ág/J (0.441 lb/h/shp)
02/8	74.59 Ág/J (0.442 lb/h/shp)
01/12	73.2 Ág/J (0.434 lb/h/shp)
01/9	70.87 Ág/J (0.420 lb/h/shp)

Contract Price

June 2000, engines for 399 twin-engined NH90, `about US$1 billion'.

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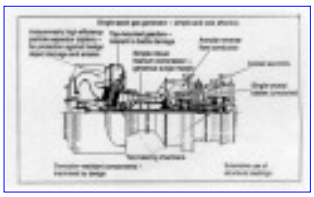
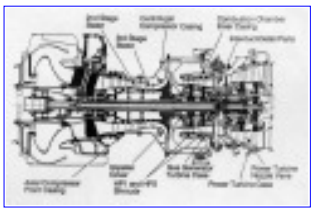
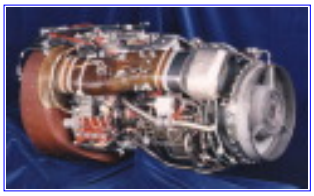


Major features of basic RTM 322-01



Additional features of 322-01

Cutaway drawing of basic RTM 322-01 with particle separator



RTM 322-01/8

RTM 322-01/9

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AERO-ENGINES - TURBOFAN, INTERNATIONAL

Date Posted: 22 March 2002

Jane's Aero-Engines 02

ROLLS-ROYCE TURBOMECA LIMITED

ROLLS-ROYCE TURBOMECA ADOUR

US military designation: F405

The Adour was designed for the SEPECAT Jaguar. The whole engine is simple, robust and of modular design.

Bench testing began at Derby on 9 May 1967. Engines for Jaguars were assembled at Derby (Rolls-Royce) and Tarnos (Turbomeca) from parts made at single sources in Britain and France. Turbomeca makes the compressors, casings and external pipework.

Following selection of the Adour for the Mitsubishi T-2 trainer and F-1 fighter/support aircraft, Ishikawajima-Harima Heavy Industries produced the Adour from 1970 under a licence agreement. In 1972, a non-afterburning Adour was selected to power the British Aerospace Hawk advanced trainer. The Adour has remained in production for this application ever since, though the assembly line is now at Bristol. The current version is the Mk 871. Engines for Australian Hawks are being assembled at Sydney by Qantas.

The following versions have been produced:

Mk 102

Original production engine for Jaguars in service with RAF and Armée de l'Air. Introduced PTA (part-throttle afterburning) giving smooth modulation in thrust over entire power range. Qualified in

1972.

Mk 104

Uprated RT172-26 version similar to Mk 804; RAF Mk 102 engines were converted to this standard.

Mk 106

Major upgrade for retrofit to 61 RAF Jaguars. Under a £70+ million (US$112+ million) contract, Rolls-Royce Bristol carried out engineering and development and the company's East Kilbride, Scotland, plant is remanufacturing Mk 104 engines with the Mk 871 core, Mk 811 afterburner and FADEC control. Thrust is increased by 29 per cent (10 per cent always available and 19 per cent on pilot selection). In May 1999, a Mk 106 successfully completed 130 hours of altitude testing at Saclay, France. Following altitude testing on a second engine close to production standard, two flight engines were delivered to British Aerospace in autumn 1999. The flight-test Jaguar flew with one Mk 106 and one Mk 104 in June 2000, and with two Mk 106 engines in August. Deliveries of 122 Mk 106 engines are due from May 2000 to April 2005. These are intended to keep 61 Jaguars operational until at least 2008, though a crippling shortage of funds may demand their withdrawal earlier.

Mk 151

Non-afterburning version for Hawk. Internal components and certification temperatures identical to Mk 102 and Mk 801A. Qualified in 1975. In November 1996, Rolls-Royce Aero-Engines Services Ltd signed a record 10-year contract to support all RAF Adour engines (Mks 104, 151).

Mk 801A

Similar to Mk 102. Japanese designation TF40-IHI-801A. For Mitsubishi T-2 and F-1. Qualified in 1972.

Mk 804

Uprated engine for Jaguar International. Rating with full afterburner at M0.9 at S/L, ISA, increased by 27 per cent. Qualified in 1976.

Mk 811

Uprated version for Jaguar International. Revised compressor aerodynamics and hot-end improvements. In production by Hindustan Aeronautics, with increasing Indian manufactured content.

Mk 815C

Mk 804 uprated to Mk 811 performance level by conversion at overhaul.

Mk 851

Non-afterburning version of Mk 804 for export Hawk.

Mk 861

Non-afterburning version of Mk 811 for Hawk and Hawk 60. Certificated 1981.

Mk 861-49

Derated version of Mk 861, for prototype McDonnell Douglas/BAe T-45A Goshawk for US Navy. Certificated 1988.

Mk 871

Uprated version for Hawk Series 100 and 200. Fitted with new titanium fan, improved combustor and single-crystal turbine rotor blades. Certificated late 1990. By 1999 the Hawk was in service in 14 countries. Among recent supply and through-life support packages are one priced at £150 million for Australia (33 aircraft) and another priced at approximately £100 million for NATO Flight Training Canada (initially 18 aircraft).

F405-RR-401

US version of Mk 871 with minor changes for production T-45A Goshawk. Flight idle increased from 55 to 78 per cent rpm, modified Lucas fuel control for fast acceleration. Operated to severe high-cyclic usage, all rotating modules on 2,000-hour basis. In 1996, it was announced that responsibility for assembly, test and service support of all F405 engines would be transferred to Rolls-Royce Allison (now RR Corporation) but in fact this transfer did not take place.

Mk 900

Launched in May 1997, this upgraded version features numerous improvements, including: a modified compressor drum, long-life combustion chamber, single-crystal blades in the HP turbine and FADEC control. A major objective is to extend TBO to 4,000 hours. Interchangeable with earlier Adour engines, this internally funded version was being tested in 1999-2001, with qualification scheduled for mid-2002.

Mk 951

Adour Mk 951 engines will power the 24 Hawks ordered by the South African Air Force, which are not due for delivery until 2002-04. All engines in the Mk 900 family are designed to be retrofittable into existing aircraft, and earlier engines can be brought up to this standard.

More than 2,600 Adour engines have been produced, including licensed manufacture in Finland, India and Japan. The two original partners have received orders for over 2,300, with demand from different Hawk variants remaining strong from all parts of the world. Flight hours exceed 5.7 million. Engines in RAF service are supported by RR East Kilbride and DARA St Athan.

Various derived engines called RB.543 have been studied.

Type

Two-shaft turbofan with or without augmentation.

Intake

Formed by forward extension of fan casing. No radial struts or inlet guide vanes.

Fan

Two-stage. Rotating spinner, anti-iced by turbine-bearing cooling air, on front of first-stage disc. Individually replaceable blades. Fixed stators and exit vanes. Unit overhung on spring-loaded ball bearing of squeeze-film type. Full length bypass duct leading to common jetpipe or afterburner. Bypass ratio, 0.75 to 0.80. Mass flow (T-O, S/L, static), Mk 102, 43.14 kg (95.1 lb)/s; Mks 104, 151, 801, 804, 42.64 kg (94 lb)/s; Mks 811, 815, 42.46 kg (93.6 lb)/s; Mk 106, 43.50 kg (95.9 lb)/s; Mk 851, 42.87 kg (94.5 lb)/s; Mk 861, 42.91 kg (94.6 lb)/s; F405-401, Mk 871, 44.68 kg (98.5 lb)/s; Mk 951, 47.63 kg (105 lb)/s.

Compressor

Five-stage compressor on HP shaft. Large diameter double-conical shaft for rigidity with bolted curvic couplings. Wide-chord blades of titanium. Steel stator blades. Overall pressure ratio 11.0 to 11.3.

Combustion Chamber

Annular, with 18 air spray fuel nozzles and two igniter plugs. Lucas engine fuel system.

HP Turbine

Single-stage, air-cooled. Mk 871 and F405 with DS blades.

LP Turbine

Single-stage. Mk 871 and F405 with single-crystal blades. Squeeze-film bearings.

Jetpipe

Mks 151, 851, 861 and 871 have plain fixed-orifice jetpipe handling core and bypass flows. Other marks have a fully modulated afterburner of compact, short-length design incorporating four concentric but staggered spray rings and vapour gutters. Plain annular mixer. Twin rhodium/platinum catalytic igniters between inner gutters. Variable nozzle has eight master and eight slave petals, positioned by eight-sided frame, moved axially by four fuel-operated nozzle rams. Afterburner fuel flow and nozzle system by Dowty Fuel Systems, with vapour-core pump.

Control System

Except Mks 106 and 900, Lucas hydromechanical system derived from Dart. Mks 106 and 900, Hamilton Standard FADEC.

Dimensions

Length overall:	
Mks 102, 104, 106, 801A, 804, 811	2,970 mm (117 in)
Mks 151, 851, 861, 861-49, 871	1,956 mm (77 in)
Inlet diameter (all)	559 mm (22 in)
Max width (all)	762 mm (30 in)
Max height (all)	1,041 mm (41 in)

Weight, Dry

Mk 102, 801A	704 kg (1,552 lb)
Mk 104, 106, 804	713 kg (1,571 lb)
Mk 151	553 kg (1,220 lb)
Mk 851	568 kg (1,252 lb)
Mk 861	577 kg (1,273 lb)
Mk 811	738 kg (1,627 lb)
F405-RR-401	592 kg (1,306 lb)
Mk 871, 900	603 kg (1,330 lb)

Performance Ratings

(S/L T-O)

Mk 102, 801A	32.5 kN (7,305 lb st)*
Mk 104	35.1 kN (7,900 lb st)*
Mk 106 (see above)	38.6 kN (8,690 lb st)
Mk 151, 851	23.1 kN (5,240 lb st)
Mk 804	35.8 kN (8,040 lb st)*
Mk 861	25.4 kN (5,700 lb st)
F405-RR-400	24.2 kN (5,450 lb st)
Mk 811 (dry)	24.51 kN (5,520 lb st)
	37.4 kN (8,400 lb st)*
F405-RR-401	26.6 kN (5,990 lb st)
Mk 871, 900	26.2 kN (5,900 lb st)

**With afterburner*

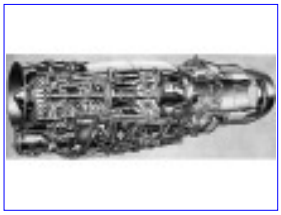
Specific Fuel Consumption

T-O, as above:	
Mk 102	51.08 mg/Ns (1.80 lb/h/lb st)
Mk 811, F405	21.1 mg/Ns (0.78 lb/h/lb st)
Mk 861	21.0 mg/Ns (0.74 lb/h/lb st)
M0.8, 11,890 m (39,000 ft)	typical 27 mg/Ns (0.955 lb/h/lb)

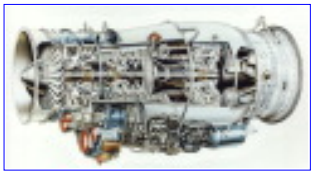
UPDATED

Adour Mk 804 in Jaguar International

Adour Mk 151



Cutaway drawing of Adour Mk 104



Cutaway drawing of Adour Mk 871

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 22 March 2002

Jane's Aero-Engines 02

JSF - JOINT STRIKE FIGHTER

ROLLS-ROYCE CORPORATION
Indianapolis, Indiana 46206, USA
GENERAL ELECTRIC AIRCRAFT ENGINES
Evendale, Ohio 45215, USA
PRATT & WHITNEY, GOVERNMENT ENGINES
West Palm Beach, Florida 33410, USA
ROLLS-ROYCE MILITARY AERO ENGINES
Filton, Bristol BS12 7QE, UK

Because of its unique character and importance, this programme is included in this product even though it is not for an engine but for a family of related aircraft. In financial terms, the programme is already the largest in the world, and ultimately it could become twice as large. It involves different versions of two different engines.

Funded primarily by the US Naval Air Systems Command, the JSF (Joint Strike Fighter) programme was created in November 1994 by the merger of two precursor programmes, the CALF (Common Affordable Light Fighter) and JAST (Joint Affordable Strike Technology). The initial objective was to create a multirole tactical aircraft for the US armed forces, but in December 1995 the UK became a partner, and subsequently many other countries have followed or have expressed a wish to do so. To meet the demands of different customers the programme includes a Short-Take-Off and Vertical Landing (STOVL) version. By 1998 this version was expected to account for at least 750 of the initial planned total of 3,002 aircraft for US and UK customers.

At that time these customers were foreseen as: US Air Force, 1,763; US Navy, 480; US Marine

Corps, 609; Royal Air Force (90+, see below); and Royal Navy, 60. These aircraft would replace (at least) the F-15/15E, F-16, F/A-18, and AV-8B/Harrier/Sea Harrier. Their value, including spares, is tentatively put at US$1,000 billion. The RAF is studying how a JSF could in addition replace the Jaguar and Tornado GR.4, which would increase the RAF buy to considerably more than 300.

For the STOVL version(s) the programme investigated propulsion by a main engine plus either a shaft-driven forward lift fan or a gas-coupled forward lift fan. By mid-1996, the gas-coupled fan had been eliminated, and Allison Engines (now Rolls-Royce Corporation) had started work on a three-stage shaft-driven LiftFan (registered name, see under that company in USA section), and selected Lucas Aerospace (now TRW Lucas Aerospace) to supply the shaft clutch control system. Likewise, there were originally three competing airframe teams: Boeing, Lockheed Martin and McDonnell Douglas/British Aerospace/ Northrop Grumman. The third team, the only one with jet-lift and STOVL combat experience, was eliminated in November 1996, though as McDonnell Douglas was by August 1997 merged into Boeing the latter acquired that company's jet-lift expertise and staff. Lockheed Martin entered into a technology-transfer agreement with Russia's Yakovlev Corporation, and in plan view their final design bears a very close resemblance to the stillborn Yak-43. Subsequently the two rivals each built and tested two different versions (two aircraft in three versions, in the case of Lockheed Martin) as outlined in the accompanying table.

This table lists the original flight-test aircraft, which were powered by versions of the Pratt & Whitney F119 (which as described in the USA section is in production as the engine of the F-22 Raptor). General Electric developed an alternate (that is, alternative) JSF engine, the YF120. From these are being developed refined engines for the production aircraft: the Pratt & Whitney F135 and the General Electric F136.

As had been predicted, on 26 October 2001 the Department of Defense announced that the winner of the JSF programme was Lockheed Martin, teamed with BAE Systems and Northrop Grumman, with a family of aircraft designated F-35. Though it was a ``winner takes all'' contest, Boeing is expected to become heavily involved, and companies in many countries are lobbying for shares of work. It is the intention that production F-35 versions shall be capable, without modification, of accepting either of the alternative engines, and that ``from the pilot's viewpoint there will be no difference''.

The technology involved is discussed and illustrated in the introductory article `Military engines'.

JSF-F119

Derivatives of Pratt & Whitney's F119 engine were selected by the weapon-system contractors to power the Conventional Take-Off and Landing (CTOL) and Short Take-Off and Vertical Landing (STOVL) versions of both families of competing prototype aircraft, the Boeing X-32 and Lockheed Martin X-35. The JSF-F119 was chosen because it was the lowest-risk, lowest-cost approach benefiting from the investment already made by the US government and the maturation of the F119 engine established through development of the Lockheed Martin F-22 Raptor. Pratt & Whitney was awarded a US$801.8 million contract to develop CTOL and STOVL propulsion system variants for each competitor aircraft.

In these applications the engine company found that it could use the bill-of-material F119 core, which is in production for the F-22. The fan and low-pressure turbines are F119 derivatives, which build upon lessons learned in the F-22 development and incorporate the latest in design technologies to meet the requirements of the JSF aircraft.

Pratt & Whitney collaborated with Rolls-Royce Military Aero Engines Ltd (RRMAEL) at Bristol in development of the Boeing X-32 propulsion system. The British company supplied the exhaust ductwork, lift-nozzle module and jet screen for the system. The lift nozzles were activated in the

STOVL mode to provide the primary component of vertical thrust for the aircraft. The jet screen and auxiliary control system provided for manoeuvring capability during STOVL operation.

On the Lockheed Martin X-35 lift/propulsion system, Pratt & Whitney again collaborated with Rolls-Royce to provide STOVL propulsion. In this case the principal partner was Rolls-Royce Corporation (previously Allison), which developed a separate lift fan, LiftFan being a registered trademark. This rotates on a vertical axis, driven directly from the main engine via a shaft and clutch to provide additional lift (vertical thrust) during STOVL operation. For STOVL operation, RRMAEL developed a three-bearing swivel duct, derived from that of the Soyuz R-79V (described in the Russian section), which rotates the main-engine thrust through 90º, as well as roll offtake bleed-air ducts for aircraft control.

Assembly of the first JSF engines started in September 1997, with a view to the first engine going to test at West Palm Beach in early 1998. Indeed, it had hoped to beat this schedule, but in the event ground testing of the F119 version for the CTOL variant of the Lockheed Martin X-35 began on 15 June 1998, with the engine for the CTOL Boeing X-32 following shortly afterwards. Ground testing of the more complex propulsion systems of the STOVL aircraft began in mid-November 1998. Illustrations show the F119-SE614 for the STOVL X-32 running on outdoor test stand C-14, and the F119-SE611, complete with Rolls-Royce's shaft-driven fan, for the STOVL Lockheed Martin X-35 running on indoor test stand A-9.

In 1998-2000 the design of the different versions of X-32 and X-35 was altered on several occasions. These changes had little effect on the engine.

Pratt & Whitney's JSF Programme Manager, Robert Cea, noted that the testing on stand A-9 included Rolls-Royce's three-bearing vectoring main nozzle. Before mid-1999 the complete propulsion system was moved to the specially prepared outdoor stand C-12 to enable thrust to be measured along all six (2×3) axes. The company was particularly pleased that it achieved flight clearance with the original aerodynamic configuration of the LP turbine and fan. All the parts of the four flight-test aircraft came together in 2000. Both the CTOL engines, the JSF119-611C and JSF119-614C, completed their Flight Clearance testing in February/March of that year, and two examples of the relevant engine were delivered to each airframe contractor at Palmdale in March.

The first JSF119-614S STOVL engine was installed in a mere four hours in the first Boeing X-32B in July 2000. At that time Boeing had run 190 ground tests of the X-32B propulsion system, transitioning between CTOL and power-lift modes in between 1 and 3 seconds. In contrast, at this time Lockheed Martin hit a low point, experiencing problems with the JSF119-611S lift fan, including gearbox misalignment, an overheated bearing and a clutch failure. The clutch failed on 12 July 2000; after completing 24 engagements without difficulty, it failed at an overload of 110 per cent (87 per cent of rated engine power, the design target being 82 per cent). The simpler 611C engine, for the X-35A and X-35C, completed 193 hours of ground testing in April 2000.

The first JSF to fly was the Boeing X-32A CTOL version on 18 September 2000. Next came the Lockheed Martin X-35A CTOL on 24 October 2000, and following a brief flight programme (which included an excursion beyond M1.0 and an air refuelling) this aircraft flew back from Edwards to Palmdale to be converted into the X-35B. Meanwhile, Lockheed Martin completed the X-35C, with a larger folding wing and carrier equipment. This version began its flight-test programme on 16 December 2000.

The potentially more challenging STOVL versions were not far behind. Again, Boeing was first, the X-32B rising on engine lift on 19 June 2001. On that date and on the following day this unusual-looking aircraft - which, like the X-32A but unlike the intended production versions, has no horizontal tail surfaces - demonstrated hovering, and transitions between conventional flight and STOVL. On 27 June,

following transfer to NAS Patuxent River, the X-32B made a CTO, accelerated to high-speed flight and finished with a VL at a weight of about 13,200 kg (29,000 lb). Boeing completed its X-32B flight-test programme with Flight 77 on 28 July, still at Patuxent. On more than 100 occasions the tailless aircraft had transitioned from hover to wingborne flight in from one to three seconds, but it had not made a VTO, which is not a requirement. Lacking a lift fan, Boeing had found it difficult to achieve a VTO, but this was achieved after the selection of the Lockheed Martin aircraft.

Meanwhile, Lockheed Martin had begun testing the X-35B in a hover pit at Palmdale in March 2001. On 24 June this aircraft made its first sustained hover. On 9 July it made a VTO, climbed to about 2,743 m (9,000 ft), disengaged the lift fan at 334 km/h (180 kt) and accelerated to M1.08. On 16 July the X-35B made a VTO from Palmdale at about 16,100 kg (35,500 lb) and made its first VL from conventional flight. Pilot Simon Hargreaves noted that at the 760-m (2,500-ft) altitude of Palmdale the lift thrust was approximately 11 kN (2,500 lb) less than at sea level. The final series of tests, all at Edwards or Palmdale, were STOs followed by a supersonic dash and final VL, completed on 30 July. By mid-August 2001 testing of all four aircraft had gone remarkably well, and it was stated with confidence that the CTOL and STOVL versions of both these early demonstrator aircraft could be flown safely by average pilots, and could indeed be the starting point for a major production programme.

Immediately following the JSF downselection, on 26 October 2001, Lockheed Martin was awarded a US$18.98 billion contract for the SDD (System Development and Demonstration) phase. In this phase, though partners listed above are heavily involved, all assembly of aircraft and engines will be by the prime contractors. The SDD and later production X-35 aircraft will all be significantly different from the concept demonstrators. The plan is to start testing the first of 22 SDD aircraft within 48 months of contract signature. Of these, eight will be ground test vehicles, leaving 14 for the flight programme, comprising five F-35A (USAF), four F-35B (USN, and possibly RN/RAF) and five F-35C (USMC and possibly RN/RAF). The first F-35A is expected to be delivered to the USAF Integrated Test Force at Edwards in 2005, closely followed by the first delivery to NAS Patuxent River; seven aircraft are to be based at each test centre.

The first LRIP (low-rate initial production) contract is expected to be signed in 2005, production by a large industrial team (almost certainly to be multinational) thereafter ramping up to an initial 196 per year. This figure is likely to be increased by foreign sales, and by the needs of partners in the manufacturing programme. Already, by January 2002, SDD collaborative partners included Canada, Denmark, Norway and Italy, with Israel, Singapore and Turkey joining as FMS (foreign military sales) participants. The Netherlands had long been discussing joining as a collaborative partner, and at least seven other countries -- including Finland, Poland and Australia -- were engaged in evaluating the prospects. The intention is that all initial aircraft shall be powered by the Pratt & Whitney F135 engine. By 2002 no decision had been announced on whether, or at what point, the alternative GE F136 engine would be brought into the production programme. Production deliveries are expected to begin in 2008.

Available details of the General Electric F136 and Pratt & Whitney F135 will be found in separate entries in the USA section. Entries headed F-35 will be found under Rolls-Royce in the UK section and Rolls-Royce Corporation in the USA section.

Pratt & Whitney JSF119 propulsion systems

Prime contractor	**Boeing**		**Lockheed Martin**		
Engine	SE614C	SE614S	SE611C	SE611S	SE611C

Application	X-32A	X-32B	X-35A	X-35B	X-35C
Configuration	374A	374B	230A	230B	230C
Customer	USAF/USN	USMC/RN/RAF	USAF	USMC/RN/RAF	USN
T-O thrust (K lb)	c41	c18 + 15 + 4	c41	c15 + 18 + 4	c41
First flight	18 Sept 00	19 June 01	24 Oct 00	23 June 01	16 Dec 00
Note: thrust for STOVL versions is in order: main nozzle, subsidiary nozzles or fan, and reaction-control jets.					

JSF-F119: Characteristics:

Type

Twin-spool afterburning turbofan.

Fan

Three stages, integrally bladed rotors, larger than F119.

Core

F119/F-22 Production Module.

LP Turbine

Two-stage design based on F119/F-22.

Augmentor

Advanced Design Based On F119/F-22.

Nozzle

X-32: 2-dimensional convergent-divergent with pitch vectoring. X-35: conventional axi-symmetric nozzle for CTOL X-35A; compact axi-symmetric mounted on three-bearing swivel duct for STOVL X-35B.

Control System

5th generation Full-Authority Digital Electronic Control (FADEC), fully integrated with aircraft vehicle management system.

Performance

X-32: 185 Kn (41,570 Lb St) Class. X-35: 170 kN (38,200 lb st) class.

JSF-F120

The team of General Electric Aircraft Engines at Cincinnati, Allison Advanced Development Company (retaining the original name for military contractual purposes, but actually part of Rolls-Royce Corporation) at Indianapolis and Rolls-Royce Military Aero Engines Ltd at Bristol, for convenience called `GAR', has been called `a historic fighter engine collaboration'. GE and Allison have collaborated since 1993 on the Integrated High-Performance Turbine Engine Technology (IHPTET) programme. This led first to the GE JSF-F120, and by 2001 to the programme for the F136, which is described under Rolls-Royce in the UK section and under GE and Rolls-Royce Corporation in the USA section.

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AERO-ENGINES - TURBOFAN, INTERNATIONAL

Date Posted: 22 March 2002

Jane's Aero-Engines 02

INTERNATIONAL AERO-ENGINES AG

IAE V2500

V2500-A1

In service on the A320. IAE supplies the complete package, including the nacelle (by B F Goodrich/Shorts). Testing of the engine began in December 1985. A flight programme on a Boeing 720B in Canada was completed in 35 hours in spring 1988, and every ingestion and fan-blade-off test was passed first time - believed to be an industry record. The first pair of propulsion systems was delivered to Airbus Industrie in March 1988 and the V2500 was certificated in June that year.

The first V2500-powered A320 flew on 28 July 1988 and entered service in May 1989. A 120-minutes ETOPS approval was awarded in January 1992. Subsequent development to higher thrust has been achieved by increasing the core air flow and aerodynamic changes. Pressure ratio of the LP compressor is increased by adding a fourth stage. All current engines in the A5 (Airbus) and D5 (Douglas) series have common fans and cores.

In 1998 IAE announced the Phoenix package, which upgrades the hot section of V2500-A1 engines by introducing A5 technology. Phoenix changes include redesigned combustor bulkhead segments with improved coatings for longer lives, a thin 1st-stage HP turbine outer air seal for better performance retention, and improved cooling and thermal-barrier coating on the 1st-stage HP turbine vanes (stators) for increased life. The package is expected to increase on-wing time of A1 engines by an average of 25 per cent, with better performance retention and significantly lower maintenance parts costs. The first

Phoenix Standard A1 engine was delivered to America West in January 1999. By late 1999 all indications were that the upgraded engines are meeting expectations.

By January 2002 193 A1-powered aircraft were operating. They had then logged over 8 million hours in over 4.2 million cycles. The lead engine had achieved 36,000 hours on-wing.

V2522-A5

Rated at 97.86 kN (22,000 lb st). Four-stage LP compressor, as in all A5 and D5 versions. Bypass ratio 4.9. Pressure ratio 26.5. For A319.

V2524-A5

Rated at 106.75 kN (24,000 lb st). Certificated on A319 December 1996.

V2527-A5

Flat rated at 117.88 kN (26,500 lb st). Bypass ratio 4.80. Pressure ratio 27.4. For A320. Flight testing from 1992. First delivery (to United) November 1993. An enhanced engine, the **V2527E**, is available for high-altitude operations, while the **V2527M** is tailored to A320 and corporate jet applications. During pre-certification testing the first Airbus Corporate Jetliner (ACJ) flew non-stop from Santiago to Paris, 6,918 n miles (7,969 miles, 12,825 km).

V2525-D5

Flat rated at 111.21 kN (25,000 lb st). Bypass ratio 4.8. Pressure ratio 27.2. For MD-90-30. First delivery (to Delta) April 1995.

V2530-A5

Rated at 139.67 kN (31,400 lb st). Bypass ratio 4.6. Pressure ratio 31.6. For A321-100. Flight testing began March 1993. First delivery (Lufthansa) March 1994.

V2533-A5

Rated at 146.8 kN (33,000 lb st) to power A321-200. Bypass ratio 4.5. Pressure ratio 33.4. Launched by Aero Lloyd April 1995. FAA-certificated August 1996.

In June 1999, IAE announced that it was evaluating a proposed new variant, the **V2500-A7**. This is not at present an active programme.

Since it was introduced to the market the V2500 has gained ground over its competitor. Its market share was 37 per cent in 1997, 55 per cent in 1998, 51 per cent in 1999, 59 per cent in 2000 and 82 per cent in 2001. By mid-1999 a total of 75 customers in 35 countries had ordered over 3,000 engines valued at over US$18 billion. For A5 engines basic SVR in August 2001 was 0.048 per 1,000 hours, IFSD rate 0.003 and dispatch reliability 99.71 per cent. For D5 engines corresponding figures were 0.127, 0.006 and 99.89. By August 2001 over 700 aircraft had been delivered, flying over 17 million hours.

IAE states ``The V2500 offers the lowest fuel burn, the lowest noise and the best overall emissions across the Airbus single-aisle range." The immediate future of the engine could scarcely be better. In

1998 its excellence brought an unprecedented flurry of orders including (March) engines for 90 firm and 87 option A320-family aircraft for a group of South American airlines, (August) a follow-on from United taking its A319/320 fleet to 133, and (September) engines for 59 firm and 129 option A320-family aircraft for British Airways. Significant wins in the first half of 1999 included Spanair and JetBlue (see Contract Price).

The BA order was especially significant. This airline had never previously bought any Airbus aircraft, and had a large 737 fleet, the majority with CFM56 engines. Thus, even if it changed to the A320 family, it might have been expected to stay with the CFM56. According to IAE's Barry Eccleston, ``This competition wasn't just fought on price but on economics''. BA was most concerned with life-cycle costs, and IAE offered an aggressive Fleet Hour Agreement for all off-wing maintenance, accessories and nacelles.

In October 2000 IAE and Airbus announced that the V2527M-A5 had been selected as the reference power plant for the ACJ (corporate jet). The partners stressed the engine's low noise and emissions, and ability to deliver additional range. In 2001 IAE introduced the Vista programme, aimed at improving both products and services. All shareholders support the V2500's long-term progress, and intend to introduce new products when the single-aisle market requires them.

The primary features of the V2500-A5/D5 are as follows:

Type

Two-spool subsonic turbofan.

Fan

Single-stage with wide-chord shroudless blading. Diameter 1,613 mm (63.5 in). Pressure ratio 1.7. Bypass ratio 4.6. Mass flow 389.2 kg (858 lb)/s.

LP Compressor

Four stages, bolted to rear of fan to boost inlet to core. (Three stages in A1 version.)

HP Compressor

Ten stages of blading supported by a drum rotor. Inlet guide and first three vane stages variable. Overall pressure ratio 31.6.

Combustion Chamber

Annular segmented construction eliminates hoop stresses and provides low emissions and uniform exit temperatures.

HP Turbine

Two stages of air-cooled blading (one single-crystal) in powder metallurgy discs. Active tip clearance control.

LP Turbine

Five stages of uncooled blading in welded and bolted rotor. Active clearance control.

Jetpipe

Full length with reverser. Cowl load sharing to minimise case deflections. Acoustically treated.

Accessories

FiatAvio gearbox module on fan case. Sumitomo starter motor.

Control System

Full-Authority Digital Electronic Control (FADEC) to provide command outputs for engine fuel flow, stator vane angle, bleed modulation, turbine and exhaust case cooling, oil cooling, ignition and reverser functions. Supplied by Hamilton Standard, with Woodward fuel management unit.

Dimensions

Length (flange to flange)	3,200 mm (126 in)
Fan diameter	1,613 mm (63.5 in)

Weight, Dry

Bare engine:	
V2500-A5	2,359 kg (5,200 lb)
V2500-D5	2,540 kg (5,600 lb)
Complete nacelle:	
V2500-A5	3,402 kg (7,500 lb)
V2500-D5	3,583 kg (7,900 lb)

Performance Ratings

(installed)

T-O, S/L, ISA	133.4 kN (30,000 lb st) to ISA + 15ºC
Cruise M0.8, 10,670 m (35,000 ft)	25.6 kN (5,752 lb)

Specific Fuel Consumption

Cruise M0.8, 10,670 m; 35,000 ft, installed	16.26 mg/Ns (0.575 lb/h/lb)

Contract Price

November 1996 contract for engines for 31 A320s, US$430 million; March 1997, engines for eight A320s, about US$125 million; price per shipset, about US$13 million; April 1998, engines for 175 A319/320, US$2.3 billion; August 1998, for 188 A319/320, US$2.5 billion; for two A319, US$25 million; early 1999, 21 + 25 A320 family, up to US$540 million; (25 + 50, up to US$900 million; June 1999, 50 A320 family, US$600 million. February 2000, 22 A321, US$265 million, and 20 A320 family, US$240 million.

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Longitudinal section through V2500 showing work split (2000)

Cutaway drawing of V2500-A5 (1998)

V2500

Inspecting a V2500 fan

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 22 March 2002

Jane's Aero-Engines 02

IAE - INTERNATIONAL AERO-ENGINES AG

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IAE is a management company which was set up on 14 December 1983 to direct the entire programme for the V2500 turbofan worldwide. The company is an international consortium comprising: Rolls-Royce 32.5 per cent; UTC (Pratt & Whitney) 32.5 per cent; JAEC (Japanese Aero Engines Corporation, comprising IHI, KHI and MHI) 23 per cent; and MTU (DaimlerChrysler Aerospace) 12 per cent. Each partner is responsible for a particular module allocated according to shareholding. Engines are assembled and tested at Pratt & Whitney and Rolls-Royce.

JAEC is responsible for the fan (derived from that of the RB211-535E4) and LP compressor, Rolls-Royce for the HP compressor and external gearbox, Pratt & Whitney for the combustor, HP

turbine and turbine exhaust case and MTU for the LP turbine.

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AERO-ENGINES - TURBOSHAFT, INTERNATIONAL

Date Posted: 22 March 2002

Jane's Aero-Engines 02

MTU TURBOMECA ROLLS-ROYCE GmbH

MTR 390

This turboshaft engine has been designed to power the Franco-German Tiger UH-T, HAP and HAC helicopters. It is suitable for a range of military and civil applications in helicopters - including the EC400/600, Bell 442, ALH and Lynx - and for fixed-wing aircraft in single and twin installations and for the Cargolifter CL160 airship.

The main characteristics of the modular engine are: ample emergency power reserve for OEI operation, high alternating-output shaft power capability, low fuel consumption under part load, good acceleration, low life-cycle cost, easy handling and simple maintenance. It has a high-performance FADEC, and an engine monitoring system for flight and maintenance crew support.

Design studies have been extended to derivatives for potential future applications. A 6,000 rpm drive version (MTR 390L) and a direct-drive version (MTR 390T) have been defined.

MTR 390 design was completed in 1988, and the first engine run took place at MTU in 1989, two months ahead of schedule, with flight engines for the first Tiger prototypes delivered in September 1990. First flights in a Panther testbed and the prototype Tiger took place on schedule in 1991.

The development programme continues with maturity and mission-simulation testing. A civil certification programme is also in progress. More than 16,000 test hours have been run, including 6,200 in flight. The initial production version was certificated by the JAA in January 1996, followed by LBA civil certification in 1997.

Production Investment (PI) - based on around 1,000 engines for 427 Tiger aircraft - has been confirmed, and MTR is also seeking additional applications. A production contract for 320 engines for

the first production batch of helicopters (80 each France/Germany) was expected in October 1998, but this contract was not signed by the German BWB until 13 January 2000, which seriously delayed production. Under the original programme deliveries of two versions of helicopter, the HAP for France and the PAH-2 for Germany, would have begun in April 1997.

This has slipped by five years, and deliveries actually began in January 2002 with two engines ``delivered on schedule" for an attack/escort Tiger. Under the contract signed in 2000 the associated 332 engines (including 12 spare) would be delivered at the uneconomic rate of three per month, ending in 2011. Fortunately, other customers are appearing, beginning (in August 2001) with an initial 22 for the Australian Army, with considerable local industrial involvement, with deliveries from 2004. Among other near-term possible customers are Spain, Poland and Turkey.

Though the MTR company is owned equally by its principals, the production workshares reflect support by the three governments. Snecma (now owner of Turbomeca) has a workshare of 40 per cent, and is responsible for the compressor system and output gearbox. MTU also has a 40 per cent share, and has been assigned the combustion chamber and HP turbine. Rolls-Royce's share is 20 per cent, its chief contribution being the power turbine. Whether these percentages will vary slightly as a result of participation by customer industries remains to be seen.

Type

Free turbine turboshaft.

Compressor

Two centrifugal stages for erosion and FOD resistance. Mass flow 3.2 kg (7.05 lb)/s. Pressure ratio 13.

Combustion Chamber

Annular reverse-flow, with airblast fuel injectors for low emissions.

HP Turbine

Single-stage gas generator turbine with high performance blade cooling, single-crystal blades and powder-metal disc.

LP Turbine

Two-stage free power turbine with shrouded blades.

Output

Reduces the speed of the power turbine to the output shaft speed of 8,000 rpm.

Accessories

The accessory gearbox in the upper part of the gearbox provides the support and drive for the front- and top-mounted engine equipment.

Control System

FADEC, with engine monitoring system.

Oil System

Integral oil system, with engine-mounted tank and oil cooler with fan.

Dimensions

Length overall	1,078 mm (42.4 in)
Width overall	442 mm (17.4 in)
Height overall	682 mm (26.9 in)

Weight, Dry

	169 kg (372.6 lb)

Performance Ratings

(uninstalled, ISA, S/L)

Super contingency (OEI, 20 s)	1,160 kW (1,556 shp)
T-O	958 kW (1,285 shp)
Max continuous	873 kW (1,171 shp)

Specific Fuel Consumption

T-O	77.74 µg/J (0.460 lb/h/shp)
Max continuous	78.93 µg/J (0.467 lb/h/shp)

Contract Price

Initial 320 engines and associated spares DM430 million (US$223.4 million).

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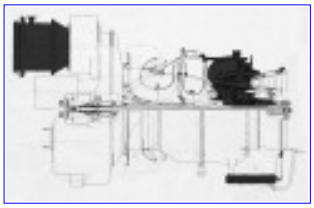
MTR 390

Cutaway of MTR 390



Longitudinal section through MTR 390

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 24 October 2001

Jane's Aero-Engines 11

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Formed in August 1986, Eurojet Turbo GmbH is a consortium of FiatAvio of Italy (Fiat), Industria de Turbo Propulsores of Spain (ITP), MTU of DaimlerChrysler Aerospace in Germany (MTU) and Rolls-Royce of the United Kingdom (RR). Eurojet Turbo GmbH was established to co-ordinate the design, development, manufacture and support of the EJ200 engine for the Eurofighter 2000 (export designation, Eurofighter Typhoon).

Within the consortium, each partner company is fully responsible for its allocated workshare. To make best use of the technologies available in each country, the principle of participation with other partners has been introduced on the majority of the modules. The development workshare is as follows:

Fiat (20 per cent) has responsibility for the LP turbine, LP shaft, interstage support, reheat system, gearbox and air/oil system and is participating in the intermediate casing.

ITP (14 per cent) has responsibility for the convergent/divergent nozzle, the jetpipe, exhaust diffuser, bypass duct and external dressing.

MTU (30 per cent) has responsibility for the LP and HP compressors, and is participating in the HP turbine; it also has overall system design responsibility for the Full Authority Digital Engine Control

unit (FADEC).

RR (36 per cent) has responsibility for the combustion system, HP turbine front-bearing support and intermediate casing, and is participating in the LP and HP compressors, LP turbine, the interstage support and reheat system.

Engine build and test during development and production is at each partner's facilities. Each partner will provide comprehensive support for engines of its own national air force.

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AERO-ENGINES - TURBOPROP, INTERNATIONAL

Date Posted: 18 September 2001

Jane's Aero-Engines 11

AERO PROPULSION ALLIANCE

TP400

A preliminary agreement on the proposed engine for what had become the AMC A400M was finally reached on 30 August 2000. On that date all six participating companies signed a Memorandum of Agreement to produce a single collaborative type of turboprop engine to power that equally international aircraft. The TP400 is described as "the only truly European engine ever designed".

The shareholdings and workshares were agreed as follows:

FiatAvio, 8.0 per cent, responsible for the PGB (power gearbox, transmitting the drive to the propeller).

ITP, 13.6 per cent, responsible for the front frame, intermediate case, hot strut, exhaust case and dressings.

MTU, 24.8 per cent, responsible for the IPT, LPT, HPC, DECU and final assembly and test.

Rolls-Royce, 24.8 per cent, responsible for overall integration and for the IPC, combustor and power shaft.

Snecma, 24.8 per cent, responsible for the HPC (lead partner), HPT, DECU (lead) and AGB (accessory gearbox).

TA, 4.0 per cent, responsible for lubricating-oil equipment.

The original announced customers will be supported by approximately 1,000 engines. As this was written, in mid-2001, the actual contracts for either aircraft or engines had yet to be signed, but at least preliminary design work was at last going ahead in the partner companies. It was then stated that, once these contracts had been signed, the timescale to first engine to test would be 24 months (an almost unbelievably short period), to engine certification 48 months, to A400M first flight 51 months and to first delivery 71 months.

By mid-2001 few details of the TP400 had emerged, though much can be deduced from the excellent cutaway drawing of an installed engine reproduced here. The following are preliminary details:

Type

Three-shaft turboprop.

Intake

Bifurcated, with an oval inlet on each side of the nacelle. From each inlet the duct curves sharply inward, expelling ice to the rear.

Propeller

Single-rotation with eight blades. In 2001 Dowty (Smiths) and Ratier-Figeac were still in competition. The former's R400 propeller has a diameter of 5.3 m (17 ft 6 in), and transmits 7,830 kW (10,500 shp) at 842 rpm, the cruise rpm being 730. Provision to absorb full power in forward or reverse pitch with rapid pitch-change.

Gearbox

Two-stage eipicyclic with single-helical gears. Final drive on engine centreline, incorporating torquemeter.

Compressor

Independent LP and HP axial compressors. The LP spool has five stages, with a constant external diameter. No variable stators. The HP has six stages, the constant-diameter line passing through the mid-span of each blade. IGVs and next two stators variable. OPR 22.

Combustion Chamber

Annular, incorporating latest Rolls-Royce practice for minimal emissions (APA list "smoke, radar, infra-red and gaseous").

Turbine

Three independent turbines. Single-stage HP with aircooled blades rotating at high speed to drive HPC. Single-stage IPT driving the LP compressor. Three-stage LPT with rapidly increasing diameter driving the propeller gearbox. Provision for propeller brake and for immediate fuel cutoff following sudden absence of drive torque.

Jetpipe

Single centreline pipe angled slightly downwards.

Dimensions

Length overall	3,500 mm (137.8 in)

Weight, Dry

	1,830 kg (4,034 lb)

Performance Rating (S/L)

Max T-O	"over 7,457 kW, 10,000 shp, installed"

Pending further details, the two entries for the competing predecessor engines are repeated, to provide background.

UPDATED

Cutaway of complete TP400 engine installation (2002)

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AERO-ENGINES - TURBOFAN, INTERNATIONAL

Date Posted: 18 September 2001

Jane's Aero-Engines 11

EUROJET TURBO GmbH

EUROJET EJ200

The EJ200 is an advanced turbofan designed for Mach numbers of about 2. It is fully modular, and allows for on-condition maintenance with built-in engine health monitoring and test equipment. Low maintenance and life-cycle cost, along with high reliability, have been prime design criteria. The EJ200 engine development programme was structured in four phases:

Phase 1, Technology Acquisition, which started in 1985.

Phase 2, Design Verification, first Design Verification Engine (DVE) run in November 1988.

Phase 3, Full Scale Development (FSD), first prototype engines delivered to the customer in 1994 for flight testing. Development Aircraft (DA3) first flight with EJ200 engines 4 June 1995 (DA1 and DA2 had previously flown with RB199 engines).

Phase 4, Flight Evaluation and Full Certification: By March 1996, over 40 flights had been performed by the EF2000 Development Aircraft (DA3) with the first set of prototype engines satisfactorily completing approximately 120 engine running hours. To March 1996, DA3 had completed a flight envelope of up to 11,000 m (36,000 ft) from 296 km/h (184 mph, 160 kt) up to M1.2.

The technology used within the design and development of the engine, and the extensive testing and its integration in the basic design concept, has led to an engine free from any serious mechanical or engine dynamics problem. The EJ200 is designed for an overhaul life of 6,000 hours or 25 years, with a maintenance interval greater than 400 hours. In the Eurofighter an engine can be changed by four men in 45 minutes.

By April 1996, the EJ200 FSD engine had run more than 5,200 hours, of which over 1,600 had been in altitude test centres. Under the Flight Development Programme, about 400 engine running hours had been accomplished. The first phase of flight testing was completed on 18 April 1996, with outstanding results.

In April 1998, it was announced that production had been authorised, following the signature of `umbrella' contracts for production investment, production and integrated logistic support for 1,500 engines for 620 aircraft. At that time, 21 development engines had logged 2,500 hours of altitude testing and 1,700 flight hours in all seven Eurofighter prototypes, including the re-engined DA1 and DA2. Among other technical milestones, these aircraft have demonstrated dry supercruise capability. In September 1998, contracts were signed for the first tranche of 148 aircraft, at a fixed cost of DM14 billion or US$8.2 billion. Subsequent tranches for the four partner nations are expected to be for 236 aircraft each. The first production engine contract, signed at the same time, covers 363 engines.

EJ200 Mk 101

This is the initial production version, for at least the first tranche of Eurofighter Typhoon aircraft. Component manufacture began later in 1998, engine certification for development purposes was achieved in November 1999, and deliveries of production engines against the initial contract for 363 engines began in late 2000, with 10 engines due for delivery by the end of that year. By mid-2000 running time exceeded 15,000 hours on 28 engines and flight time exceeded 3,000 hours. Engine certification was achieved in that year, and on 3 January 2001 Rolls-Royce announced the completion of assembly and testing of the first production engine.

Following the completion of the bench and flight certification programme on the Mk 101 engine, Eurojet was awarded the engine's Technical Certificate from Netma, the NATO EF2000/Tornado Management Agency. By summer 2001 each of the Eurojet partners had conducted acceptance tests on production engines completed on their own assembly lines, preparatory to delivery to the Eurofighter partners. The first delivery took place from Rolls-Royce's Filton (Bristol) factory on 12 July 2001.

The partners are now delivering what is currently planned to be a further 1,381 EJ200 engines for the four original airframe partners. It is expected that this total will be at least doubled by Typhoon export orders. According to published information, all engines for the first two tranches of aircraft will be to the same standard, without even the vectoring nozzle (see later). Tranche 3, the first aircraft of which are to enter service in 2010, will be the first to incorporate any significant upgrades.

Growth strategy involves several paths. Normal development to increase thrust begins with the Growth 1 level, for which the engine designation **EJ230** has been published. This would introduce a new LP compressor with 10 per cent greater airflow and a pressure ratio of 4.6. This engine would offer thrust increases of up to 20 per cent dry and 15 per cent reheat (augmented). For the period after 2005, Growth 2 would increase flow through both the fan and core and a second increase in pressure ratio. Demonstrator programmes would be expected to confirm 30 per cent increase in both dry and augmented thrust over the initial production EJ200.

Marking a sharp and welcome about-face over Eurofighter's original stance, the Spanish partner ITP has made impressive progress in the development of a totally new vectoring nozzle. First tested on an EJ200 engine in July 1998, this uses three rings (inner, intermediate and outer) positioned by a single hydraulic system to control throat area, exit area and vector angle. Maximum mechanical deflection (in any direction) is 20º, giving a maximum jet (fluid-dynamic) deflection of 23.5º. Maximum slew rate is 110º/s. Major components are titanium, the petals are nickel alloy (ceramics are being studied) and friction elements are anti-abrasion steels and nickel alloys.

Since early 1999, the Rockwell/DASA X-31 Vector has been considered as an almost tailormade

flight-test aircraft. Today Rockwell is Boeing and DASA is EADS, who are jointly with the US Navy restoring the X-31 to flight status after a six-year lay-up. There is no funding for testing with a vectored EJ200, but in 2001 the Spanish Ministry of Defence was discussing the possible use of the X-31 for either vectored EJ200 testing or for testing the ITP nozzle on the existing F404 engine, under the US Foreign Military Sales programme.

A special advanced version of EJ200, probably with the ITP vectoring nozzle, is being studied for single-engined applications. The immediate target aircraft is the planned Saab JAS39-Plus Gripen, in which BAE Systems is a participant. A second version of EJ200 is being considered for naval applications, with the prospect of launch of a carrier-based version of Eurofighter, a study of which has been funded.

The following describes the current production engine.

Type

Two-shaft turbofan.

LP Compressor

No inlet guide vanes. Three stages all blisk, with 3-D transonic blades of robust large-chord section. Overhung ahead of high-capacity ball bearing and forward roller bearing. Mass flow 75 to 77` kg (165 to 170 lb)/s. Bypass ratio about 0.4. Pressure ratio over 4.0.

HP Compressor

Five stages, with first-stage variable inlet guide vanes and blisk rotors with wide-chord 3-D aerofoils for high surge margin. Shaft supported in front ball and rear roller bearings. OPR 26.0.

Combustion Chamber

Fully annular, with 20 airspray fuel nozzles for minimum smoke and emissions. Thermal barrier coating.

HP Turbine

Single stage with 3-D single-crystal blades. DS stators with thermal barrier coatings on inner and outer platforms. Both turbine bearings in single interstage support frame with brush seals throughout.

LP Turbine

Single stage with 3-D single-crystal blades. Both rotors have tip-clearance control.

Jetpipe

Integrally stiffened metal bypass duct made in bolted sections. Bypass and core flows enter afterburner.

Afterburner

High-efficiency burn-then-mix type. Low-drag fuel injectors (15 × 4) in cold stream, 15 primary vaporiser burners and 15 long radial hot-stream injectors with air-cooled manifolds. Air-cooled screech-damping heat shield.

Nozzle

Fully variable convergent/divergent with 24 master/slave petals driven by cam/roller system. Exhaust

area optimised for mission performance.

The accompanying ITP computer images show the principle of the new vectoring nozzle, which by June 1999 had logged over 80 hours' testing, of which 15 were in full reheat (augmented). This nozzle uses balance-beam principles to use part of the energy in the jet to manipulate the nozzle, reducing the work required from the actuators, of which there are four in all. The nozzle has been deflected in any direction at 110º/s to a maximum angle of 23.5º, generating a side force of 20 kN (4,486 lb). Studies have investigated deflections up to 35º. Installed in the Eurofighter, the improved boat-tail shape is calculated to improve takeoff thrust by 2 per cent and supersonic-cruise thrust (10,973 m, 36,000 ft; M1.2 max dry) by 7 per cent. Other improvements include a reduction of 25 per cent in T-O run, reduction of 3 per cent in both cruise drag and mission fuel and increase of 7 per cent in sustained twin rate. Other advantages are discussed in the introductory feature Military Engines.

Accessories

Central gearbox on underside, driven via tower shaft in interstage support. Rotating oil tank gives positive artificial gravity during manoeuvres. Minimum weight dressing with maximum accessibility.

Control System

FADEC, with integrated health-monitoring. In March 1999, Lucas Aerospace was awarded a £300 million contract for `a broad range of EJ200 controls'.

Dimensions

Length overall	3,988 mm (157 in)
Inlet diameter	740 mm (29.0 in)

Weight, Dry

	990-1,034 kg (2,180-2,280 lb)

Performance Ratings

Uninstalled, S/L, ISA:	90 kN (20,000 lb st) class
Max dry	60 kN (13,500 lb st) class

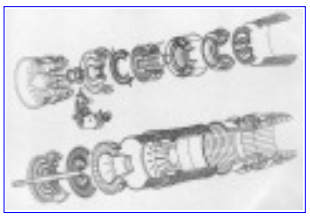
Specific Fuel Consumption

Max, as above	47-49 mg/Ns (1.66-1.73 lb/h/lb st)
Max dry, as above	21-23 mg/Ns (0.74-0.81 lb/h/lb st)

Contract Price

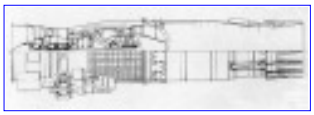
Initial requirement for the four partners is 1,500 engines, approximately US$8 billion.

UPDATED



Exploded drawing of EJ200

EJ200 engine at RRMAE, Bristol

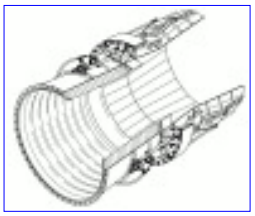


Longitudinal section through EJ200

EJ200s in assembly at Munich (1998)

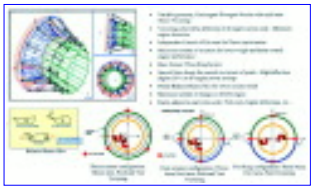
Production EJ200 at FiatAvio, Caselle (2000)

Cutaway drawing of EJ200



Nozzle of the current production EJ200 (1999)

Computer image of the ITP vectoring nozzle (1999)



ITP computer images of the vectoring nozzle (2000)

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AERO-ENGINES - TURBOPROP, INTERNATIONAL

Date Posted: 18 September 2001

Jane's Aero-Engines 11

TPI TURBOPROP INTERNATIONAL GmbH

TPI M138

TPI TURBOPROP INTERNATIONAL GmbH

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On 15 February 2000, the above companies announced the formation of Turboprop International, a company specifically created to design, develop, produce and support the M138 engine, which these partners hope will be selected to power the A400M. Percentage shareholdings are given below. Contact details are given above, even though during the development phase the Turboprop International management team has been based alongside AMC in Toulouse. The CEOs of the four partners hailed this development as `a significant step for consolidation of the European aero-engine industry.'

The M138 was still only a candidate for the propulsion of the A400M. Said to be `the most powerful turboprop in the western world', it was based on the Snecma M88 fighter engine. Other inputs were to be provided by the Eurojet EJ200 and the MTU/FiatAvio ADP (Advanced Ducted Propulsor, a demonstrator programme in which MTU originally worked with Pratt & Whitney).

The workshares were originally allocated as follows:

FiatAvio: 22 per cent, with responsibility for the main and accessory gearboxes and a share in the LP turbine.

ITP: 12 per cent, with responsibility for the inlet casing and dressing, intermediate casing and turbine exhaust casing.

MTU: 33 per cent, with responsibility for the LP compressor and the main share in the LP turbine.

Snecma: 33 per cent, with responsibility for the core engine (HP compressor, combustor and HP turbine).

TA of Belgium (which see) has been admitted as an `associate partner', and TPI `is prepared to incorporate additional partners from the A400M participating nations.'

The partners emphasise that, with ``minor modifications'' the M138 would have a power growth potential of at least 30 per cent (one brochure states 40 per cent). A particular feature would be the ability to generate negative thrust in flight without having to vary propeller pitch (for example, to make possible steep approaches).

The partners stated ``While the M138 is a defence programme, it will be launched in accordance with commercial guidelines. A joint engine management company will be floated in time to submit the official bid to the aircraft manufacturer."

The following description relates to the M138, as planned before the agreement with Rolls-Royce:

Intake

Annular, with integral particle separator. Aluminium alloy, with thermal ice protection.

LP Compressor

Four stages, with first three stator stages variable. Incorporates EJ200 technology.

HP Compressor

Six stages, with first three stator stages variable. Derived from M88-2.

Combustion Chamber

Annular, ceramic coated, with 16 airblast fuel nozzles. Derived from M88-2.

HP Turbine

Single stage, with air-cooled single-crystal blades. Derived from M88-2.

LP Turbine

Three stages, 3-D aerodynamics, based on technology from EJ200 and ADP.

Output

Compound epicyclic propeller gearbox driven off front of LP compressor, with drive to single-rotation propeller. Large-diameter ball and roller bearings with provision for resolving propeller loads, and incorporating torquemeter and brake.

Accessories

Hydraulic pumps, generator and other items mounted on accessory gearbox driven by bevel tower shaft off front of HP compressor.

Propeller

Eight composite blades, diameter 5.18 m (17 ft). Propulsion system optimised for cruise Mach number 0.68 to 0.72.

Control System

Dual-channel FADEC for gas generator, propeller and maintenance, including health monitoring.

Performance Rating

(S/L) 1999 estimate

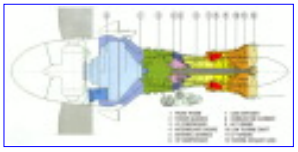
T-O	8,000 kW (10,730 shp)
Nominal	6,000 kW (8,046 shp)
2000 estimate:	
T-O up to	10,440 kW (14,000 shp)

Specific Fuel Consumption

(9,449 m; 31,000 ft at M0.68)	88-100 µg/J (0.52-0.59 lb/h/shp) depending on cruise power

UPDATED

Longitudinal section of M138 (1998)



Longitudinal section of M138 (2001)

Full-scale mockup of M138 at MTU (1998)

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AERO-ENGINES - TURBOPROP, INTERNATIONAL

Date Posted: 18 September 2001

Jane's Aero-Engines 11

ROLLS-ROYCE DEUTSCHLAND GmbH

ROLLS-ROYCE BR700-TP

ROLLS-ROYCE DEUTSCHLAND GmbH

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The German company expressed confidence it would win the propulsion contract from AMC for the A400M. It brought a detailed scale model engine to the 1997 and 1999 Paris Air Shows, where a spokesperson said, "Alone of the contenders, we are committing significant sums, and many man-hours, to ensuring that we have an engine available within the timescale of the aircraft programme. The BR700-TP will of course be a modular engine, with a core derived from that of the BR710, plus a specially designed LP turbine driving a new propeller gearbox".

The predecessor company BMW Rolls-Royce made its first BR700-TP proposal to AMC in December 1998. It made a second proposal on schedule on 31 May 1999. This second document was a Memorandum of Understanding which outlined the technical specification (making reductions in engine weight and fuel flow, as required by the potential customer), as well as the business plan with integrated logistics support. The second offer was based on a European consortium in which the final selection of partners and suppliers was still to be defined. In announcing the MoU the design power was increased from 7,457 kW (10,000 shp) to the value given below.

The BR700-TP was planned to run 17 months after selection, during a 42-month engine development phase.

Integrated logistic support by RR Military Aircraft Engines. In August 2000, this was overtaken by the agreement of Rolls-Royce to collaborate with the five TPI companies in order to produce a joint design, as outlined in the AMC introduction. The following features were planned for the BR700-TP, now to be merged with the M88 core in a three-shaft engine:

Intake

1999: Circular, surrounding rear of spinner. Front of cowl incorporates oil radiator and shaft-driven accessories. 2000: changed to bifurcated, one inlet duct on each side. Differential inlet geometry to ensure that foreign matter does not enter the compressor.

Compressor

Ten stages, with first five stators variable. No core booster.

Combustion Chamber

Annular, low emissions, similar to BR710.

HP Turbine

Two stages, air-cooled, similar to BR710.

LP Turbine

Three stages, 3-D aerodynamics, constant rotational speed.

Output

In-line compound epicyclic gearbox with ratio of about 8. Assigned to APT (Aerospace Power Transmission, formed by ZF Luftfahrttechnik and Hispano-Suiza), in co-operation with Trud design bureau of Samara. Eight-blade reverse-pitch propeller with diameter of about 4.88 m (16 ft).

Control System

FADEC control of propulsion system, with rapid thrust change by modulating propeller pitch.

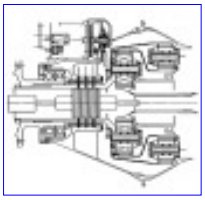
Performance Rating

(T-O, S/L)

Design power	about 8,948 kW (12,000 shp)

UPDATED

Longitudinal section through BR700-TP as schemed in 1999 (1999)



Provisional drawing of BR700-TP gearbox

The revised inlet arrangement proposed for the BR700-TP in 2000 (2001)

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 18 September 2001

Jane's Aero-Engines 11

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In 1979, the air forces of France, Germany and the UK began to consider the need to replace the C-130, C.160 and other airlifters by a purpose-designed aircraft with a larger fuselage cross-section. By 1982, a firm programme was being agreed, called FIMA. Over the subsequent 15 years this became Euroflag and then FLA (Future Large Aircraft), with turbofan engines.

In a controversial decision, it was agreed in May 1994 that the engines should be turboprops. It then occurred to the

project manager, that, as engines traditionally take much longer to develop than aircraft, work ought to begin to try to develop a suitable engine. There is no shortage of suitable cores, but there is little experience outside Russia and Ukraine with reduction gearboxes of the required power (6,000 to 10,440 kW, 8,000 to 14,000 shp). There is only one known facility in Western Europe that is capable of developing such a gearbox, a test rig commissioned by Rolls-Royce with capacity up to 11,200 kW (15,000 shp) and provision for driving pusher or tractor contrarotating variable-pitch propellers. In 1998, the management was transferred from Euroflag in Rome to Airbus at Toulouse, and the FLA became the A400M.

By September 1996, several partners had completed preliminary studies of the gearbox and had begun engineering design. Likewise Dowty Rotol, Ratier and Hamilton Standard (now Hamilton Sundstrand) were all well advanced with the design of the propeller. The design power was 7,087 kW (9,500 shp), but this has predictably tended to rise. A particular requirement is rapid and powerful pitch change for very steep landings 'twice as steep as the C-130J limit' with immediate reverse pitch on touchdown, followed by a fine positive pitch suitable for taxiing up what might be a steep slope.

By August 1999, contenders by AlliedSignal [Honeywell] and Rolls-Royce [Allison] had been eliminated, and the only proposals announced as a possible A400M power plant were a Twin-Pac proposal by Pratt & Whitney Canada based on paired PW150 power sections, the Snecma M138 (described later under TPI) and BMW Rolls-Royce BR700TP (described later under Rolls-Royce Deutschland). A decision was expected in July 1999, but AMC postponed this, supposedly over workshare arguments and proposals for a `joint solution'. This is unfortunate, because (unless the D-27 were chosen) the engine is likely to determine the timescale of the entire programme. Nothing has been said regarding discussions with Progress (Ukraine), whose D-27 - a complete propulsion system already in existence - has been excluded on what appear to have been political grounds.

The widely used letters AMC have variously been said to stand for Airbus Military Company and Airbus Military Corporation. In fact the company was registered in December 1998 as Airbus Military SAS, denoting *Société aux Actions Simplifiées.*

Over the years the number of participating governments, and their intended A400M purchase, has naturally varied. The following table sums up the position in mid-2001:

Country	**Original**	**1999**	**2000**	**2001**	**First delivery**
Belgium	12	7	7	7	2014
France	50	50	50	50	2008
Germany	75	75	73	73	2008
Italy	44	10	16	16*	2014
Luxembourg	0	0	1	1	2014
Portugal	0	0	4	3*	-
Spain	36	27	27	27	2010
Turkey	26	26	26	10	2008
UK	45	25	25	25	2008
Total	**288**	**220**	**229**	**193/212***	

* Failed to sign for domestic political reasons.

In fact, while drastically cutting its buy, the UK was the first nation to commit to an order, in July 2000. It then stated that it wished Rolls-Royce to be involved in the propulsion, a diplomatic way of saying it rejected the competing M138 engine. Snecma Chairman J-P Bechat said "I understand that. So we should find a way to have a common project." Over the

previous year the `winner takes all' attitude had, at least at the political level, given way to a search for a common merged solution, largely by trying to mate the previously unrelated modules. By mid-2000 this was well on the way to completion, to meet a planned EIS of 2007. Predictably, in the year since then, EIS has slipped to 2008, as noted in the table above.

It had been confidently tipped, not least by programme leaders, that the whole A400M/TP400 programme would at last be launched at the Paris airshow in June 2001. In fact, the only document signed (on 19 June) was a Memorandum of Understanding by six of the nations to commit to 193 aircraft. Italy was distracted by parliamentary elections and Portugal by a funding review, while of those that did sign Germany was threatened by budgetary constraints and Turkey cut its buy by over 60 per cent.

By August 2001 officials in the partner nations were trying to come up with some kind of delivery schedule, unit price and funding arrangement. The 22 years of talk have helped nobody except the makers of rival products, such as the C-130J and C-17.

UPDATED

Artist's impression of A400M with chin inlets to engines (1994)

A 1995 model of A400M. The slightly swept wing dated from when the aircraft was to have had turbofan engines

Artist's impression of the turboprop-powered Airbus A400M Future Large Aircraft (2002)

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AERO-ENGINES - TURBOFAN, INTERNATIONAL

Date Posted: 04 July 2001

Jane's Aero-Engines 10

CFM INTERNATIONAL SA

CFM INTERNATIONAL CFM56

US military designation: F108

In June 1971 Snecma was looking for a powerful partner to help it develop the M56, a commercial turbofan in the 10-tonne (98.1 kN, 22,046 lb) class. The partner proved to be GE, which was enthusiastic. Unfortunately its contribution was the core of the F101 (which see), and this caused prolonged difficulties with the US State Department, Treasury and CIA. After prolonged effort by the two partners the CFM56 (Commercial Fan, M56) was allowed to go ahead, provided Snecma did not see inside the supposedly secret core, and paid a royalty of US$20,000 per engine. A further clause prohibited ``tariffs against US aircraft imports into the European Community". With these and other stipulations the deal was allowed by President Nixon on 30 May 1973.

The first CFM56 demonstrator ran at GE Evendale on 20 June 1974. At that time the US view was still that the collaborative engine was ``a giveaway of US technology... ruinous to US aerospace trade and destructive of our jet engine industry." Nothing could have been further from the truth.

The world of aerospace is full of irony. The HP spool, which caused all the trouble, is to be replaced in future CFM56 versions by a completely new design. Moreover, in the first eight years hardly any engines were sold, and in 1979, just before the first CFM56 version was certificated, the decision had been taken to cancel the whole programme because of the absence of customers. By a matter of days it was allowed to survive, and today the CFM56 is by far the best-selling civil aircraft jet engine.

By November 2000 firm sales exceeded 14,200 and options took the overall total to 18,637, with a sales value exceeding US$71 billion. The 10,000th engine was handed over at a ceremony on 13 June 1999. At that time the CFM56 was in service in five main models (treating the Dash-5C as different from earlier Dash-5s), covering thrusts from 82.55 to 151.78 kN (18,500 to 34,000 lb st). In addition, the CFM56-9 was in advanced development and major new variants were planned. By February 2001 the original five versions had found 333 customers in 104 countries, and 11,349 engines had logged over 161 million hours in over 100 million cycles in 5,205 aircraft. Overall, dispatch reliability was 99.97 per cent, IFSD rate 0.003 and SVR 0.076.

Snecma is responsible for the fan, LP booster, LP turbine and accessory gearbox, and provides installation design. GE provides the HP core, main fuel control and system design integration. Final assembly is performed at Evendale (Cincinnati) and - in a reversal of an original US government decree - by Snecma at Villaroche.

The top-selling version is the Dash-3, but this has now been overtaken in technology by a later variant, the Dash-7; this is the exclusive engine on future 737 models, of which 1,010 were ordered in the first four years.

Reflecting the general upturn in the airline market, orders for all CFM56 versions totalled 334 in 1994, 550 in 1995, 792 in 1996, 1,340 in 1997, 1,389 in 1998, 986 in 1999 and 1,184 in 2000.

The following are current versions:

CFM56-2C

Certificated 8 November 1979, under FAR Pt 33 and JAR-E, at 106.80 kN (24,000 lb st); a 97.90 kN (22,000 lb) T-O rating to 30ºC (86ºF) is used to re-engine the DC-8-60 to Super 70 Standard. The first customer was Delta, in 1979, and scheduled operations began on 24 April 1982. In April 1998, the high-time engine had flown 40,100 hours. Total flight time was 12.2 million hours. Later figures are tabulated. By 1989 a total of 110 DC-8-70 aircraft were operating, and in 2001 the total was still 110. The first CFM56 to be delivered was on a DC-8 of United, delivered in March 1982. In 2001 this aircraft, now with LanChile, had completed 48,350 hours and 15,707 cycles since re-engining.

CFM56-2B

Certificated 25 June 1982 at 97.90 kN (22,000 lb st), flat rated to 32ºC (90ºF), the CFM56-2B1 was selected by the US Air Force for its KC-135A tanker re-engining programme on 22 January 1980. First flight of a KC-135R took place on 4 August 1982 and production F108-CF-100 engines power KC-135R aircraft delivered from late 1983. The CFM56-2B1 also powers the C-135CFR tankers of the French Air Force. This engine was also selected by the Turkish Air Force in 1995 to power its KC-135K tanker.

CFM56-2A

Certificated 6 June 1985 at 106.80 kN (24,000 lb st), flat rated to 35ºC (95ºF), the CFM56-2A2 and -2A3 powers the US Navy E-6 communications aircraft, the Royal Saudi Air Force E-3 and the KE-3 tanker, and the E-3D for the Royal Air Force (UK) and the French Air Force. These applications require a long-duration oil tank capacity, reverser and gearbox to accommodate two high-capacity integrated drive generators. In March 1998, two engines were continuing on wing on an E-6 after 10,000 hours without shop visit. Four more passed that figure in June 1998. In March 2001 the RAF celebrated 10 years of CFM power, noting that in 2,500 missions over Bosnia, only two suffered engine-caused cancellation.

By April 1998, a total of 486 aircraft was in service powered by the Dash-2B and Dash-2A combined. A year later they had flown 6.9 million hours, the high-time engine having logged 10,012 hours in 7,910 cycles. Engine-caused SVR was 0.06. For later figures, see table.

By November 2000 a total of 2,781 engines of the overall CFM56-2 family had been sold. Of these, 2,593 had been delivered.

CFM56-3B1

Derivative of CFM56-2, rated at 89.00 kN (20,000 lb st), flat rated to 30ºC (86ºF), with smaller fan. This engine powers the Boeing 737-300. It first ran in March 1982. US and French certification was granted on 12 January 1984 and it entered airline service in December 1984. Rerated at 82.55 kN (18,500 lb st), it now powers the 737-500, which entered service in February 1990.

CFM56-3B2

Certificated at 97.90 kN (22,000 lb st), flat rated to 30ºC (86ºF), on 22 June 1984. For 737-300 and 737-400 with improved payload/range from short, hot, high

airfields. The 737-400 entered service in September 1988. In September 1997, an engine of Germania set an industry record by logging 30,000 hours (over 11,000 cycles) without removal.

CFM56-3C1

Rated at 104.50 kN (23,500 lb st) for 737-400. Certificated December 1986. Currently offered as common engine for all 737 models at 82.55 to 104.50 kN (18,500 to 23,500 lb st). All Dash-3 models ETOPS qualified.

Boeing produced 1,987 aircraft of the 737 Classic (-300/-400/-500) family, to support which CFM initially produced 4,433 CFM56-3 engines. The last was delivered on 17 December 1999, for a 737-400 for Czech Air Lines. At that time the CFM56-3 flight time exceeded 88 million flight hours. One engine was still on-wing at over 31,700 hours, and the fleet average exceeded 14,000 hours before initial shop visit. IFSD rate was 0.003. Engine-caused IFSD rate was 0.002. Engine-caused shop visit rate (12-month rolling average) was 0.080, and dispatch reliability (now often called D&C, for delays and cancellations) 99.98 per cent. The high-time engine had logged 43,509 hours in 42,468 cycles. For later figures, see table. The only significant problems have been seven inflight failures of the HP turbine rotor rear shaft seal since 1995 and three failures of the accessory-gearbox starter drive shaft (similar to those of the CFM56-7). All affected shafts had been replaced by early 1999.

On 5 September 1999 the CFM56-3B1 engine of a 737-300 of Continental apparently suffered a major core failure about one minute after take-off on a scheduled flight. Debris damaged the aircraft's fin, and other parts (including a large portion of reverser cowl) fell to the ground. Attention focused on the No 7 HP compressor stage.

By November 2000 the total number of orders for CFM56-3 engines had reached 4,482, of which 4,468 had been delivered. Total flight time had then exceeded 100 million hours. A number of engines had flown more than 30,000 hours without a shop visit, the high-time engine being on a Hapag Lloyd aircraft leased to Malev, which had exceeded 33,600 hours with a comfortable EGT margin.

To reduce cost of ownership and prolong on-wing life, by 2001 work had almost been completed on a core improvement programme. The key features are to reblade the HP compressor using the 3-D technology of the -5B/P and -7, and to reblade the HP turbine rotor with blades of different material. By March 2001 all testing of the improved CFM56-3 engine had been completed, except for icing. Among other improvements, EGT is reduced by -15ºC, and sfc is reduced by at least 1 per cent. In May 2001 Southwest Airlines signed a US$300 million agreement to purchase 300 core-upgrade kits. It also signed a five-year US$1.2 billion extension to its MCPH (maintenance cost per hour) agreement with GE Engine Services.

CFM56-5A1

Launched September 1984 for A320. Has the fan diameter of the CFM56-2, with improved aerodynamics in all LP and HP components, advanced clearance control features and FADEC. Nominal rating is 111.30 kN (25,000 lb st), flat rated to 30ºC (86ºF). Certificated 27 August 1987 and entered service April 1988. Outstanding reliability resulted in 120-minute ETOPS certification. By April 1999 a total of 500 A320 and A319 aircraft had flown over 13.62 million engine hours, with the high-time engine having accumulated 27,651 hours in 21,243 cycles. Engine-caused SVR was 0.086, and dispatch reliability 99.95 per cent. For later figures, see table.

CFM56-5A3

Rated at 117.90 kN (26,500 lb st). This engine is expected to meet the specific airline requirements for A320 growth aircraft. Certificated February 1990.

CFM56-5A4, -5A5

Rated respectively at 97.90 and 104.50 kN (20,000 and 23,500 lb st), these two versions power the A319. Certificated.

CFM56-5B1

New -5C4 core. Rated at 133.50 kN (30,000 lb st). Certificated in February 1994 for A321.

CFM56-5B2

High-performance derivative with core of -5C4. Rated at 137.90 kN (31,000 lb st) for A321. First run (FETT) 25 October 1991 at Villaroche, France. Certificated in May 1993, entry into service took place in March 1994. To reduce NO_x emissions by more than 45 per cent compared with a -5B with a Single Annular Combustor (SAC), CFMI offers an optional Double Annular Combustor (DAC) on CFM56-5Bs. Jointly certified in September 1994 by the US FAA and the French DGAC, the first -5BDAC installed on an A321 entered service in January 1995 with Swissair. The turbine rear frames of DAC engines suffered cracking, and a redesigned frame entered service from July 1997. By April 1999, the CFM56-5B family, in service on all members of the Airbus single-aisle family, had logged 1,500,000 flight hours since April 1994. Excluding retrofit programmes, the engine-caused SVR was then 0.025, IFSD 0.001 and dispatch reliability rate 99.96 per cent. CFMI claims ``The 5B has a lower SVR than the competition and the cost of each visit is nearly US$300,000 less". CFMI President Laviec says ``We can never be quite as efficient as the competitor, because the V2500 has a two-stage HP turbine; but we are shorter and lighter and have fewer hot-section parts, so claim 35 per cent lower maintenance costs. We also claim better performance retention because of anti-erosion coating on the HP spool, so after the first year or two we are as good on fuel burn".

CFM56-5B3

Proposed to airlines at 146.80 kN (33,000 lb st) to power longer-ranged A321 at 89 tons MTOW, which first flew April 1996 for EIS second quarter 1997. CFM56-5B3/P certificated on A321-200 at 142.35 kN (32,000 lb st) in March 1997.

CFM56-5B4

Version of -5B derated at 120.10 kN (27,000 lb st), flat rated to 45ºC (113ºF); available to power A320.

CFM56-5B5, -5B6

Alternatives for A319, respectively 97.9 kN (22,000 lb st) and 104.5 kN (23,500 lb st).

CFM56-5B/P

New 3-D technology has been used to redesign the blading of the HP compressor, HP turbine and LP turbine, enabling cycle temperatures to be reduced for any given thrust, thus reducing sfc by 3 per cent and extending on-wing life and reducing maintenance costs. The B/P core is now standard for all Dash-5 versions and has also served as the basis for the Dash-7 and Dash-9 families. Flight tested on A320 January 1996, jointly certificated in spring 1996 by FAA and French DGAC at six ratings from 97.9 to 142.0 kN (22,000 to 32,000 lb st). Entered service (with optional double-annular combustor) with Swissair A319 July 1996. The 142.0 kN (32,000 lb st) CFM56-5B3/P for the long-range A321-200 was certificated in March 1997. Dispatch reliability (March 1999) 99.95.

Following pressure from Air France, a version of the CFM56-5B was formally adopted in August 1999 as an alternative engine on the A318. Details of the engine chosen and its rating are awaited.

By 1 January 2000 CFM56-5 engines had been chosen to power more than 1,250 of the 2,300 single-aisle Airbus aircraft ordered by that time. Engines in service had logged 17 million flight hours in 11 million cycles, with a dispatch reliability rate of 99.96 per cent and SVR of 0.054. CFMI states that CFM56-5 engines average 14,000 hours on-wing prior to initial shop visit, and more than 10,000 hours after overhaul.

By November 2000 total orders for CFM56-5A/-5B engines had reached 4,193, of which 1,864 had been delivered. The high-time engine, on an A320 operated by Premiair, had reached 30,900 hours.

By February 2000 the IFSD rate for the CFM56-5A was standing at an outstanding 0.002 per 1,000 hours, while SVR was 0.069, and D&C rate was 99.95. Corresponding figures for the CFM56-5B were 0.003, 0.041 and 99.97. Data for March 2001 are tabulated.

In 2001 both CFM partners were testing a further package of CFM56-5A/-5B improvements. From the second quarter of 2001 an improved HPT rotor blade

has been available. It is made in significantly less costly material, yet, partly by virtue of redistributed cooling-air holes, will extend life. Another upgrade is the chevron nozzle, probably to have 18 trailing-edge `sawteeth', which on test is showing noise reductions of up to 11 EPNdB over Stage 3. The package is completed by nacelle and reverser modifications. A full engine test is due in November 2001, with the engine becoming available in the first quarter of 2003.

CFM56-5C2

Advanced fan, new four-stage LP compressor, active clearance control HP spool, upgraded turbine section (five-stage LP, new frame, modulated clearance, new aerodynamics throughout), integrated exhaust nozzle mixer and FADEC. Rated at 138.80 kN (31,200 lb st). Powers A340, certificated 31 December 1991, entry into service February 1993. By April 1999 the 954 Dash-5C engines in service had flown 7.72 million hours. The engine-caused SVR was then 0.085, IFSD 0.007 and dispatch reliability 99.85 per cent.

CFM56-5C3

Growth version rated at 144.60 kN (32,500 lb st). Certificated 31 December 1991 at 950ºC TET. Entered service early 1994. In 1999 it was planned to test a fan with wide forward-swept blades of hollow titanium alloy (see Tech 56).

CFM56-5C4

Growth version certificated October 1994 at 151.25 kN (34,000 lb st) for A340; entered into service April 1995 with Kuwait Airlines. In February 2000 CFMI signed an MoU with the Russian planemaker Ilyushin and TAPO (Tashkent production factory) to develop an export version of the Il-76MF with the original PS-90A engines replaced by the CFM56-5C4.

By November 2000 total orders for CFM56-5C engines had reached 1,562, of which 871 had been delivered. Total flight time was in excess of 10 million hours. In February 2001 the IFSD rate for all CFM56-5C versions was 0.009 per 1,000 hours, the SVR 0.064, and the D&C rate (dispatch reliability) adjusted to that of a twin-engined aircraft was 99.80. See table for later.

CFM56-5C/P

Despite the less-demanding nature of long-haul operations, the CFM56-5C family of engines initially did not equal the exceptional reliability of other versions (IFSD rate was well in excess of 0.01). This has been addressed by various core improvements. One modification is to use `3-D aero' rotor blading throughout the HPC. The other is to incorporate a new HPT nozzle ring and the latest type of HPT rotor blading. The 5C/P programme was launched in April 2000, and the FETT will be in November 2001. So far rig testing indicates that the complete engine will not only have longer troublefree life on wing, but will also show a 13ºC reduction in EGT, and improvement of 1.1 per cent in sfc. Certification and the start of flight testing are due in February 2002, aircraft certification in April 2003 and EIS in May 2003.

CFM56-7B

Matches core of Dash-5B/P with new snubberless wide-chord fan with solid titanium blades (see diagram), active clearance control and FADEC. Rated at 82.55-117.40 kN (18,500-26,400 lb st) for 737-600/-700/-800. Early problems included the need to strengthen the fan and its retention, redesign the fan containment, stiffen the jetpipe and rewrite the FADEC-II stall detection and recovery software. Flown on 747 testbed January 1996. Today, the Dash-7B is exceeding predictions, with sfc bettered by more than 8 per cent relative to Dash-3, and with over 50ºC EGT margin relative to equivalent rating on Dash-3C1 (expected to translate into 20 per cent longer time on-wing). Maintenance costs 15 per cent lower than Dash-3C1. Certificated by FAA and DGAC October 1996, in which month the first DAC (Double Annular Combustor) Dash-7 began testing. Flight testing of the 737-700 began in early February 1997, and this version was certificated on 7 November 1997 with EIS a month later. The 737-800 first flew in July 1997, with EIS in April 1998. The 737-600 with DAC entered service with SAS in late 1998. In November 1997 Alaska Airlines launched the CFM56-7 on the super-stretched 737-900, scheduled for certification in 2000. A variant of the CFM56-7B rated at 121.4 kN (27,300 lb st) will power the 737 AEW&C aircraft, as chosen by the Australian Defence Force. A military CFM56-7

also powers the US Navy Boeing C-40 (737 variant).

Following a number of NG737 shut-downs, the FAA issued an Airworthiness Directive in April 1998 requiring replacement of starter-shafts and drive gears and more frequent inspection of chip detectors. A further FAA AD was issued on 2 July 1998. The problem was identified as a mid-1996 decision by Hispano-Suiza to discontinue shot-peening of starter drive shafts. In September 1999 Snecma Villaroche began testing a CFM56-7 with wide-chord forward-swept blades of solid titanium (see Tech56 later).

By March 2000 the CFM56-7B had logged 2.4 million hours in over 1.2 million cycles. Dispatch reliability was 99.94 per cent, engine-caused SVR 0.013 and IFSD rate 0.015. Later figures are tabulated. The FAA granted 120-minute ETOPS in December 1998 and 180-minute in September 1999.

By February 2001 the engine of the 737 Wedgetail AEW&C aircraft, for the RAAF (see above) and predicted for other customers including South Korea and Turkey, had begun testing, a major modification being increased shaft power to the alternators to provide power for the radar installation. At that time a total of 5,619 CFM56-7B engines had been ordered or committed, of which 1,453 had been delivered. Total flight time was 1 million hours, but this will rise at a rapidly accelerating rate as engines are delivered (in 2001 CFM was delivering over 1,000 engines annually, about half being Dash-7Bs).

At that time IFSD rate on the CFM56-7B was standing at 0.003; SVR was a remarkable 0.022, while D&C (dispatch reliability) was steady at 99.94.

CFM56-9

Originally known as the CFM56-Lite, this largely new engine based on the Dash-5B/P core was to be rated at 82.55 to 102.28 kN (18,500 to 23,000 lb st) for various projected aircraft in the 100-seat class, including the IPTN N2130 and Chinese A31X. To assist selection for the latter aircraft, President Laviec has revealed that the People's Republic of China has been offered `participation in the design, development, manufacture and assembly of the LP turbine, and final assembly of the engine'. At the 1997 Paris Air Show, CFMI and AVIC (see under China) announced a joint leadership council, chaired by the presidents of the two organisations, to foster industrial co-operation and joint programmes.

Features of the CFM56-9, shown in accompanying drawings, include a fan significantly smaller than that of any other CFM56, with solid titanium blades, driven together with a two-stage core booster by an LP turbine with only three stages; other choices are a single annular combustor and FADEC-II control. Although CFMI sees a market for 1,200 user aircraft over the next 20 years, and has already completed more than 70 per cent of the design work, a go-ahead will await launch of one of the proposed aircraft. Then, said CFMI in late 1997, ``certification will be achieved within three years, making possible an EIS in 2001". In fact, by 2001 the expected market had still not yet materialised.

CFMXX

Snecma has long studied an airline engine based on the core of the M88-2 fighter engine. Originally called CFM56XX, this has now become a significantly different engine, with a 2,134 mm (84.0 in) fan and a new HP spool which makes use of the technology of the Snecma HP compressor of the GE90 to give a T-O rating of 200 kN (45,000 lb st). In 1995 Snecma obtained French government funding for this engine. The principal envisaged application, the growth versions of A340, will now initially be powered by the RR Trent 500, but this agreement is not exclusive.

Tech56

In late 1997 CFMI began long-term planning for the introduction of new technology. Called Tech56, a three-year programme is seeking information on which to base the design of a new generation of engines in the CFM56 thrust class. The following are major items in the programme:

Fan: This crucial component is the most externally obvious area in which the direct competitor is ahead. To catch up CFMI has designed a completely new fan with wide-chord `swept' blades without snubbers. The first test examples of this new fan have 25 solid blades and a diameter of 1,549 mm (61 in), to match them to the CFM56-7. Rig testing began in late 1999, and by April 2000, performance, crosswind, acoustic and blade-off tests had all been completed, the blade-off test at Evendale being called `outstanding'. In testing on a CFM56-7 the new fan is said to have demonstrated 2 per cent greater airflow at unchanged peak efficiency, with ``the promise of 5-6 per cent more thrust and 1 per cent fuel-burn improvement". By spring 2001 testing was well advanced on a 1,735 mm (so-called 68 in) fan with swept-forward hollow blades of titanium alloy. Different blade forms are being tested, with encouraging results, and by March 2001 a fan to the latest standard was running at Peebles on a modified 5C engine in performance, cross-wind, large bird, acoustics and blade-out tests. Most of these

tests should have been completed by the end of 2001.

HP compressor: GE's existing spool is one of the oldest features of the current family of engines, with a pressure ratio of 10 achieved with nine stages. GE has now designed a completely new HP spool which achieves a p.r. of 15 with only six highly loaded stages. The difficult part is achieving good stall-free operation over the entire thrust range, and GE has validated this with a test in late March 2000 followed by a second in early 2001. By March 2001 excellent results were being achieved with an advanced HPC with forward-swept rotors, stages R1 and R2 being blisks, and bowed and swept stators. Rotor blade tips are of special robust form, and the casing has received a new surface treatment. Compared with the standard HPC, the new spool has only 968 blades (compared with 1,518), operates at 29 per cent higher pr and has only three variable stators. Build 1 achieved excellent airflow and mechanical responses, and exceeded predicted stall margins. Build 2 is designed for even higher efficiency, and was to run in the second quarter of 2001.

Combustor: A further major upgrade is the TAPS (twin-annular pre-swirl) combustion chamber. One of its features is fuel nozzles (supplied by Parker Hannifin) based on microlaminate technology. Whereas GE's previous DAC (Dual Annular Combustor) reduced NO_x at the expense of worse hydrocarbons and CO, the TAPS combustor promised to cut all ICAO-mandated emissions by some 50 per cent. Following extensive sector evaluations, a complete TAPS combustor went on test in July 2000. Rig tests since 1998 have shown NO_x to be a mere 38 per cent of the previous level. Validation on an engine is due later in 2001. Among other tests, 6 tonnes of hail have been ingested, with 20 tonnes in prospect.

Turbine: One of the ways CFM56 sales have been hit by the V2500 is in the superior efficiency of the rival's two-stage HP turbine. CFMI has not disclosed whether it will bring in a second HP turbine stage; indications are that it will upgrade the existing single stage. Particular attention is being paid to HP/LP losses, and V-P Bill Clapper has said that, while existing engines have co-rotating spools, future CFM engines will have them turning in opposite directions. In military engines counter-rotation has enabled inter-turbine stators to be eliminated. Future LP turbines will have improved airfoils. By March 2001 outstanding results had been achieved with the latest design of HPT/LPT on a counter-rotating rig. HPT pressure ratio has been increased by 15 per cent, with 10 per cent fewer airfoils, half the trailing-edge shock and a 22 per cent reduction in cooling airflow. Overall, the number of LPT airfoils has been reduced by 25 per cent, whilst beating the design efficiency by 1 per cent.

Brush seals: Various forms of brush seal are being tested in an effort virtually to eliminate leakage losses, even in hot areas subject to large pressure differences. This work is being carried out on rigs by a Russian company in Samara. By early 2001 leakage had been reduced by 40 per cent, with higher values in prospect.

Nozzle: Tests on a `chevron' or `sawtooth' nozzle exit cone have shown a reduction of 3 dB in sideline noise, at the cost of a small penalty in performance.

New engines incorporating Tech56 improvements are expected to be decided upon in early 2002, for certification around 2007. The smallest version may replace the Snecma engine in the 15-tonne class which the French government agreed to fund in 1994.

CFM56 fleet summary

As of 31 March 2001, engine-caused, 12-month rolling average.

	CFM56-2A	CFM56-2B	CFM56-2C	CFM56-3	CFM56-5A	CFM56-5B	CFM56-5C	CFM56-7B
EIS	05.1986	1984	04.1982	12.1984	04.1988	03.1994	02.1993	12.1997
Engines/aircraft	193/41	1,849/441	515/109	4,474/1,973	1,032/462	923/424	886/194	1,678/781
Engine flight hours	1.23 M	7.01 M	13.65 M	103.37 M	18.66 M	4.78 M	13.82 M	6.74 M
Engine flight cycles	0.51 M	3.17 M	5.62 M	73.55 M	11.64 M	3.12 M	2.18 M	3.45 M
High-time engine hours	12,512	9,841	46,487	49,588	33,960	14,808	38,231	13,145

High-time engine cycles	4,480	4,097	18,332	47,898	24,729	14,212	6,052	8,681
SVR/1,000 hours	0.101	0.035	0.140	0.088	0.060	0.036	0.059	0.024
IFSD/1,000 hours	0.000	0.008	0.018	0.002	0.004	0.003	0.007	0.002
Dispatch reliability %	100.00	99.99	99.91	99.99	99.93	99.96	99.80 *	99.95

* Equivalent to 99.90 for two-engined a/c.

Type

Two-shaft subsonic turbofan.

Fan

Single-stage axial. Forged titanium disc holding (CFM56-2) 44 titanium blades each with a tip shroud to form a continuous ring; (CFM56-3) 38 titanium blades each with part-span shroud; (CFM56-5) 36 titanium blades each with part-span shroud; (CFM56-7 and -9) 24 solid titanium wide-chord blades. Max rpm (2) 5,280, (3) 5,490, (5A) 5,100, (5B) 5,200, (5C2, 5C3) 4,800, (5C4) 4,960, (7) 5,380, (9) 6,192.

LP Compressor

Three axial stages (four on -5B and -5C, two on -9), on titanium drum bolted to fan disc. A ring of bleed doors allows core air flow to escape into fan duct at low power settings.

HP Compressor

Nine-stage rotor with three stages of titanium blades and remainder of steel. Stator vanes steel, with first four stages variable. Max rpm (except Lite) 15,183; Lite 14,605.

Combustion Chamber

Machined ring, fully annular, with advanced film cooling. Optional Dual Annular Combustor (DAC) is offered on CFM56-5B and CFM56-7B.

HP Turbine

Single-stage with air-cooled nozzle and rotor airfoils, using advanced technology materials. HP system carried in two bearings.

LP Turbine

Four-stage (five on -5C, three on -9) with tip shrouds.

Jetpipe

Fixed-area core pipe with convergent nozzle, forced mixer on -5C. Constant-diameter fan duct of sound-absorbent construction. Outer cowl and engine cowl form convergent plug nozzle, with airframe-mounted reverser (not on the CFM56-5C).

Accessories

(CFM56-2 and -5) Gearbox in front sump transmits drive from front of HP spool to transfer gearbox on underside of fan case. Air starter at transfer gearbox (-2) or accessory gearbox (-5). (CFM56-3 and -7B) Side-mounted accessory drive gearbox with transfer gearbox; air starter pad on accessory gearbox.

Control System

Hydromechanical with electronic trim (-2, -3); FADEC (Full Authority Digital Electronic Control) on -5 and -7B families.

Oil System

Dry sump design. The lubrication module is produced by Techspace Aero of Belgium.

Dimensions

Length, excl spinner (flange to flange):	
CFM56-2	2,430 mm (95.7 in)
CFM56-3	2,360 mm (93.0 in)
CFM56-5A	2,422 mm (95.4 in)
CFM56-5B	2,601 mm (102.4 in)
CFM56-5C	2,616 mm (103.0 in)
CFM56-7B	2,507 mm (98.7 in)
CFM56-9	2,329 mm (91.7 in)
Fan diameter:	
CFM56-2, -5A, -5B	1,735 mm (68.3 in)
CFM56-3	1,524 mm (60.0 in)
CFM56-5C	1,836 mm (72.3 in)
CFM56-7B	1,549 mm (61.0 in)
CFM56-9	1,422 mm (56.0 in)

Weight, Dry

CFM56-2A2	2,187 kg (4,820 lb)
CFM56-2B1	2,119 kg (4,671 lb)
CFM56-2C series	2,102 kg (4,635 lb)
CFM56-3B1	1,940 kg (4,276 lb)
CFM56-3B2, -3C	1,951 kg (4,301 lb)
CFM56-5A1	2,337 kg (5,154 lb)
CFM56-5A3, 5A4, 5A5	2,266 kg (4,995 lb)
CFM56-5B	2,456 kg (5,413 lb)
CFM56-5C	2,644 kg (5,830 lb)
CFM56-5C (propulsion system)	3,990 kg (8,796 lb)
CFM56-7B	2,384 kg (5,257 lb)

CFM56-9	not finalised

Performance Ratings

CFM56 Variant	2A3	2B1	2C1	3B1	3B2	3C1	5A1	5A3	5A4	5A5
Certification	Jun 85	Jun 82	Nov 79	Jan 84	Jun 84	Dec 86	Aug 87	Feb 90	Feb 96	Feb 96
T-O thrust kN	106.8	97.90	106.8	89.00	97.90	104.50	111.30	117.90	97.90	104.50
lb st	24,000	22,000	24,000	20,000	22,000	23,500	25,000	26,500	22,000	23,500
mass flow kg/s	369.2	355.6	357.4	297.1	309.8	322.0	386.5	397.4	370.1	381.9
lb/s	814	784	788	655	683	710	852	876	816	842
flat-rate TºF(ºC)	90(32)	90(32)	86(30)	86(30)	86(30)	86(30)	86(30)	86(30)	113(45)	99(37.2)
BPR	5.9	6.0	6.0	5.0	4.9	5.0	6.0	6.0	6.2	6.0
Uninstalled 35 k, M0.8, ISA:										
max climb kN	25.62	24.24	24.02	21.62	23.40	24.64	24.98	24.98	24.98	24.98
lb	5,760	5,450	5,400	4,860	5,260	5,540	5,616	5,616	5,616	5,616
OPR	31.8	30.5	31.3	27.5	28.8	30.6	31.3	31.3	31.3	31.3
max cruise lb	4,980	4,969	4,980	4,650	5,040	5,370	5,000	5,000	5,000	5,000
Red line EGTºC	930	905	930	930	930	930	890	915	890	890
fan rpm	5,280	5,280	5,280	5,490	5,490	5,490	5,100	5,100	5,100	5,100
core rpm	15,183	15,183	15,183	15,183	15,183	15,183	15,183	15,183	15,183	15,183
CFM56 Variant *(contd)*	**5B1**	**5B2**	**5B3**	**5B4**	**5B5**	**5B6**	**5B7**	**5C2**	**5C3**	**5C4**
Certification	Feb 94	May 93	Jun 96	Feb 94	Mar 96	Oct 95	Jun 99	Dec 91	Mar 94	Oct 94
T-O thrust kN	133.50	137.90	148.12	120.10	99.79	104.50	120.10	138.80	144.60	151.25
lb st	30,000	31,000	33,300	27,000	22,000	23,500	27,000	31,200	32,500	34,000
mass flow kg/s	427.7	433.6	439.1	408.2	371.04	382.8	408.2	464.9	474.0	483.1
lb/s	943	956	968	900	818	844	900	1,025	1,045	1,065
flat-rate TºF(ºC)	86(30)	86(30)	86(30)	111(43.9)	113(45)	113(45)	111(43.9)	86(30)	86(30)	86(30)
BPR	5.5	5.5	5.4	5.7	6.0	5.9	5.7	6.6	6.5	6.4

Uninstalled 35 k, M0.8, ISA:										
max climb kN	28.556	28.556	28.556	25.04	25.04	25.04	28.556	33.735	33.735	34.86
lb	6,420	6,420	6,420	5,630	5,630	5,630	6,420	7,585	7,585	7,838
OPR	34.4	34.4	34.4	32.6	32.6	32.6	34.4	38.3	38.3	39.2
max cruise lb	5,840	5,840	5,840	5,020	5,020	5,020	5,840	6,910	6,910	7,410
Red line EGT ºC	950	950	950	950	950	950	950	950	965	975
fan rpm	5,200	5,200	5,200	5,200	5,200	5,200	5,200	4,800	4,800	4,985
core rpm	15,183	15,183	15,183	15,183	15,183	15,183	15,183	15,183	15,183	15,183
CFM56 Variant *(contd)*	**7B18**	**7B20**	**7B22**	**7B24**	**7B26**	**7B27**	**9B18**	**9B20**	**9B23**	
Certification	Dec 96	Dec 96	Dec 96	Dec 96	Dec 96	Dec 96	-	-	-	
T-O thrust kN	86.775	91.67	101.015	107.6	116.99	121.43	82.55	89.00	102.28	
lb st	19,500	20,600	22,700	24,200	26,300	27,300	18,500	20,000	23,000	
mass flow kg/s	307.1	315.7	330.2	341.1	353.35	359.25	268.0	281.7		
lb/s	677	696	728	752	779	792	591	621		
flat-rate TºF(ºC)	86(30)	86(30)	86(30)	86(30)	86(30)	86(30)	86(30)	86(30)		
BPR	5.5	5.4	5.3	5.3	5.1	5.1	5.17	5.08	about 5	
Uninstalled 35 k, M0.8, ISA:										
max climb kN	26.51	26.51	26.51	26.51	26.51	26.51				
lb	5,960	5,960	5,960	5,960	5,960	5,960				
OPR	32.7	32.7	32.7	32.7	32.7	32.7				
max cruise lb	5,420	5,450	5,450	5,480	5,480	5,480	4,200	4,200		
Red line EGT ºC	950	950	950	950	950	950				
fan rpm	5,380	5,380	5,380	5,380	5,380	5,380				
core rpm	15,183	15,183	15,183	15,183	15,183	15,183				

Specific Fuel Consumption

(cruise, as above)

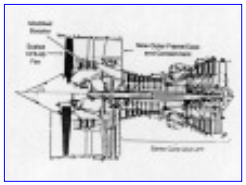
CFM56-2A2	18.72 mg/Ns (0.661 lb/h/lb)
CFM56-2B1	18.61 mg/Ns (0.657 lb/h/lb)
CFM56-2C series	18.44 mg/Ns (0.651 lb/h/lb)
CFM56-3 (all)	18.55 mg/Ns (0.655 lb/h/lb)
CFM56-5A1, -5A3	16.87 mg/Ns (0.596 lb/h/lb)
CFM56-5B	16.98 mg/Ns (0.600 lb/h/lb)
CFM56-5C (with mixer)	16.06 mg/Ns (0.567 lb/h/lb)
CFM56-7	17.06 mg/Ns (0.603 lb/h/lb)
CFM56-9	target 9% below CFM56-3

Contract Price

September 1997, CFM56-3 for 12 737s, US$85 million; CFM56-5A for 50 A319, US$500 million; CFM56-5B for 12 A319/320/321, US$150 million; October 1997, CFM56-5B/P for 120 A319/320/321, US$1.4 billion; November 1997, CFM56-7 launched on 737-900, 10 aircraft `about US$100 million'; March 1998, CFM56-7B for 25 737-800, over US$300 million; October 1998, CFM56-5C4 engines for 10 firm, 7 option, A340-300, US$350 million.

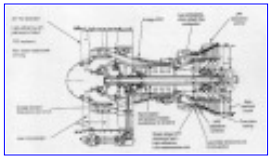
UPDATED

CFM56-2

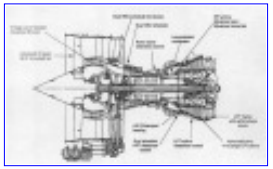


Comparative sections of the CFM56-2 (lower half) and -3 (upper)

CFM56-3

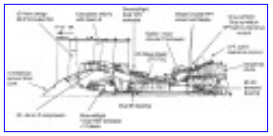


CFM56-3 design features



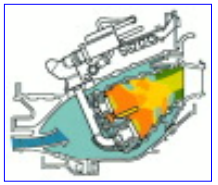
CFM56-5A design features

CFM56-5B



CFM56-5B design features

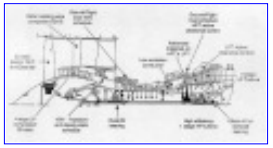
CFM56-5B double-annular combustor (1998)



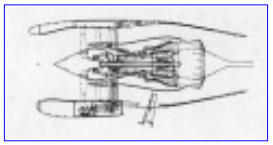
The DAC, introduced on the CFM56-5B (2000)

CFM56-5C (first engine, showing mixer nozzle)

CFM56-5C



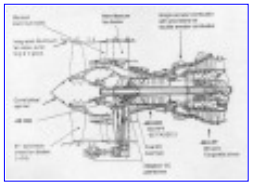
CFM56-5C design features



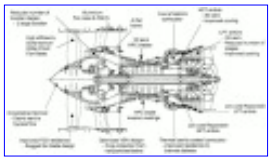
Longitudinal section of CFM56-5C complete nacelle, showing (lower half) reverser in operation

Rear view of CFM56-5C
(1998)

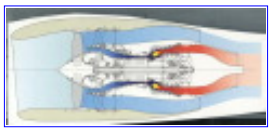
CFM56-7



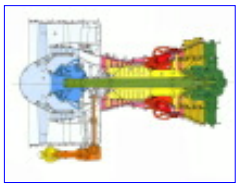
CFM56-7 design features



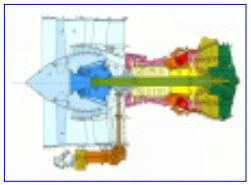
CFM56-9 design features
(2000)



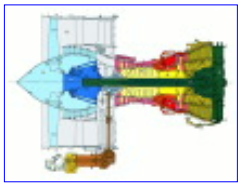
Longitudinal section of CFM56-9 nacelle



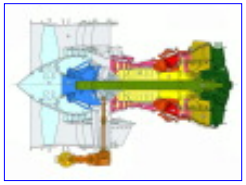
CFM56-3 cross-section; Key: Fan and booster major module: (1) fan and booster, (2) no.1 and no.2 bearing support, (3) inlet gearbox (IGB) and no.3 bearing, (4) fan frame; Core major module: (5) HPC rotor, (6) HPC forward stator, (7) HPC rear stator, (8) combustor casing, (9) combustor liner, (10) HPT nozzle, (11) HPT rotor, (12) LPT nozzle (stage 1); Low pressure turbine major module: (13) LPT, (14) LPT shaft, (15) LPT frame; Accessory drive module: (16) transfer gearbox (TGB), (17) accessory gearbox (AGB)
(2001)



CFM56-5B with DAC; Key: Fan and booster major module: (1) fan and booster, (2) no.1 and no.2 bearing support, (3) inlet gearbox (IGB) and no.3 bearing, (4) fan frame; Core major module: (5) HPC rotor, (6) HPC forward stator, (7) HPC rear stator, (8) combustor casing, (9) combustor liner, (10) HPT nozzle, (11) HPT rotor, (12) LPT nozzle (stage 1); Low pressure turbine major module: (13) LPT, (14) LPT shaft, (15) LPT frame; Accessory drive module: (16) transfer gearbox (TGB), (17) accessory gearbox (AGB)
(2001)



CFM56-5C; Key: Fan and booster major module: (1) fan and booster, (2) no.1 and no.2 bearing support, (3) inlet gearbox (IGB) and no.3 bearing, (4) fan frame; Core major module: (5) HPC rotor, (6) HPC forward stator, (7) HPC rear stator, (8) combustor casing, (9) combustor liner, (10) HPT nozzle, (11) HPT rotor, (12) LPT nozzle (stage 1); Low pressure turbine major module: (13) LPT, (14) LPT shaft, (15) LPT frame; Accessory drive module: (16) transfer gearbox (TGB), (17) accessory gearbox (AGB) (2001)



CFM56-7; Key: Fan and booster major module: (1) fan and booster, (2) no.1 and no.2 bearing support, (3) inlet gearbox (IGB) and no.3 bearing, (4) fan frame; Core major module: (5) HPC rotor, (6) HPC forward stator, (7) HPC rear stator, (8) combustor casing, (9) combustor liner, (10) HPT nozzle, (11) HPT rotor, (12) LPT nozzle (stage 1); Low pressure turbine major module: (13) LPT, (14) LPT shaft, (15) LPT frame; Accessory drive module: (16) transfer gearbox (TGB), (17) accessory gearbox (AGB) (2001)

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 04 July 2001

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Chairman and Chief Executive Officer: Pierre Fabre
Executive Vice-President: Bill Clapper
Vice-President, Marketing: H Depp

CFM International (CFMI), a joint company, was formed by Snecma of France and General Electric of the United States in 1974 to provide management for the CFM56 programme and a single customer interface.

In January 1998, it was reported that CFMI and RKBM (see in Russian section) had discussed the possibility of the Rybinsk company assembling the CFM56, and possibly manufacturing the engine under licence. In the first instance the engine might be the CFM56-3 or -7 for the 737, but other applications could be re-engined Il-76 (see CFM56-5C4), Il-86 and Tu-154 aircraft. Such a proposal has not been confirmed by CFMI.

In 1999 the CFM56, which had a discouragingly slow start, became the best-selling civil aircraft engine in history. Details are given later. In 2000-01 CFM claimed ``54 per cent of the entire market over 100 seats''. Indeed, in 2000 CEO Laviec said ``I believe CFMI's share of (narrow-body) orders will increase to 85 per cent over the next five years''.

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 01 May 2001

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ROLLS-ROYCE SNECMA

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Tel: (+44 20) 72 35 36 41
Fax: (+44 20) 72 45 63 85

Managing Director: Guillaume Giscard d'Estaing

Whereas the grouping that produced the Olympus 593 engine for Concorde (see next entry) was a loose partnership, in 2000 the two engine companies decided to form a jointly-owned company dedicated to developing the next major generation of engines for military aircraft. Thus, Rolls-Royce Snecma (increasingly, the latter name is being rendered with only the S capitalised) is a single company, formed on 15 February 2001. It is based at the London office of Rolls-Royce Turbomeca, whose telephone and fax numbers are temporarily given above. The Managing Director was previously Marketing Manager of Turbomeca.

The new company is owned 50/50 by the two partners. Among other things, it will manage the propulsion studies supporting the AMET (advanced military engine technology) and FOAS (future offensive air system) programmes.

NEW ENTRY

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AERO-ENGINES - TURBOFAN, INTERNATIONAL

Date Posted: 01 May 2001

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AEROSUD (PTY) LTD

SMR-95

This engine is a derivative of the Klimov RD-33 with installational changes to enable it to serve as a common engine able to power the later versions of the MiG-21, and the Mirage III and 5 and Mirage F1. Features include a completely new accessory gearbox, on the engine's underside, gas-turbine starter, oxygen injection for automatic relight under all flight conditions, twin 40-kVA alternators to give greatly increased electrical power, high-capacity fuel and hydraulic pumps, and an advanced hydromechanical fuel control giving `carefree' handling. Development began in 1990.

SMR-95A: The first SMR-95 was configured to fit the Mirage F1.AZ. The accessory drive gearbox is the AADG-52. No change in aircraft centre of gravity or thrust line was necessary, and only a slight shortening of the fuselage was needed to match the new and more efficient multiflap nozzle. Flight testing began in mid-1994, and has been brilliantly successful. The extra thrust transforms flight performance, while the combat radius is significantly extended by the greatly reduced fuel consumption, reduced engine weight (about 200 kg, 440 lb) and increase in permissible take-off weight of about 1,000 kg (2,205 lb). Further advantages are better stability during gunfire and dramatically faster engine acceleration, the time from flight idle to maximum rpm being only 3.5 seconds, with another 2.0 seconds being needed to bring in full afterburner.

Some 70 test flights were made with the re-engined F1.AZ, and the results were outstanding. Overall, radius of action was increased by 25 per cent, and combat effectiveness increased by factors from 1.2 to 3.0. However, the final decision was to retire the 30 F1.AZ aircraft and put them up for disposal (the replacement being the Gripen). This decision remains in 2001, but Marvotech (see separate entry) is conducting a phased flight-test programme in support of an upgraded F1 initiative which will involve not only the new engine but also a glass cockpit and new avionics matched to Vympel R-73 missiles. This is seen as applicable to other Mirage F1 users.

SMR-95B: Flight testing in Super Cheetah D2 (South African Mirage III derivative) began in early 1995. In this case, the accessory gearbox is the AADG-53. This resulted in even greater improvements, the increase in

permissible take-off weight being approximately 3,000 kg (6,615 lb). Later in 1995 Marvol, Aerosud and Armscor (the South African defence procurement and marketing agency) collaborated in a contract to upgrade Cheetah fighters of the SA National Defence Force. By 2001 the Marvotech group was engaged in marketing activities not only for the Cheetahs - some of which are being retained by the SANDF pending the delivery of Gripens, postponed from 2002 until 2007 - but also for other users of Mirage 3 and 5 family aircraft. The immediate prospect is Chile, with the Pantera (Mirage 50).

While flight testing was in progress the SANDF decided to purchase new-generation fighters (Saab/BAE Gripen) and retire the Dassault aircraft. The SMR-95 project was terminated. However, the improvements conferred by the Russian engine made the old French aircraft so much better than prediction that in 2000 it was decided not only to resume testing but also to seek foreign buyers, and Chile at once expressed interest in an SMR-95 upgrade of the FAC's Panteras (Mirage 50s).

The Klimov designation of the SMR-95 is RD-33N. Compared with previous RD-33 versions, the main difference is that the accessories are grouped underneath, driven by a tower shaft through the 6 o'clock strut. Less obvious are a modified lubrication system and altered control inputs to suit the airframe.

These are the first single-engine installations of engines of the RD-33 family. In such applications Klimov had established TBO at only 150 hours, but in 1994 a slightly modified RD-33 passed a 600-hour test, and following the introduction of a new combustion chamber the design goal has been raised to 2,000 hours. Moreover, the SMR-95 is expected to be re-rated at the same level as the RD-33K, of 88 kN (19,335 lb st).

Dimensions

Similar to Klimov RD-33

Weight, Dry

Complete power plant	1,225 kg (2,701 lb)

Performance Ratings (S/L)

Max augmented	81.4 kN (18,300 lb st)
Max dry	49.4 kN (11,110 lb st

Specific Fuel Consumption

Max dry thrust	21.66 mg/Ns (0.765 lb/h/lb st)

UPDATED

SMR-95

Installing SMR-95 in Super F1.AZ

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 01 May 2001

Jane's Aero-Engines 10

AEROSUD-MARVOL

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Managing Director: Dr Paul Potgieter
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MARVOL (GENERAL TECHNOLOGIES PTY LTD)

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Director-General: Yuri Golovin

These two groups have collaborated very successfully in developing a fighter engine derived from the Klimov RD-33 (see under Klimov in Russian section) but tailored as a retrofit engine for older fighters. This is described next. Marvol is a Russian partnership embracing Klimov Corporation and A I Mikoyan design bureau (jointly linked in RusJet), General Technologies, MAPO, Baranov and EGA. Aerosud comprises Atlas Aviation, Kentron, ATE, General Technologies, Xcel and French partners. It is the South African design authority for Mirage upgrades, which first established a relationship with RusJet studying a light utility transport.

In late 2000 these partners further linked with Promexport (an agency of the Russian government and defence industry) and Armscor (the South African defence procurement agency) to form Marvotech. This has been established to promote Russian/South African defence sales. Chairman is Joe Modise, former South African Defence Minister.

Also see Marvotech, later in the International section.

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 12 April 2001

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ROLLS-ROYCE TURBOMECA - ROLLS-ROYCE TURBOMECA LIMITED

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Co-operation between Rolls-Royce and Turbomeca started in 1965, when Rolls-Royce was licensed to use specified Turbomeca patents. A joint company, Rolls-Royce Turbomeca Ltd (RRTM), was formed in June 1966 to control the programme for the Adour. The relationship was extended under the Anglo-French helicopter agreement of 1967. This involved Turmo and Astazou engines, for RAF/Army Puma and Gazelle helicopters, being part manufactured, assembled and supported by Rolls-Royce. In return, Gem engines for French Navy Lynx helicopters were part manufactured, assembled and supported by Turbomeca.

In 1980, RRTM launched development of the RTM 321 turboshaft demonstrator. This ran for the first time in November 1983, and provided valuable background for the RTM 322, which was launched in May 1984.

Today, different versions of both the Adour and the RTM 322 are in production for various

fixed-wing aircraft and helicopters. The Adour has been licensed to three other countries.

The office of Turbomeca Ltd, the London support agency of Labinal's Turbomeca Microturbo Division, is co-located with RRTM at the same address (see introduction to Turbomeca, France).

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 09 January 2001

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MARVOTECH - MARVOTECH (PTY) LTD

CHAIRMAN: Joe Modise

This company was formed in mid-2000 to promote collaborative Russian/South African defence systems. Led by the former South African Minister of Defence, it has been established under an agreement between Marvol (see Aerosud-Marvol earlier in this section), Promexport (representing the Russian government and defence industry) and the South African arms company Armscor.

Its first major programme in the field of Aerospace is the flight-testing of Mirage/Cheetah aircraft fitted with the RD-33 engine (see under Klimov, Russia). This is in support of the SMR-95 engine programme (see under Aerosud-Marvol). According to Mikhail Dmitnev, Russian Deputy Minister for Science and Technology, this company, or one yet to be formed between it and MiG, Promexport and the missile firm Vympel, is to carry out assembly, support and upgrading of MiG-29 aircraft in southern Africa.

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 23 November 2000

Jane's Aero-Engines 09

ROLLS-ROYCE SNECMA

AMET

Rolls-Royce and SNECMA are poised to establish a joint venture (JV) focused on developing a new engine for Europe's next generation of combat aircraft. ``We are very close to an agreement'', a senior SNECMA official told *Jane's Defence Weekly*. It was aimed to have the new company up and running before the end of 2000.

``The new Anglo-French company will initially be focused on capturing research and development (R&D) contracts associated with the Advanced Military Engines Technology (AMET) programme, which is jointly funded by the French and UK defence ministries.''

``AMET is a bilateral risk-reduction effort aimed at producing a fighter engine to succeed the Eurojet EJ 200 in the Eurofighter multirole combat aircraft and the SNECMA M88 in the Dassault Aviation Rafale. AMET and the new JV are designed to avoid any overlap of effort and resources within France and the UK in related fields of endeavour.''

``Both Rolls-Royce and SNECMA will put their own money into the JV, but most of the funding is expected to come from government-funded research contracts.''

``Short-term activity is covered under packets of R&D work associated with AMET and national research programmes, said sources close to the discussions.''

``Projections for the long term are also healthy as they centre on the development of a new engine for the aircraft or unmanned combat air vehicle (UCAV) that will replace the tornado GR4 and strike versions of the Mirage 2000.''

``It is the medium to long term that is uncertain,'' officials close to the discussions said. ``We are having difficulty in identifying the funding line in current budgets in both France and the UK, even

though, industrially, we are ready."

``To test technologies for the EJ 200, which is still in the latter stages of development, Rolls-Royce ran a demonstrator engine, the XG-40, with a 10:1 power-to-weight ratio, for much of the 1980s and 1990s. If the Tornado/Mirage replacement is a manned aircraft it will need an engine power-to-weight ratio of at least 15:1."

``The UK has already set a tentative in-service date for its Future Offensive Air System - the aircraft or UCAV that results from all the studies and technology demonstrator work currently scheduled or planned - of around 2017".

NEW ENTRY

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 23 November 2000

Jane's Aero-Engines 09

ROLLS-ROYCE SNECMA - SNECMA

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These two companies collaborated on the power plant for Concorde, described following. In 1996 they were jointly studying engines for the next generation of fighters for the post-2020 period.

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AERO-ENGINES - TURBOFAN, INTERNATIONAL

Date Posted: 17 August 2000

Jane's Aero-Engines 08

TURBO-UNION LTD

TURBO-UNION RB199

The RB199 is a three-spool turbofan offering low fuel consumption for long-range dry cruise and approximately 100 per cent thrust augmentation with full afterburner for short take-off, combat manoeuvre and supersonic acceleration. An integral thrust reverser system is available. It was the first military engine with FADEC without hydromechanical back-up.

In-service experience of over 4.59 million flying hours, at low level in the most arduous conditions, has proven the resilience of the RB199 to birdstrike and foreign object damage (FOD). This is a direct result of the relatively short, rigid rotating assemblies held between the small bearing spans in a three-spool layout.

Over 4,500 engines have been produced, with the final 100 (including spares) engines for AYII (Saudi) having been delivered in 1996. The present engine family comprises:

Mk 103

Standard production engine, with integral thrust reverser, for Panavia Tornado IDS variants.

Mk 104

Identical to the Mk 103 other than the dressings and the jetpipe, which is extended by 360 mm (14 in) to provide up to 10 per cent greater thrust and reduced specific fuel consumption. The Mk 104 is the standard production engine for Tornado ADV variants.

Mk 104D

The power plant for the BAe experimental aircraft programme (EAP) advanced technology demonstrator.

Mk 104E

The interim power plant for Eurofighter DA1 and DA2 to enable flight testing to begin.

Mk 105

Similar to the Mk 103, the Mk 105 incorporates an increased mass flow LP compressor producing higher pressure ratios. In addition to a 10 per cent thrust increase, these improvements also give significant reductions in life cycle cost. In service as the power plant for the German Tornado ECR.

The following description refers to the Mk 103, with the responsibility for each module in brackets:

Type
Three-shaft turbofan with afterburner and reverser.

LP Compressor
(Rolls-Royce) Three-stage axial of titanium alloy. Casing of three bolted sections leading to titanium bypass duct (MTU). Rotor of three discs welded together. Rotor blades secured by dovetail roots, all with snubbers. Mass flow (103, 104) 74.6 kg (164 lb)/s, (105) 75.3 kg (166 lb)/s. Bypass ratio 1.0-1.1.

IP Compressor
(MTU) Three stages of titanium alloy. Rotor has welded discs in which blades are secured by dovetails.

HP Compressor
(MTU) Six-stage; material changes from titanium at front to heat resisting alloy at rear, except stator blades are heat resisting steel throughout. Rotor discs secured by 10 through-bolts, carrying blades by dovetail roots. Bevel drive to gearbox. Overall pressure ratio greater than 23.

Combustion Chamber
(Rolls-Royce) Annular flame tube fabricated from nickel alloy, bolted at rear end between outer casing, forged and chemically milled in nickel-iron alloy and inner casing of nickel alloy. Carries 13 double-headed fuel vaporisers which give combustion without visible smoke. Two igniter plugs. Hot-streak injector for afterburner ignition.

HP Turbine
(Rolls-Royce) Shrouded single stage with single-crystal blades. Entry temperature 1,317ºC. Rotor blades and stator vanes air-cooled.

IP Turbine
(MTU) Shrouded single stage with single-crystal blades. Air-cooled stator vanes and rotor blades.

LP Turbine
(Fiat) Two-stage with shrouded hollow uncooled rotor blades. Air-cooled Stage-1 vanes (stators).

Afterburner
(Rolls-Royce) Front end of titanium fabricated jetpipe carries afterburner in which bypass air and core gas burn concurrently without a mixing section. For core flow, two gutter flameholders fed by upstream atomisers. For bypass flow, reverse colander with radial extensions, each containing vaporising primary burner, between which multiple jets inject remainder of afterburner fuel. Fully modulated augmentation.

Reverser
(MTU) External two-bucket type driven via flexible shafts by motor using HP air. In stowed position outer

skins form aircraft profile. Deployment takes 1 second at any thrust setting from idle to maximum dry.

Nozzle
(Fiat) Variable area, short petal, convergent nozzle operated by shroud actuated by four screwjacks, driven by fourth-stage HP air motor via flexible shafting. Each of 14 master and 14 secondary petals is precision cast in cobalt alloy which minimises friction.

Accessories
(MTU) Accessory gearbox on underside of intermediate casing (quick attach/detach coupling) carries hydromechanical portions of main and afterburner fuel systems, oil tank and pump and output shaft to aircraft gearbox carrying KHD gas-turbine starter/APU.

Control System
Electronic (analog and digital versions available) main engine control unit uses signals from pilot's lever and power plant sensors. Afterburner fuel from engine-driven vapour core pump.

Dimensions

Length overall: Mk 103	3,251 mm (128 in)
Mk 104	3,607 mm (142 in)
Mk 105	3,302 mm (130 in)
Intake diameter: Mks 103, 104	719 mm (28.3 in)
Mk 105	752 mm (29.6 in)

Weight, Dry
(excl reverser):

Mk 103	965 kg (2,107 lb)
Mk 104	976 kg (2,151 lb)
Mk 105	991 kg (2,185 lb)

Performance Ratings (S/L, ISA)

Max dry: Mks 103, 104	40.48 kN (9,100 lb st)
Mk 105	42.50 kN (9,650 lb st)
Max afterburning: Mk 103	71.17 kN (16,000 lb st)
Mk 104	72.95 kN (16,400 lb st)
Mk 105	74.10 kN (16,700 lb st)

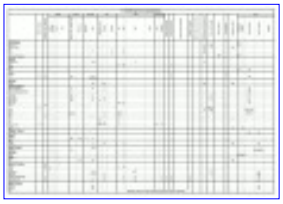
Specific Fuel Consumption (Max dry)

Mk 103, 104	18.4 mg/Ns (0.649 lb/h/lb st)
Mk 105	18.43 mg/Ns (0.650 lb/h/lb st)

UPDATED

Assembling two of the final batch of RB199 engines for the AYII contract

Cutaway drawing of RB199 Mk 103



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AERO-ENGINES - TURBOJET, INTERNATIONAL

Date Posted: 17 December 1999

Jane's Aero-Engines 07

SNECMA

ROLLS-ROYCE SNECMA OLYMPUS

Though private restoration groups are trying to return Vulcan bombers (Olympus 301) to airworthy status, the only Olympus engine currently in operation is the Concorde power plant, the Olympus 593 Mk 610-14-28. This has a convergent/divergent exhaust nozzle, thrust reverser and afterburner system.

Preflight Olympus development engines, designated 593D, were used for bench testing from mid-1964. The first of the Olympus 593 flight-type engines made its initial test run in November 1965. A Vulcan testbed, with a single Olympus 593 mounted beneath its fuselage in a representative Concorde half-nacelle, assisted flight development from September 1966 to July 1977. Concordes have been flying since March 1969.

Production standard Olympus 593 engines powered preproduction and production Concordes. In March 1974, a production standard engine, the Olympus 593 Mk 610, successfully completed an official 150 hour type test. Full certification was achieved in April 1975, when total running time exceeded 40,000 hours.For political reasons only 14 Concordes entered service, seven each for BA and Air France, in 1976. On 2 March 1999, the 30th anniversary of Concorde flying, an average fleet total of 10 aircraft had flown 920,000 hours. This total included considerably more than 600,000 hours at supersonic speeds, easily exceeding the total supersonic time of all other aircraft in the Western world.In 1999 the time for the fastest flight NY-London was just under 2 hours 53 min.

The following description refers to the production engine, the 593 Mk 610:

TYPE: Axial-flow, two-spool turbojet with partial afterburning.

INTAKE: Fabricated titanium casing, with zero-swirl five-spoke support for the front LP compressor bearing. In the Concorde, the engine is installed downstream of an intake duct incorporating auxiliary intake and exit door systems and a throat of variable profile and cross-section.

LP COMPRESSOR: Seven-stage axial-flow type, with all blading and discs manufactured from titanium. Single-piece casing machined from a stainless steel forging, electrochemically machined.

HP COMPRESSOR: Seven-stage axial-flow compressor. The first three stages of blades are made from titanium alloy. Remaining stages are made from a heat-resistant material due to very high compressor delivery temperatures during supersonic flight. Steel single-piece casing. Mass flow 186 kg (410 lb)/s. Overall pressure ratio 15.5:1.

INTERMEDIATE CASE: Titanium casing, with vanes supporting LP and HP thrust bearings. Drives for engine-mounted aircraft and engine auxiliary drive gearboxes are taken out through the intermediate casing.

COMBUSTION CHAMBER: Annular cantilever mounted from the rear. Fabricated as single unit from nickel alloy, with all joints butt-welded to ensure reliability. Electrochemically machined. The combustion system burner manifold and the main support trunnions are located around the delivery casing. Total of 16 vaporising burners, each with twin outlets, bolted directly into chamber head. Fuel injectors are simple pipes which enter each vaporiser intake with no physical contact. Combustion leaves virtually no visible smoke in the propulsive jet.

HP TURBINE: Single-stage turbine, with cooled stator and rotor blading.

LP TURBINE: Single-stage, with cooled rotor blades. LP driveshaft coaxial with HP shaft.

JETPIPE: Comprises a straight jetpipe and a pneumatically actuated variable primary convergent nozzle which permits maximum LP-spool speed and turbine-entry temperature to be achieved simultaneously over a wide range of compressor-inlet temperatures. Single-ring afterburner with programmed fuel control as a function of main-engine fuel flow. Monobloc secondary nozzle with each twin nacelle manufactured from Stresskin panels. Each power plant terminates in a pair of `eyelids' which form a variable-area secondary divergent nozzle and thrust reverser. The eyelid position is programmed to maintain optimum power plant efficiency through all the flight regimes: take-off, subsonic cruise and supersonic cruise. When completely closed they act as thrust reversers.

MOUNTING: Main trunnions on horizontal centreline of the delivery casing. Allowance for expansion contained within aircraft pickups. Front stay from roof of the nacelle picks up on the top of the intake casing.

ACCESSORIES: Beneath the compressor intermediate casing are two gearboxes, both mechanically driven off the HP shaft (the LP shaft only has a pulse-probe signal source and provision for hand or mechanical turning). The LH gearbox drives the main engine oil pressure/scavenge pumps and the first-stage fuel pump and fuel control unit. The RH gearbox drives the aircraft hydraulic pumps and integrated-drive generator/alternator.

STARTING: SEMCA air-turbine starter drives the HP spool. Dual high-energy ignition system serves igniters in the annular chamber.

CONTROL SYSTEM: Lucas system, incorporating a mechanically driven first-stage pump and a second-stage pump driven by an air turbine which is shut down at altitude cruise conditions as fuel requirements can be met by the first-stage pump alone. The first-stage pump also supplies afterburner fuel. A fuel-cooled oil cooler is incorporated.An Ultra electronic system - the world's first FADEC in service - with integrated-circuit amplifier, provides combined control of fuel flow and primary nozzle area. Afterburner fuel is controlled by an ELECMA electrical control unit. The fuel system of the production Olympus 593 is substantially lighter than the one previously in use, and it operates at pressures of about one-half those on the earlier system. It also has improved maintenance and installation characteristics.

The principal difference is that the piston-type HP pump is replaced by an air turbopump. At altitude cruise conditions, sufficient pressure is available from the first-stage pump alone and the air turbopump is shut down.

FUEL SPECIFICATION: DERD.2494 Issue 7, AIR 3405B (3rd edition, amendment 1), ASTM D-1655-71 (Jet A) and ASTM D-1655-71 (Jet A1).

OIL SYSTEM: Closed system, using oil to specification DERD.2497, MIL-L-9236B. Pressure pump, multiple scavenge pumps and return through Serck fuel/oil heat exchanger.

DIMENSIONS:

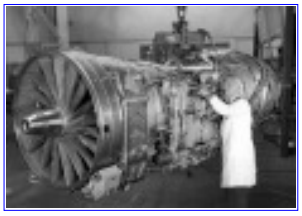
Length (flange to flange)	4,039 mm (159 in)
Length (flange to nozzle)	7,112 mm (280.0 in)
Diameter (inlet)	1,212 mm (47.75 in)

WEIGHT, DRY:

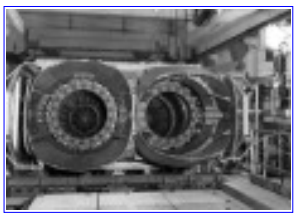
Bare engine	2,971 kg (6,550 lb)
With afterburner, reverser and nozzle	3,175 kg (7,000 lb)

PERFORMANCE RATINGS (T-O, S/L, ISA):

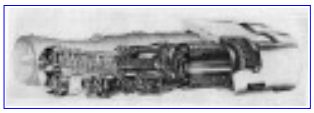
Dry	139.4 kN (31,350 lb st)
Afterburner	169.2 kN (38,050 lb st)



The full designation of the Concorde engine is Olympus 593 Mk 610 (1996)



Nozzles of the Mk 610 (Type 28), showing one open, the other partly closed for subsonic cruise (1996)



Cutaway drawing of twin-engine Concorde nacelle (1996)

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 17 December 1999

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CHAIRMAN: R Tomlinson
MANAGING DIRECTOR: Karlheinz Koch

Formed in 1969 as a European engine consortium comprising Rolls-Royce plc of the UK (40 per cent), MTU Motoren- und Turbinen-Union München GmbH of Germany (40 per cent), and FiatAvio SpA of Italy (20 per cent). The consortium was established to design, develop, manufacture and support the RB199 turbofan for the Panavia Tornado aircraft.

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 17 December 1999

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MANAGING DIRECTOR: H Fischer
TECHNICAL DIRECTOR: M Lauvaux
PROGRAMME DIRECTOR: A Jansen
CUSTOMER SUPPORT DIRECTOR: G Schuberth
COMMERCIAL DIRECTOR: H Seguinotte

This company is owned equally by the three participants. It was set up in 1989 to produce and subsequently support the MTR 390 engine, and to act as contractor for customers.

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AERO-ENGINES - TURBOSHAFT, INTERNATIONAL

Date Posted: 17 December 1999

Jane's Aero-Engines 07

RM-GE - RYBINSK MOTORS - GENERAL ELECTRIC JOINT VENTURE

CT7

Under this designation Rybinsk Motors and GE Aircraft Engines will jointly develop a family of turboshaft, turboprop and derived engines all based on the T700/CT7 described under General Electric (USA section). These engines, to be rated at 1,119-1,492 kW (1,500-2,000 shp), will be co-produced by the Joint Venture at Rybinsk, and subsequently supported by Rybinsk Motors.

CT7 turboprop

Under an agreement between GE Aircraft Engines and Sukhoi Design Bureau, twin engines of this type will power the S-80 multirole transport. The prototype of this 26-passenger aircraft is scheduled to fly in late 2000, powered by US-built GE CT7-9 engines in the 1,305-kW (1,750-shp) class, driving Hamilton Standard four-blade propellers. Originally the S-80 was to have been powered by Russian TVD-1500S engines driving six-blade Stupino propellers.

CT7 turboshaft

Two engines of this type are expected to power the 16-passenger Kamov Ka-64 helicopter, which was originally scheduled to fly in 1998. The Ka-64 is the intended production version of the long-awaited Ka-62, which had previously been announced as having Rybinsk RD-600 engines, with the option of GE CT7-2D1 engines (1,212 kW, 1,625 shp) for export.

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 17 December 1999

Jane's Aero-Engines 07

RM-GE - RYBINSK MOTORS - GENERAL ELECTRIC JOINT VENTURE

RYBINSK MOTORS (Russia)

GE AIRCRAFT ENGINES (USA)

MANUFACTURER DETAILS

In 1996 these two companies signed an agreement to establish a Joint Venture for co-production in Russia of GE aircraft engines and their industrial and marine derivatives. These engines will subsequently be supported by Rybinsk Motors. One of the initial products will be the LM2500. This is the family of engines in the 14,920-18,650-kW (20,000-25,000-shp) class with a gas generator derived from the TF39, used for ship propulsion, gas pumping and electricity generation.A second initial product is the CT7, which is described next. Also see Rybinsk-GE (Russia).

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AERO-ENGINES - TURBOFAN, INTERNATIONAL

Date Posted: 17 September 1999

Jane's Aero-Engines 06

ROLLS-ROYCE ALLISON ENGINE CO

ROLLS-ROYCE ALLISON TF41

Manufacturers' designations: Rolls-Royce Spey RB.168-62 and -66, Allison Model 912-B3 and -B14

This engine is purely military and no longer in production and retains the Allison name.

In August 1966 Allison and Rolls-Royce were awarded a joint contract by USAF Systems Command for the development and production of an advanced version of the RB.168-25 Spey turbofan, to power LTV A-7D Corsair II fighter-bomber aircraft for the USAF. The requirement was to provide an engine offering maximum thrust increase over the TF30-P-6 powering USN A-7As.

Development and production were undertaken jointly by Rolls-Royce and Allison, with Rolls-Royce supplying parts common to existing Spey variants and Allison, which manufactured under licence, being responsible for items peculiar to the TF41. This provided an approximately 50/50 division of manufacturing effort, but with Allison also undertaking assembly, test and delivery.

TF41-A-1

Design of the RB.168-62 started in June 1966 and the engine was given the USAF designation TF41-A-1. Major change compared with the RB.168-25 was the move forward of the bypass flow split

into the LP compressor, to give a larger three-stage fan followed by a two-stage IP compressor, all five stages being driven by the two-stage LP turbine. The number of HP compressor stages was reduced from 12 to 11, the HP turbine remaining at two stages. These modifications raised the mass flow to 117 kg (258 lb)/s, and the bypass ratio from 0.7 to 0.76.

Other design changes compared with the RB.168-25 included omission of the fan inlet guide vanes, the first rotor stage being overhung on a bearing supported by the first-stage stator vanes. The fan and IP compressor are of more modern aerodynamic design, and the HP and LP turbine nozzle throat areas were increased to pass the additional flow. The HP turbine is of modified aerodynamic design, and an annular exhaust mixer replaced the RB.168-25's chuted design.

First run of the TF41-A-1/RB.168-62 was at Rolls-Royce, Derby, in October 1967, the first Allison engine following at Indianapolis in March 1968. Development continued ahead of schedule, delivery of the first production TF41-A-1 being made in June 1968.

TF41-A-2

Ordered in 1968, a second version of the TF41 is the A-2, developed for the US Navy to power the LTV A-7E Corsair. Differences are slight, although the thrust rating is appreciably increased by raising the engine speed. This required restressing the disc of the low-pressure turbine and high-pressure compressor. Mass flow was slightly increased, the bypass ratio being 0.74. The engine has additional protection against corrosion.

Allison delivered a total of 1,440 TF41 engines, the last being shipped in 1983. In combat service both versions of the TF41 have shown outstanding reliability.

The two production versions are known to Rolls-Royce as the RB.168-62 and RB.168-66; the corresponding Allison designations are Model 912-B3 and 912-B14. The following description refers basically to the TF41-A-1; where the A-2 differs, the data for that engine are given in brackets.

TYPE: Two-shaft turbofan.

INTAKE: Direct entry, fixed, without intake guide vanes.

COMPRESSOR: Two-shaft axial. Three fan stages, two intermediate stages on same shaft and 11 high-pressure stages. All rotor blades carried on separate discs. Fan and LP rotor blades of titanium, held by dovetail roots in slots broached in discs which are bolted together through curvic couplings and similarly attached to the stubshafts. HP rotor blades also of titanium except stages 9, 10 and 11 of stainless steel, the first HP stage being pinned and the remainder being dovetailed into broached slots; discs similarly bolted together but driven through a splined coupling to the shaft. LP rotor carried in three roller bearings and HP by two, with central ball location bearing and intershaft ball bearing. LP casing of steel and aluminium; HP casing of stainless steel, both split at horizontal centreline. Stainless steel LP stator blades slotted laterally into casing, intermediate stators welded to inner casing subassembly rings. HP stator blades of stainless steel, slotted laterally into casing. Overall pressure ratio 20 (A-2, 21.4); mass flow 117 kg (258 lb)/s (A-2, 119 kg; 263 lb/s). Compressor pressure ratio, 6.2; mass flow, 67 kg (148 lb)/s. Bypass ratio 0.75.

COMBUSTION CHAMBER: Tubo-annular, with 10 interconnected Ni-Co alloy flame tubes in steel outer casing. Duple spray atomising burner at head of each chamber. High-energy 12-joule igniter plug in chambers 4 and 8.

HP TURBINE: Two stages: All rotor blades forged Ni-Co, first stage cooled. Disks Inco 901, NGVs cast Ni-Co with air cooling.

LP TURBINE: Two stages. Solid rotor blades in steel discs, uncooled NGVs with disc cooling air piped through first stage. All discs bolted to shafts.

JETPIPE: Fixed, heat-resistant steel.

MOUNTING Main ball-type trunnions on compressor intermediate casing; rear tangential steady-type at rear of bypass duct.
ACCESSORIES: External gearbox driven by radial shaft from HP system; provision for starter, fuel boost pump, two hydraulic pumps, HP fuel pump, fuel control, HP tachometer, CSD and alternator, permanent-magnet generator, LP fuel pump and oil pumps. Additional low-speed (LS) gearbox, driven from LP shaft, serving LP rotor governor and tachometer.
STARTING: Integral gas turbine (air turbine).
CONTROL SYSTEM: Hydromechanical high-pressure system with automatic acceleration and speed control. Emergency manual override of automatic features. Variable-stroke dual fuel pump.
FUEL SPECIFICATION: JP-4 (A-2, JP-5).
OIL SYSTEM: Self-contained, with engine-mounted tank, fuel/oil heat exchanger and gear-type pump; pressure 3.51 kg/cm^2 (50 lb/sq in). Tank capacity: A-1, 4.5 litres (1.2 US gallons, 1 Imp gallon); A-2, 10.3 litres (2.72 US gallons, 2.27 Imp gallons).
DIMENSIONS:

Length overall	2,610 mm (102.6 in)
Intake diameter	953 mm (37.5 in)
Height overall	1,026 mm (40.0 in)

WEIGHT, DRY:

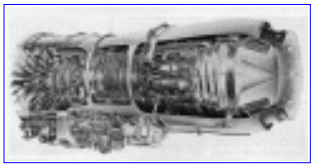
A-1	1,440 kg (3,175 lb)
A-2	1,470 kg (3,241 lb)

PERFORMANCE RATINGS (T-O, S/L, ISA):

A-1	64.5 kN (14,500 lb st)
A-2	66.7 kN (15,000 lb st)

SPECIFIC FUEL CONSUMPTION: (T-O, as above):

A-1	17.92 mg/Ns (0.633 lb/h/lb st)
A-2	18.32 mg/Ns (0.647 lb/h/lb st)



Cutaway drawing of TF41 (1996)

TF41-A-2 (1996)

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AERO-ENGINES - MANUFACTURER, INTERNATIONAL

Date Posted: 17 September 1999

Jane's Aero-Engines 06

ROLLS-ROYCE ALLISON

MANUFACTURER DETAILS

ROLLS-ROYCE plc
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ROLLS-ROYCE ALLISON ENGINE CO

MANUFACTURER DETAILS

ALLISON ENGINE CO
PO Box 420, Indianapolis, Indiana 46206-0420, USA

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AERO-ENGINES - TURBOFAN, INTERNATIONAL

Date Posted: 17 September 1999

Jane's Aero-Engines 06

JV - JOINT VENTURE

SPW

The SPW family of turbofans will be wholly new and uncompromised, and in no sense derivatives of any existing engines by either partner. They are designed to cover the thrust spectrum 12-16K, and to win markets from GE, Rolls-Royce and BMW Rolls-Royce.

At the 1996 Farnborough airshow, P&WC Chairman David Caplan said ``The split is not across the hot and cold boundaries. Pratt & Whitney Canada are responsible for the compressor and the fan (LP) turbine, while SNECMA are handling the fan and the compressor (HP) turbine".

Predictably, basic design objectives include low cost of ownership (implying minimal parts-count), the best possible environmental performance and 10,000 hour on-wing reliability from EIS. The partners have completed a considerable amount of preliminary work, including some component testing, but as noted below, in mid-1999 the engine's development was on hold.

Additional risk-sharing partners have been courted from the outset. An obvious possible move would be for MTU to take over the LP turbine, a field in which the German company has a high reputation as an existing supplier to many companies including P&WC. On the other hand, another market leader, RoSEC, might consider that supplying the FADEC would not help one of its partners, Rolls-Royce. This engine would be an excellent launch platform for the new-concept Papillon reverser by Hurel-Dubois.

SPW12

An engine in the 12K (53.4 kN, 12,000 lb st) bracket would be suitable for large long-range business aircraft. It is envisaged as the engine of the AIRjet-100, should this be launched.

SPW14

This is the baseline SPW engine, for the ATR project in the 70-seat class. This aircraft will need engines rated at about 62 kN (13,940 lb st), hence the SPW14 designation. No development schedule for the SPW14 has been published, though David Caplan has said that the target would be to achieve certification ``in less than 3.5 years from go-ahead''. Originally this was expected to mean 2001, but a firm application has taken a long time to emerge. In December 1996 the SPW14 was selected by ATR as a finalist in the competition to power what at that time was called the AI(R) twin-jet. In 1997 this aircraft was restyled AIR-70, reflecting the number of seats, and in 1998 it was again redesignated as the 70-75-seat AIRjet-200. By this time the SPW14 was the firm engine choice, but in mid-1999 the aircraft had still not been launched, and the development of the engine was continuing to be on hold. At that time the tentative in-service date for the AIRjet-200 was 2003. ATR also plans a stretched 95-105-seat version called AIRjet-300, which would have an engine of slightly greater thrust (possibly to be called SPW15).

SPW16

The basic design is being planned to accommodate growth to this thrust level. Such an engine would be a candidate for the Japanese YS-X and IPTN N-2130. The AE-100 has also been mentioned, but this is in a heavier category.

The following describes the SPW14:

TYPE: Two-shaft turbofan.

FAN: Single stage, with 22 snubberless titanium wide-chord blades in solid titanium hub. No core booster.

COMPRESSOR: A single spool, probably EB-welded, comprising a four-stage axial (F414 derived) followed by a single centrifugal. Variable inlet guide vanes and first-stage stators.

COMBUSTION CHAMBER: Minimum length unfolded single annular, with multiple airblast fuel nozzles. P&WC-style multiple pipes lead air from centrifugal diffuser.

HP TURBINE: Two air-cooled stages.

LP TURBINE: Three stages.

ACCESSORIES: Below fan case, driven by diagonal tower shaft from HP spool.

DIMENSIONS (baseline, rough estimates):

Fan diameter	1,146 mm (45.1 in)
Length overall	2,540 mm (100 in)
WEIGHT DRY:	In the region of 1,361 kg (3,000 lb)

PERFORMANCE RATING (T-O, S/L):

Bracket for family	53.4-71.2 kN (12,000-16,000 lb st)

SPECIFIC FUEL CONSUMPTION:

Not disclosed but cruise target probably near 17 mg/Ns (0.6 lb/h/lb).

Longitudinal section through SPW14 (preliminary) (1997)

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JV - JOINT VENTURE

MANUFACTURER DETAILS

SNECMA

F-75724 Paris, France

PRATT & WHITNEY CANADA

Longueuil, Quebec

On 17 April 1996, the above companies issued the following statement:

`` SNECMA of France and Pratt & Whitney Canada have signed a Memorandum of Understanding (MoU) to jointly develop and market a new turbofan engine in the 12,000 to 16,000 lb thrust class.

With close to 6,000 aircraft in airline service around the world powered by engines from SNECMA or P&WC, the two partners bring to this programme a vast experience with both major and regional airlines together representing more than 120 million flight hours with over 800 `regional airlines and major carriers'.

This agreement brings together two of the world's leading aero-engine companies into a new transatlantic alliance that will leverage their respective technologies, industrial capabilities and global customer support presence to provide the most competitive, cost-effective engine for 70 to 90 passenger turbofan-powered regional aircraft."

This announcement caused gasps of astonishment. On the one hand SNECMA appeared to be competing with the CFM56-9 (previously CFM56-Lite), which they are developing in partnership with GE, and with GE's own CF34-8 family. At the same time the Canadian firm appeared to be entered into head-on conflict with their United Technologies parent's PW6000.

In fact there is no conflict, and the Joint Venture has the blessing of the boards of both SNECMA and United Technologies. The French partner, whose name has come first in all announcements, has taken pains to confirm that its new Chairman fully supports the venture. The MoU of April 1996 was followed by signature of a definitive agreement at the 1996 Farnborough airshow.

Identified as the SPW family, the new engines are not (at present, at least) intended to be developed to thrusts higher than 16K (16,000 lb st), which is the lower end of the spectrum for the PW6000 and below the baseline of the CFM56-9 (though at a press conference at the 1996 Farnborough airshow GE said ``That's a decision by SNECMA with Pratt, their decision entirely'').

At the same airshow the partners announced a market broadened to ``58 to 90-passenger regional transport aircraft together with heavy long-range business jets''. The partnership was identified as the JV (Joint Venture). Incidentally P&WC is predominantly a French-speaking company.

Following the decision by Aérospatiale, Alenia and BAe not to go ahead with the AI(R) 70, the Joint Venture partners decided in January 1998 to shelve the SPW14. The following entry remains, however, in case this engine is resurrected in the future.

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AERO-ENGINES - TURBOJET, IRAN

Date Posted: 30 August 2000

Jane's Aero-Engines 08

IAIO

TOLLOUE 4

TOLLOUE 4

This engine, the first aircraft engine to be revealed as designed and constructed in Iran, is claimed to have been created entirely indigenously, with the exception of reverse-engineering `certain compressor parts'. Its existence was revealed in September 1999 by Rear-Admiral Ali Shamkhani, the Iranian Minister of Defence.

A simple engine intended for the propulsion of targets, cruise missiles and other UAVs (for example, for reconnaissance and electronic warfare), the Tolloue 4 will enable the TEM design and development staff to gain experience in a hands-on manner. In the course of time, possibly with external assistance, larger and more complex engines may be expected to appear.

Type
Single-shaft turbojet.

Compressor
Three axial stages.

Combustion Chamber
Annular.

Turbine
Single-stage axial.

Jetpipe
Sheet-metal, fixed-area nozzle.

Dimensions

Length	1,330 mm (52.36 in)
Diameter	330 mm (13.0 in)

Weight, Dry

	54.7 kg (120.6 lb)

Performance Rating

T-O	3.7 kN (832.7 lb st) at 29,500 rpm

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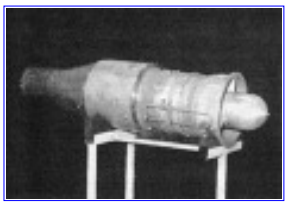

Tolloue 4 (Homa Farmehr) (2000)

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AERO-ENGINES - MANUFACTURER, IRAN

Date Posted: 30 August 2000

Jane's Aero-Engines 08

IAIO - TEM (Turbine Engine Manufacturing)

IRAN AVIATION INDUSTRIES ORGANIZATION
107 Sepahbod Gharani Avenue, Tehran
Tel: (+98 21) 882 50 43 (to 50 48)
Fax: (+98 21) 882 79 05

IAIO is a national organisation controlled by the Ministry of Defence. Though its main business is the manufacture and support of aircraft, it has a subsidiary, TEM, concerned with gas-turbine engines. In 1999 it was the intention that this organisation should become independent (though still controlled by the MoD). It has large workshops and other facilities, and in late 1999 revealed a small turbojet of its own design. Though this is intended for unmanned vehicles, it is included in *Jane's Aero-Engines* because of its potential for further development. It is the intention of Iran to become as self-sufficient as possible, especially in providing equipment for its armed forces. Whether TEM will be able to produce engines for combat aircraft remains to be seen.

NEW ENTRY

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AERO-ENGINES - TURBOPROP/TURBOSHAFT, ITALY

Date Posted: 17 August 2000

Jane's Aero-Engines 08

FIATAVIO

GENERAL ELECTRIC T700/CT7

Parts of the T700 are made by FiatAvio, while for the EH 101 helicopter the CT7-6C is being developed by GE, FiatAvio and Alfa Romeo Avio. FiatAvio jointly developed the CT7-8.

UPDATED

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AERO-ENGINES - MANUFACTURER, ITALY

Date Posted: 17 August 2000

Jane's Aero-Engines 08

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Fax: (+39 19) 69 09 59
e-mail: fi21.piaggio@interbusiness.it
Web: http://www.piaggioaero.it

The Aero-Engine Division of Piaggio manufactures and supports the following engines under licence agreements: Rolls-Royce Viper 11, 526, 540 and 632-43 turbojet and Gem 1004 turboshaft; Honeywell (AlliedSignal Lycoming) T53-L-13B/D/-703 and T55-L-712 turboshafts, and LTP-101-700A1 turboprop. Piaggio also participates in co-production under licence of the Rolls-Royce Spey 807 turbofan and has joined Rolls-Royce Turbomeca in development and production of the RTM 322-01

turboshaft. The Engine Division also develops and produces IR suppression devices.

UPDATED

Piaggio-built Gem Mk 1004 (2000)

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AERO-ENGINES - MANUFACTURER, ITALY

Date Posted: 17 August 2000

Jane's Aero-Engines 08

FIAT - FIATAVIO

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Tx: 221320 FIATAV

MANAGING DIRECTOR: P G Romiti
EXTERNAL RELATIONS: Jessica M Boriani

Tel: (+39 11) 685 83 40
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AERO-ENGINES - MANUFACTURER, ITALY

Jane's Aero-Engines 03

ALFA ROMEO AVIO SpA - ALFA ROMEO AVIO

MANUFACTURER DETAILS

I-80038 Pomigliano D'Arco, Naples
Tel: +39 (81) 8430111
Tx: 710083 ARAVIO

CHAIRMAN: Gen Fulvio Ristori
MANAGING DIRECTOR: Ing Filippo De Luca

Alfa Romeo Avio was prime contractor for the manufacture, under General Electric licence, of the J85, J79 and T58. It manufactures CF6 combustors and JT9D components, and assembles PT6T engines for the AB 212. Under GE licence it is responsible for the hot section of the T64-P4D, co-produced with FiatAvio.

In February 1986 it began deliveries of GE T700-401 engines for EH 101 prototypes. It supplies components for T700 engines fitted to American helicopters and is developing new versions. It is also involved, with FiatAvio, in the development of the GE CT7-6, aimed at the EH 101, NH 90 and a new version of the A 129.

Again, via FiatAvio, it has participated in Turbo-Union's RB199 production. The company is a partner in Italian production of the Rolls-Royce Spey 807. In November 1988 it became a 6.4 per cent partner in the Rolls-Royce Tay programme.

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AERO-ENGINES - TURBOFAN, JAPAN

Date Posted: 09 January 2001

Jane's Aero-Engines 09

ISHIKAWAJIMA-HARIMA JUKOGYO KABUSHIKI KAISHA

IHI F3

Development of this turbofan began in 1976, with funding by the Japan Defence Agency (JDA) Technical Research & Development Institute. The Phase 1 **XF3-1** form had a single-stage fan with bypass ratio of 1.9, five-stage transonic compressor, 12-burner combustor and single-stage HP and LP turbines. Rating was 11.79 kN (2,650 lb st).

In 1977, JDA contracted with IHI for the **F3-20**, with reduced bypass ratio and higher turbine temperature to give a rating of 16.28 kN (3,660 lb st). This was followed by the **XF3-30** which, in 1982, was selected by the JASDF as the engine for the T-4 trainer. XF3-30 qualification was completed in March 1986. The engine is now redesignated **F3-IHI-30**, and the first production engine was delivered to JDA on 17 December 1987. Deliveries by March 2000 totalled 520, with 18 due to be produced in 2000.

The following refers to the F3-IHI-30:

Type

Two-shaft turbofan.

Fan

Two-stage axial. No inlet guide vanes. Mass flow 34 kg (75 lb)/s. Pressure ratio 2.6. Bypass ratio 0.9.

Compressor

Five stages. First two stators variable. Overall pressure ratio 11.

Combustion Chamber

Annular, with 12 duplex fuel nozzles.

HP Turbine
Single-stage, air-cooled rotor blades.

LP Turbine
Two-stage, tip shrouded.

Control System
Hydromechanical, with electronic supervisor.

Dimensions

Length	1,340 mm (52.76 in)
Inlet diameter	560 mm (22.05 in)

Weight, Dry

	340 kg (749.6 lb)

Performance Ratings (T-O, S/L)

	16.37 kN (3,680 lb st) class

Specific Fuel Consumption (T-O)

	19.83 mg/Ns (0.7 lb/h/lb st)

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F3-IHI-30

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AERO-ENGINES - MANUFACTURER, JAPAN

Date Posted: 09 January 2001

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IHI - ISHIKAWAJIMA-HARIMA JUKOGYO KABUSHIKI KAISHA (Ishikawajima-Harima Heavy Industries Co Ltd)

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Aero-Engine and Space Operations (ASO)

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PRESIDENT, ASO: Tohru Ishikawa
GENERAL MANAGER, PUBLIC RELATIONS, PLANNING AND CONTROL:
Koichi Suzuki

IHI's Aero-Engine & Space Operations specialises in the development and manufacture of aero-engines, space-related equipment, and land/marine gas turbines, as well as maintenance and repair. It has four plants (Tanashi, Kure No 2, Mizuho and Soma), and in June 2000 had 3,160 employees. The number of jet engines so far produced exceeds 4,300. Sales in 1999 were ¥2.02 billion. IHI states that this means it has 71.8 per cent of the Japanese aero-engine business.

IHI began production of the J3 turbojet using Japan's own technology in 1959. This was followed by the licensed production of the J79, T64, T58 and TF40 (Adour) engines. In recent years, the F100, F110,

T56, F3 and T700 have been added to the product line.

The company has been involved in numerous engine development projects, including the national project for the FJR710 and the Japan-Britain joint project for the RJ500. Currently, as the leader of a Japanese consortium, IHI is participating in the IAE V2500. IHI also shares in the production of the GE90, RB.211, Trent and CF34-8 and 10. IHI is working on a new engine in the 50 kN (11,240 lb st) class.

IHI is actively involved in many aspects of spaceflight.

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AERO-ENGINES - TURBOSHAFT, JAPAN

Date Posted: 30 August 2000

Jane's Aero-Engines 08

MITSUBISHI JUKOGYO KABUSHIKI KAISHA (Mitsubishi Heavy Industries Ltd)

MHI TURBOSHAFTS

MG5: Development of this turboshaft engine began in 1987, and the prototype MG5 engine, rated at 597 kW (800 shp) ran in 1991. In 1993 the Mitsubishi RP1 experimental helicopter flew with two **MG5-10** engines each derated to 447 kW (600 shp). In 1995 the **MG5-100**, with a T-O rating of 597 kW (800 shp) and FADEC control, was selected to power the Mitsubishi MH2000 twin-engined civil helicopter. This engine was certificated by the Japan Civil Aviation Bureau in June 1997. In turn, this engine was uprated to produce the MG5-110, with T-O rating of 653 kW (876 shp). This powers the MH2000A helicopter, which was certificated by the JCAB in the TA category on 24 September 1999.

TS1

Development of this turboshaft engine began in 1991. The XTS1 first ran in 1993. The production engine, the **TS1-10**, was certified by the Japan Defence Agency in 1999. This engine has a 30-minute rating of 659 kW (884 shp), and is fitted with an inlet particle separator and IR suppressor. Its first application is to power the twin-engined Kawasaki OH-1 armed scout/observation helicopter. The OH-X prototype, powered by XTS1-10 engines, made its first flight in late 1996. The first OH-1 production machine, powered by TS1-M-10 engines, was delivered in January 2000.

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AERO-ENGINES - MANUFACTURER, JAPAN

Date Posted: 30 August 2000

Jane's Aero-Engines 08

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Since 1976 MHI has produced 179 CT63 turboshafts for OH-6J helicopters. Between January 1973 and June 1981, under a licence agreement with Pratt & Whitney, MHI delivered 72 JT8D-M-9 turbofans. MHI entered into a risk- and revenue-sharing agreement on the JT8D-200 in 1984, and on the PW4000

in 1989. In collaboration with IHI and Kawasaki, MHI participates in the V2500 (see IAE in the International section). MHI has developed the TJM2 and TJM3 turbojets in the 1.96 kN (441 lb st) class for missiles and targets; 916 have been delivered since 1987. Preliminary details follow of MHI's current aero-engine programmes. MHI's own engines now power civil and military Mitsubishi and Kawasaki helicopters.

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AERO-ENGINES - TURBOJET, JAPAN

Date Posted: 17 August 2000

Jane's Aero-Engines 08

ISHIKAWAJIMA-HARIMA JUKOGYO KABUSHIKI KAISHA (Ishikawajima-Harima Heavy Industries Co Ltd)

IHI J3

The J3 turbojet was developed from 1956 by what was then the Nippon Jet Engine Co. From it were derived the following production versions:
J3-IHI-7B: T-O rating 13.7 kN (3,090 lb st). Powers Fuji T-1B trainer of the JASDF.
J3-IHI-7C: T-O rating 13.7 kN (3,090 lb st). Direct single inlet duct to match booster pod of Kawasaki P-2J of JMSDF, Replaced by 7D version.
J3-IHI-7D: T-O rating 15.2 kN (3,415 lb st). Powered P-2J booster pod, withdrawn from service 1994.
J3 afterburner: Experimental version tested 1972 at 20.2 kN (4,542 lb st).

The J3 was the first Japanese turbojet to go into production. Total deliveries of J3 engines amounted to 247.

An experimental version with augmentation reached a thrust of 20.2 kN (4,542 lb) during bench tests in December 1972. The J3-IHI-7D is an uprated version (15.2 kN, 3,415 lb st), which replaced the -7C in P-2J aircraft.

The following data apply to the J3-IHI-7C:

Type
Axial-flow turbojet.

Intake
Annular nose air intake. Anti-icing system for front support struts.

Compressor
Eight-stage axial-flow type, built of Ni-Cr-Mo steel. Rotor consists of a series of discs and spacers bolted on to shaft. Rotor and stator blades of AISI 403 steel. Stator blades brazed on to fabricated base which is fixed in

casing with circumferential T-groove. Rotor blades dovetailed to discs. Light alloy casing in upper and lower sections, flange-jointed together. Pressure ratio 4.5:1. Air mass flow 25.4 kg (56 lb)/s.

Combustion Chamber
Annular type. AISI 321 steel outer casing. L 605 steel flame tube. Thirty fuel supply pipes located in combustion chamber outer casing and 30 vaporiser tubes located at front of flame tube. Ignition by low-voltage high-energy spark plug in each side of combustion chamber.

Turbine
Single-stage axial-flow type. Disc bolted to shaft. Precision-forced blades. Rotating assembly carried in front (double ball) and rear (roller) compressor rotor bearings and rear (roller) turbine shaft bearing.

Jetpipe
Fixed-area type.

Mounting
Three-point suspension, with one pickup by a pin on starboard side of compressor front casing and a trunion on each side of the compressor rear casing.

Accessories
On gearbox under compressor front casing.

Starting
Electrical starter in intake bullet fairing.

Control System
Hydromechanical, with IHI FC-2 fuel control.

Fuel Specification
JP-4.

Oil System
Forced-feed system for main bearings and gear case. Dry sump. Vane-type positive displacement supply and scavenge pump.

Oil Specification
MIL-L-7808.

Dimensions

Length, less tailpipe	1,661 mm (65.4 in)
Length overall with rear cone	1,994 mm (78.5 in)
Diameter overall	627 mm (24.7 in)
Frontal area	0.28 m^2 (3.01 sq ft)

Weight, Dry

Bare	380 kg (838 lb)
With accessories	430 kg (948 lb)

Performance Ratings

T-O	13.7 kN (3,090 lb st)

Specific Fuel Consumption

At T-O rating	29.74 mg/Ns (1.05 lb/h/lb st)

Oil Consumption

At normal rating (max)	0.60 litres (1.06 Imp pints)/h

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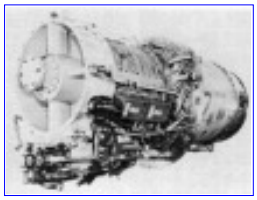

J3-IHI-7C

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AERO-ENGINES - TURBOFAN, JAPAN

Date Posted: 17 December 1999

Jane's Aero-Engines 07

HONDA R&D CO LTD

IHI XF3-400

In 1981 IHI, jointly with the 3rd Research Centre of the Japan Defence Agency TRDI (Technical Research and Development Institute) began development of this advanced derivative of the F3. It is a technology demonstrator. The first complete engine was tested in 1992 and, by March 1995, all planned testing had been successfully completed.

TYPE: Two-shaft turbofan with afterburner.

FAN: Two stages, without inlet guide vanes. Mass flow 34.8 kg (76.72 lb)/s. Pressure ratio 2.7. Bypass ratio 0.9.

COMPRESSOR: Five stages, with variable inlet guide vanes and first stator stage. OPR 14.

COMBUSTION CHAMBER: Short annular with 12 airblast fuel nozzles.

HP TURBINE: Single stage with air-cooled rotor blades.

LP TURBINE: Single stage.

CONTROL SYSTEM: FADEC with hydromechanical back-up.

DIMENSIONS:

Length	2,729 mm (107.44 in)
Inlet diameter	560 mm (22.05 in)
Max diameter	660 mm (25.98 in)
WEIGHT, DRY:	501 kg (1,104.5 lb)

PERFORMANCE RATING (T-O, S/L):

Max dry	21.4 kN (4,811 lb st)
Max afterburner	34.2 kN (7,689 lb st)

SPECIFIC FUEL CONSUMPTION (ratings as above):

Max dry	21.16 mg/Ns (0.75 lb/h/lb st)
Max afterburner	60.9 mg/Ns (2.15 lb/h/lb st)

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AERO-ENGINES - MANUFACTURER, JAPAN

Date Posted: 17 December 1999

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HONDA R&D CO LTD

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SENIOR MANAGING DIRECTOR: Junichi Araki

The Honda R&D Co has for many years studied engines for almost every surface application.In 1986, work began on jet-engine technologies. After much research, in 1991, this work prompted the design of Honda's own HFX-01 turbofan, an experimental two-shaft engine of simple and robust design. Testing of this engine has, in turn, led to the HFX20, the second prototype engine. Honda is studying the industry and market but, in late-1999, had yet to formulate a business plan for its turbofan engines.

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AERO-ENGINES - MANUFACTURER, JAPAN

Date Posted: 17 December 1999

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AERO-ENGINES - TURBOFAN, JAPAN

Date Posted: 17 December 1999

Jane's Aero-Engines 07

NATIONAL AEROSPACE LABORATORY - NAL

MITI/NAL FJR710

In the late 1960s, the Japanese government and industry, seeking an engine programme that might remain competitive for many years, decided to embark on the design of a subsonic turbofan of high bypass ratio. After a preliminary study by the NAL, funding was provided by the Ministry of International Trade and Industry in 1971 for a prototype demonstrator and test programme.

NAL has managed the design of the resulting FJR710. Manufacture of the prototype and development engines was contracted to IHI, Kawasaki and Mitsubishi. The first engine made its first run in May 1973. By the end of 1979 six engines (three FJR710/10 and three FJR710/20 with small changes) had run a total of 1,700 hours.

Phase 2 of the FJR710 programme began in 1976, and the first of three FJR710/600 engines had been completed by December 1978. This version was followed by the lower-rated FJR710/600S, rated at 47 kN (10,582 lb st), which powers the experimental Asuka four-engined QSTOL aircraft. The engine was previously flight-tested powering a Kawasaki C-1. It is still being used in experimental testing.Six 600S engines were built, with total running time of over 7,100 hours.

The following description applies to the /600 engine.

TYPE: Two-shaft high bypass ratio turbofan for subsonic commercial or military aircraft.
INTAKE: Direct annular entry around fan spinner.
FAN: Single-stage fan, with rotating spinner and inserted titanium blades with part-span shrouds. Metal

fan duct held by 10 aerofoil struts, preceded by ring of flow-straightening vanes. Bypass ratio 6.5.
COMPRESSOR: Mechanically independent HP compressor. Twelve-stage axial assembly with inserted blades of titanium and, at delivery end, high nickel alloy. Five rows of variable stator blades held in upper and lower half casings and operated by peripheral rings scheduled by hydraulic ram.
COMBUSTION CHAMBER: Smokeless annular type.
HP TURBINE: Two-stage with cooled blades.
LP TURBINE: Four-stage fan turbine.
JETPIPE: Fixed area.
DIMENSIONS (approx):

Length (flange to flange)	2,350 mm (93 in)
Diameter (inlet)	1,240 mm (49 in)
WEIGHT, DRY:	980 kg (2,160 lb)

PERFORMANCE RATINGS:

T-O	50 kN (11,243 lb st)
Cruise at 7,600 m (25,000 ft) at M0.75	13.24 kN (2,976 lb)

SPECIFIC FUEL CONSUMPTION:

T-O	10.6 mg/Ns (0.374 lb/h/lb st)
Cruise, as above	19.3 mg/Ns (0.680 lb/h/lb st)

FJR710/600S (1996)

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AERO-ENGINES - TURBOFAN, JAPAN

Date Posted: 17 December 1999

Jane's Aero-Engines 07

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Kawasaki has been a famous name in aviation since 1918. In the Second World War, 14,899 aircraft engines were delivered and, between 1942 and 1945, the NE turbojets were tested. In 1954, work resumed with overhaul of USAF J33 and J47 engines.Since 1965, the T53 turboshaft has been produced under licence; deliveries of the T53-K-13B, KT5311A and KT5313B reached 355 in 1992, since when

over 240 T53-K-703 engines have been added. In 1982, the T55-K-712 also went into licence production, deliveries now exceeding 167. KHI was a partner in J79, TF40 (Adour) and JT8D production, and today shares in the manufacture of the T56, F100, F110, TS-1, CF34-8C/D, RB211/Trent, PW4000 and, as a shareholder in IAE, the V2500.KHI also leads the ramjet group in the Ministry of International Trade and Industry research into variable-cycle turbo-ramjet propulsion for a Mach-5 transport. From 1979 to 1984, KHI's own KJ12 turbojet was produced for unmanned vehicles. KHI is a partner in production of AlliedSignal APUs, and produces Marine Olympus, Tyne and Spey engines and industrial gas turbines.

KSX

In recent years KHI has tested a turboshaft of its own design, but has requested its deletion from this product.

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AERO-ENGINES - TURBOFAN, JAPAN

Date Posted: 17 September 1999

Jane's Aero-Engines 06

HONDA R&D CO LTD

HFX20

Having gained experience with the HFX-01, this is Honda's second prototype engine. Its design started in 1997, detail design was completed in 1998, and in 1999 the engine core was on test. Testing of prototype engines is expected to start in 2000 and continue for several years.

HFX20 design objectives are listed as: better fuel economy, superior lapse rate and flat-rate capability, fewer parts and components, on-wing hot-section inspection, optimum location of LRUs and minimum noise. The HFX20 is a two-shaft engine with BPR slightly reduced to 3.9. Other details have not been disclosed, but it can be seen that the mockup has a fan with 18 shroudless blades, and a 16-lobe mixer nozzle.

The following are design targets:

DIMENSIONS:

Length	1,575 mm (62.0 in)
Fan diameter	635 mm (25.0 in)
WEIGHT, DRY:	220 kg (485 lb)
THRUST RATING (T-O, S/L):	9.8 kN (2,205 lb st) at 30ºC

SPECIFIC FUEL CONSUMPTION:

T-O	12.5 mg/Ns (0.44 lb/h/lb st)
Cruise (12,192 m; 40,000 ft at M0.8)	
	21.87 mg/Ns (0.75 lb/h/lb)

Honda HFX20 mockup (1999)

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AERO-ENGINES - TURBOFAN, JAPAN

Date Posted: 17 September 1999

Jane's Aero-Engines 06

HONDA R&D CO LTD

HFX-01

This engine was Honda's first experimental turbofan. It was undertaken in order to gain experience and provide a basis of data. Of simple and robust two-shaft design, with a BPR of 4.3, the HFX-01 was designed, developed and tested almost entirely in-house, including the engine control unit. Core testing began in 1993, the first complete engine completed a 150-hour test in 1994, and high-altitude flight testing was undertaken in 1995-96. The 70-hour flight-test programme was undertaken by Mojave-based AVTEL, using an HFX-01 mounted on the right side of the nose of a 727-100. The HFX-01 research programme was completed in 1997.

DIMENSIONS:

Length	1,180 mm (46.45 in)
Diameter (max)	710 mm (27.95 in)
WEIGHT, DRY:	192 kg (423 lb)
THRUST RATING (T-O, S/L):	8.0 kN (1,800 lb st)
SPECIFIC FUEL CONSUMPTION:	12.5 mg/Ns (0.44 lb/h/lb st)

Honda HFX-01 (1999)

HFX-01 and B-727 testbed (1999)

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AERO-ENGINES - MANUFACTURER, JAPAN

Jane's Aero-Engines 03

NATIONAL AEROSPACE LABORATORY - NAL

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The National Aerospace Laboratory (NAL) is a government establishment responsible for research and development in the field of aeronautical and space science. Since 1962 it has extended its activity to include V/STOL techniques. The decision was made in that year to initiate development of an engine, the JR 100, to fulfil the requirement for a lightweight liftjet for VTOL aircraft. The more advanced NAL/IHI JR200 was developed in 1966; the NAL/IHI JR220 was completed in 1971.

In 1971 the Agency of Industrial Science and Technology, Ministry of International Trade and Industry (MITI), funded a high bypass ratio turbofan engine (FJR710) development programme.

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AERO-ENGINES - MANUFACTURER, POLAND

Date Posted: 22 March 2002

Jane's Aero-Engines 02

PZL RZESZÓW - WYTWÓRNIA SPRZETU KOMUNIKACYJNEGO-``PZL RZESZÓW'', SA

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President, General Director: Tadeusz Cebulak
Vice-President, Commercial Director: Wladyslaw Jasiczek
Vice-President, Development Director: Ryszard Legiewicz
Business Development Manager: Andrzej Klimiec

The company was established in 1937 and, in 1994, was reorganised into a joint stock company. WSK PZL-Rzeszów employs some 5,000 people. Production includes turbine and piston engines, helicopter transmissions and a wide range of components for the world's major engine companies.

Current production at PZL Rzeszów is centred on the PZL-10W with the WR-3 reduction gearbox for the PZL Sokól, the K-15 turbojet for the I-22 Iryda and PZL-Franklin piston engines.

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AERO-ENGINES - TURBOSHAFT, POLAND

Date Posted: 01 May 2001

Jane's Aero-Engines 10

WYTWÓRNIA SPRZETU KOMUNIKACYJNEGO-``PZL RZESZÓW'', SA

PZL-10W

This turboshaft engine, designed by WSK-PZL Rzeszów on the TWD-10B core, is certified in Poland, Russia, the USA and Germany. It powers the twin-engined W-3 Sokól helicopter manufactured by WSK-PZL Swidnik.

PZL-10W2

Upgraded version with T-O rating of 736 kW (986 shp). To be available in 2000.

The following description relates to the basic PZL-10W. For the gas generator description see the TWD-10B entry.

Type

Free-turbine turboshaft.

Power Turbine

Single-stage axial, 23,615 rpm.

Accessories

Pads driven by compressor for starter, fuel-metering unit, tachogenerator and oil pumps. Pad on power-turbine casing for hydromechanical power turbine governor.

Control System

Hydroelectronic with electronic as a primary system for power turbine management. System maintains constant selected helicopter rotor speed for engine speeds from flight idle to T-O, prevents compressor and power-turbine speed and gas temperature from exceeding maximum values, controls anti-surge bleed-off valve, shuts down the engine in the case of power-turbine overspeed, and maintains preprogrammed output-shaft torque.

Fuel Specification

T-1, T-2, TS-1, RT, PSM-2, Jet A-1.

Oil System

Pressure type with one pressure and four scavenge gear pumps. Oil tank and cooler airframe mounted.

Oil Specification

B-3W, Castrol 599, Castrol 5000, Castrol 5050, ASTO 500, ASTO 555, Elf Turbo Jet II.

Dimensions

Length with exhaust pipe	1,875 mm (73.8 in)
Width: left engine	740 mm (29.0 in)
right engine	765 mm (30.1 in)

Weight

Dry	141 kg (310 lb)

Performance Ratings

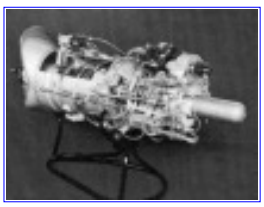
2.5 min	846 kW (1,134 shp)
30 min	736 kW (986 shp)
T-O	622 kW (888 shp)
Continuous	574 kW (769 shp)

Specific Fuel Consumption

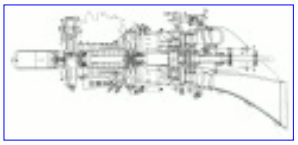
T-O	101.1 µg/J (0.60 lb/h/shp)

UPDATED

Two views of a PZL-10W cut away for display purposes



Two views of a PZL-10W cut away for display purposes



Longitudinal section through PZL-10W turboshaft (2001)

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AERO-ENGINES - TURBOPROP, POLAND

Date Posted: 12 April 2001

Jane's Aero-Engines 10

WYTWÓRNIA SPRZETU KOMUNIKACYJNEGO-``PZL RZESZÓW'', SA

TWD-10B

This turboprop engine, originally rated at 706 kW (947 shp), designed as the TVD-10B by the former Soviet Glushenkov Construction Bureau, was produced under licence by WSK-PZL Rzeszów for the An-28 STOL light transport aircraft built by WSK-PZL Mielec. This is the version described below:

PZL-10S

Redesigned to work with Hartzell five-blade propeller. Flight tested in 1999.

Type

Free-turbine turboprop.

Intake

Three radial struts, inlet guide vanes, centrally mounted starter. Hot-air ant-icing system.

Compressor

Six axial stages and one centrifugal. Blades, vanes, shaft and front and rear discs made of steel; casing, impeller and remaining discs made of titanium. Pressure ratio 7.4. Mass flow 4.6 kg (10.14 lb)/s.

Combustion Chamber

Annular with centrifugal fuel injection. Two starting units with starting fuel injectors.

Compressor Turbine

Two-stage axial with cooled nozzle. Casing with ceramic inserts. T-O rpm 29,600. TGT 887ºC.

Power Turbine

Single-stage axial.

Output

High-speed gearbox transmits power from the turbine shaft, through the intermediate shaft, to the propeller gearbox at the front of the engine. Total reduction ratio 13.387:1. Matched with 2.8 m (9 ft 2¼ in) AW-24AN or AV-25B propeller, turning at 1,800 rpm (T-O), 1,620 rpm (cruise).

Accessories

Pads driven by compressor for fuel pressure pump, pump governor, oil pumps and tachogenerator. Pads driven by power turbine for oil pumps, generator, propeller brake, tachogenerator and propeller governor.

Control System

Hydromechanical with electronic control of maximum gas temperature and power-turbine overspeed protection.

Fuel Specification

T-1, T-2, TS-1, PSM-2 and Jet A-1.

Oil System

Pressure type with separate gear pump units for gas generator and power turbine. Oil tank capacity 16 litres (4.2 US gallons; 3.5 Imp gallons).

Oil Specification

Oil mixture: 25 per cent of MK-22 or MS-20 oil and 75 per cent of MK-8, MK-8P or MS-8P oil.

Dimensions

Length with exhaust pipe	2,060 mm (81.1 in)
Width	555 mm (21.9 in)
Height	900 mm (35.4 in)

Weight, Dry

Basic	225 kg (496 lb)
Complete with all accessories	300 kg (661 lb)

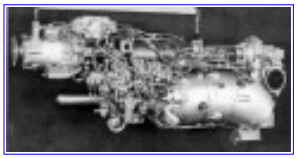
Performance Ratings

T-O	754 ekW (1,011 ehp)
Nominal	613 ekW (823 ehp)
Max cruise	547 ekW (734 ehp)

Specific Fuel Consumption

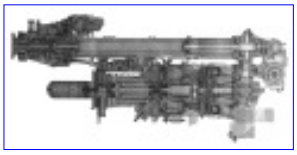
T-O	95.1 µg/J (0.564 lb/h/ehp)

UPDATED



TWD-10B

TWD-10B with inlet and jetpipe



Longitudinal section through TWD-10B (2001)

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AERO-ENGINES - TURBOSHAFT, POLAND

Date Posted: 17 December 1999

Jane's Aero-Engines 07

WYTWÓRNIA SPRZETU KOMUNIKACYJNEGO-``PZL RZESZÓW'', SA

GTD-350

This popular turboshaft engine is used on the Mi-2 helicopter produced by WSK-PZL Swidnik. Since 1966, more than 19,000 engines were built. Recently an uprated version of the engine was developed by PZL Rzeszów - the GTD-350W of 313 kW (419 shp) T-O power. The following describes the original version, which is mechanically identical.

TYPE: Free turbine turboshaft.

INTAKE: Automatic de-icing of inlet guide vanes and bullet by hot air bleed.

COMPRESSOR: Seven axial stages and one centrifugal, all of steel. Pressure ratio 6.05. Mass flow 2.19 kg (4.83 lb)/s at 45,000 rpm.

COMBUSTION CHAMBER: Reverse-flow type with air supply through two tubes. Duplex single-nozzle burner. Semiconductor igniter plug.

COMPRESSOR TURBINE: Single-stage. Shrouded blades with fir-tree roots.Temperature before turbine 970ºC (GTD-350W, 985ºC).

POWER TURBINE: Two-stage constant-speed (24,000 rpm). Shrouded blades with fir-tree roots. Discs bolted together. Turbine stators integrally cast.

JETPIPE: Twin pipes, above and below compressor delivery, handed to left or right.

OUTPUT: Two sets of gears, with ratio of 0.246:1, in magnesium alloy casing. Output speed 5,904 rpm.
ACCESSORIES: STG3 3 kW starter/generator, NR-40TA governor pump, D1 tachometer and oil pumps driven by gas generator. RO-40TA speed governor, D1 tachometer and centrifugal breather driven by power turbine.
STARTING: STG3 starter/generator suitable for operation at up to 4,000 m (13,125 ft) altitude.
CONTROL SYSTEM: Hydromechanical, with NR-40TA pump governor; RO-40TA power turbine governor; DS-40 controlling bleed valves; and electromagnetic starting valve.
FUEL SPECIFICATION: TS-1, TS-2 or Jet A-1.
OIL SYSTEM: Closed type. Gear-type pump with one pressure and four scavenge units. Cooler and tank, capacity 12.5 litres (3.30 US gallons; 2.75 Imp gallons).
OIL SPECIFICATION: B3-W (synthetic), Castrol 98 or 5000, Elf Turbojet II or Shell Turbine Oil-500.
DIMENSIONS:

Length overall	1,385 mm (54.53 in)
Max width	520 mm (20.47 in)
Width, with jetpipes	626 mm (24.65 in)
Max height	760 mm (29.9 in)
Height, with jetpipes	630 mm (24.80 in)

WEIGHT, DRY:

Less jetpipes and accessories	139.5 kg (307 lb)

PERFORMANCE RATINGS:
T-O rating (6 min) at 96% max gas generator rpm

	294 kW (394 shp)

Nominal rating (1 h) at 90% gas generator rpm

	236 kW (316 shp)

Cruise rating (I) at 87.5% gas generator rpm

	210 kW (281 shp)

Cruise rating (II) at 84.5% gas generator rpm

	173 kW (232 shp)

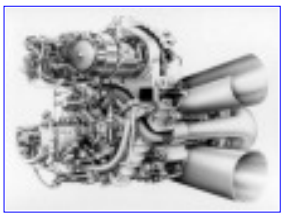
SPECIFIC FUEL CONSUMPTION:

T-O	142 µg/J (0.84 lb/h/shp)
Nominal	153 µg/J (0.91 lb/h/shp)
Cruise (I)	162 µg/J (0.96 lb/h/shp)
Cruise (II)	173 µg/J (1.02 lb/h/shp)

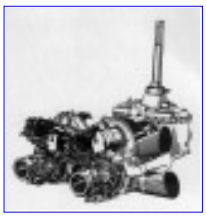
OIL CONSUMPTION:

Max	0.3 litre (0.63 US pint; 0.53 Imp pint)/h

GTD-350 (1996)



Cutaway GTD-350 (1996)



Drawing of the Mi-2 installation of twin GTD-350 engines and reduction gearbox (1996)

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AERO-ENGINES - TURBOJET, POLAND

Date Posted: 17 December 1999

Jane's Aero-Engines 07

WYTWÓRNIA SPRZETU KOMUNIKACYJNEGO-``PZL RZESZÓW'', SA

K-15

This turbojet engine was designed by Instytut Lotnictwa and WSK-PZL Rzeszów for the I-22 Iryda military advanced trainer. Since 1994, it has been in series production at WSK-PZL Rzeszów, which is also responsible for support of the engine and for its further development as the K-16. For description of the engine see the IL K-15 entry under Instytut Lotnictwa.

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AERO-ENGINES - TURBOJET, POLAND

Jane's Aero-Engines 03

INSTYTUT LOTNICTWA (Aviation Institute) - IL

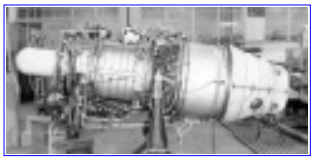
IL SO-3

This improved version of the SO-1 replaced the earlier type in production at the WSK-PZL Rzeszów. The SO-3 is intended for tropical use and incorporates minor changes in compressor, combustion chamber and turbine, data remaining the same as for the SO-1. It is fitted to all Indian TS-11 aircraft. Since 1978 the SO-3B has been developed and qualified and this is now the standard TS-11 engine, with TBO of 400 h. The compressor is entirely steel, and a revised vaporising burner and flame tube result in more uniform gas temperature entering the turbine. Data are as for the SO-1 except:
OIL SPECIFICATION: AW-30 synthetic.

WEIGHT, DRY:	321 kg (708 lb)

PERFORMANCE RATINGS:

T-O	10.8 kN (2,425 lb st) at 15,600 rpm
Max continuous	9.8 kN (2,205 lb st) at 15,100 rpm
OIL CONSUMPTION:	1.0-1.2 litres (1.7-2.1 Imp pints)/h



SO-3 (1996)

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AERO-ENGINES - TURBOJET, POLAND

Jane's Aero-Engines 03

INSTYTUT LOTNICTWA (Aviation Institute) - IL

IL SO-1

The Aviation Institute designed the SO-1 turbojet to power the Polish TS-11 Iskra (Spark) jet basic trainer. It was designed to permit the full range of aerobatics, including inverted flight. Guaranteed overhaul life was initially 200 hours. Production was handled by the WSK-PZL Rzeszów.
TYPE: Single-shaft axial-flow turbojet.
INTAKE: Annular intake casing manufactured as a cast shell. Fixed inlet guide vanes.
COMPRESSOR: Seven-stage axial-flow compressor. Drum type rotor built up of disc assemblies, with constant diameter over tips of rotor blades. Carried in ball bearing at front and roller bearing at rear. Steel stator blades bonded with resinous compound into slots in carrier rings. Casing manufactured as a cast shell in two parts, split along horizontal centreline, in aluminium alloy. Rotor originally of steel and duralumin, with first three blade rows of steel and remainder of aluminium alloy. Modified as a result of operating experience; entire compressor rotor and blades on all stages now made of steel. Pressure ratio 4.8.
COMBUSTION CHAMBER: Annular type with 24 integral vaporisers. Outer casing made of welded steel.
TURBINE: Single-stage axial-flow type. Blades attached to disc by fir-tree roots. Supported in roller bearing at front.
JETPIPE: Outer tapered casing and central cone connected by streamlined struts. Nozzle area adjusted by exchangeable inserts.
ACCESSORIES: Gearbox mounted at bottom of air intake casing and driven by bevel gear shaft from

front of compressor.
STARTING: 27 V starter/generator and bevel gear shaft, by aircraft battery or ground power unit, mounted on air intake casing.
CONTROL SYSTEM: Two independent systems supplied by one pump. Starting system consists of six injectors, with direct injection. Main system consists of 12 twin injectors with outlets towards the vaporisers.
OIL SYSTEM: Open type for rear compressor and turbine bearings, supplied by separate pumps. Closed type for all other lubrication points, fed by separate pumps.
OIL SPECIFICATION: Type AP-26 (synthetic).
DIMENSIONS:

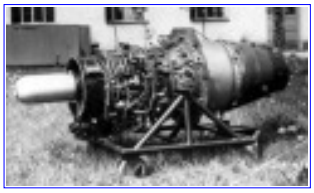
Length overall	2,151 mm (84.7 in)
Width	707 mm (27.8 in)
Height	764 mm (30.1 in)
WEIGHT, DRY:	303 kg (668 lb)

PERFORMANCE RATINGS:

T-O	9.8 kN (2,205 lb st) at 15,600 rpm
Max continuous	8.7 kN (1,958 lb st) at 15,100 rpm

SPECIFIC FUEL CONSUMPTION:

At T-O rating	29.6 mg/Ns (1.045 lb/h/lb st)
OIL CONSUMPTION	0.8 litres (1.4 Imp pints)/h



SO-1 (1996)

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AERO-ENGINES - TURBOJET, POLAND

Jane's Aero-Engines 03

INSTYTUT LOTNICTWA (Aviation Institute) - IL

IL K-15

This turbojet was announced in Summer 1988. It was developed to power the production I-22 Iryda twin-jet trainer. It features numerous borescope ports and parametric measures to permit on-condition operation. Series engines are produced by PZL-Rzeszów.
TYPE: Single-shaft turbojet.
INTAKE: Simple forward-facing pitot-type in cast aluminium, with three anti-iced struts.
COMPRESSOR: Six axial stages. Rotor EB-welded in maraging steel, with inserted blades (stages 1 to 3) titanium, (stages 4 to 6 and shrouded stator vanes) stainless steel. Casing light alloy, with two blow-off valves. Mass flow 23.5 kg (51.8 lb)/s. Pressure ratio 5.3.
COMBUSTION CHAMBER: Short annular type welded from machined rings. 18 vaporising burners, six starting atomisers and two high-energy igniters.
TURBINE: Single-stage. Disc of H46 steel, attached via Hirth coupling, carrying forged ES 867 blades. Vanes cast in ZS6K. TET 870ºC.
JETPIPE: Simple fixed-area.
ACCESSORIES: Gearbox at bottom of intake casing driven by spur gear from front of main shaft. Pads for engine fuel and oil pumps, aircraft hydraulic pump and tachogenerator.
STARTING: 27-V 9-kW starter/generator in nose bullet.
CONTROL SYSTEM: PZL Hydral fuel control of hydromechanical type, with overspeed and overtemperatrue limiters and electronic control of blow-off valves.
FUEL SPECIFICATION: Kerosene PSM-2 or (Russian) TS-1.

OIL SYSTEM: Fully aerobatic self-contained recirculatory, except for total-loss supply to rear main bearing.
OIL SPECIFICATION: Type SDF synthetic.
DIMENSIONS:

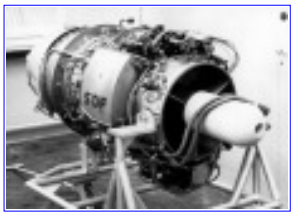
Length overall	2,006 mm (78.98 in)
Width	725 mm (28.54 in)
Height	892 mm (35.12 in)
WEIGHT, DRY:	340 kg (750 lb)

PERFORMANCE RATINGS:

T-O (S/L)	14.7 kN (3,307 lb st) at 15,900 rpm
Max continuous (S/L)	11.5 kN (2,585 lb st) at 15,025 rpm
Cruise (6,000 m; 19,685 ft, M 0.6)	4.14 kN (930.6 lb)

SPECIFIC FUEL CONSUMPTION:

T-O, as above	28.49 mg/Ns (1.006 lb/h/lb st)
Cruise, as above	32.62 mg/Ns (1.152 lb/h/lb)



IL K-15 (1996)

IL K-15 (1996)

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AERO-ENGINES - TURBOFAN, POLAND

Jane's Aero-Engines 03

INSTYTUT LOTNICTWA (Aviation Institute) - IL

IL D-18A

This completely new engine was first run on 16 April 1992.
TYPE: Two-shaft turbofan.
AIR INTAKE: Direct pitot intake without inlet guide vanes.
FAN: Two-stage axial, with steel blades and stators. EB-welded steel rotor carried in ball and roller bearings. Mass flow 38.4 kg (84.66 lb)/s. Pressure ratio 2.07. Bypass ratio 0.7.
COMPRESSOR: Five-stage compressor on HP shaft. Stainless steel blades and stators. Rotor consists of two parts bolted together, both EB-welded, connected with HP turbine by Hirth coupling. Overall pressure ratio 8.
COMBUSTION CHAMBER: Annular type with 18 integral vaporisers, six starting atomisers and two high-energy igniters.
HP TURBINE: Single-stage carried in roller bearings. Forged ES 867 blades, cast ZS6K vanes, Hirth coupling. TGT 900ºC.
LP TURBINE: Single-stage.
JETPIPE: Plain fixed convergent nozzles for both core gas and bypass flow.
MOUNTING: Two main pads on intermediate casing. One rear strut on either side of centreline.
ACCESSORIES: Accessory gearbox driven by power offtake from front of HP shaft. Drives engine fuel and oil pump, starter/generator and aircraft hydraulic pump.
STARTING: 9 kW (12 hp), 27 V starter-generator supplied by aircraft battery or ground power unit.
CONTROL SYSTEM: Full-authority digital electronic control.

OIL SYSTEM: Integral oil system with vane pumps. Oil/fuel heat exchanger.
OIL SPECIFICATION: Synthetic, type SDF.
DIMENSIONS:

Length	1,940 mm (76.37 in)
Inlet diameter	590 mm (21.25 in)
Width	750 mm (29.52 in)
Height	900 mm (35.43 in)
WEIGHT, DRY:	380 kg (837.7 lb)

PERFORMANCE RATINGS:

T-O	17.65 kN (3,968 lb st) at 17,500 rpm

SPECIFIC FUEL CONSUMPTION:

At T-O rating	20.96 mg/Ns (0.74 lb/h/lb st)

IL D-18A (1996)

IL D-18A (1996)

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AERO-ENGINES - MANUFACTURER, POLAND

Jane's Aero-Engines 03

INSTYTUT LOTNICTWA (Aviation Institute) - IL

MANUFACTURER DETAILS

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MANAGING DIRECTOR: Witold Wismowski
CHIEF CONSULTANT FOR SCIENTIFIC AND TECHNICAL CO-OPERATION: Jerzy Grzegorzewski, MSc Eng

The Aviation Institute is concerned with aeronautical research and testing. It can construct prototypes to its own design.

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AERO-ENGINES - MANUFACTURER, ROMANIA

Date Posted: 17 August 2000

Jane's Aero-Engines 08

TURBOMECANICA - INTREPRINDEREA TURBOMECANICA BUCURESTI

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e-mail: turbo@dial.kappa.ro

GENERAL DIRECTOR: Ioan Serban Ciorapciu
MARKETING TEAM LEADER: Stefan Frangulea

Founded in 1975, this factory produces or supports under licence the Rolls-Royce Viper 632-41 and 633-47 and Spey 512-14DW, Turbomeca Turmo IVC, and helicopter gearboxes and rotor heads. Turbomecanica had 1,020 employees in December 1999.

UPDATED

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 18 April 2002

Jane's Aero-Engines 12

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

PS-9

In 2001 OAO 'Aviadvigatel' and PMZ, Perm Motors, began collaboration on a new family of engines for transport aircraft based on a common core. The partners, in effect on the same geographical site, were able to do this mainly because of US funding and assistance by Pratt & Whitney originally directed at the troubled PS-90.

Little has so far been published regarding this new family of engines, beyond what follows:

PS-7

Reduced-thrust derivative of baseline engine, for business jets, such as the Tu-414. Two-stage LP turbine driving fan and two-stage LP compressor. Fitted with target-type reverser. T-O rating 68.64 kN (15,432 lb st), cruise thrust (11,000 m, 36,089 ft, M0.8) 15.69 kN (3,527 lb), cruise sfc 17.7 mg/Ns (0.625 lb/h/lb).

PS-9

Baseline engine, described briefly below. Intended for such aircraft as the Be-200, Il-214 and Tu-334.

PS-14

For aircraft such as the Tu-234 and Yak-242. Significantly larger fan, and five-stage LP compressor, driven by four-stage LP turbine. Reverser of cascade type, with translating cowl. T-O rating 137.29 kN (30,864 lb st), cruise thrust (11,000 m, 36,089 ft, M0.8) 25.00 kN (5,622 lb), cruise sfc 15.43 mg/Ns (0.545 lb/h/lb).

PS-14R

Development of PS-14 with geared fan. Three-stage LP turbine driving fan gearbox and three-stage LP compressor. Intended for similar applications. T-O rating 137.29 kN (30,864 lb st), cruise thrust (same conditions) 25.49 kN (5,732 lb), cruise sfc 15.175 mg/Ns (0.536 lb/h/lb).

PS-18R

Further developed project with yet larger fan and four-stage LP compressor geared down from redesigned three-stage LP turbine. T-O rating 176.52 kN (39,683 lb st), cruise thrust (conditions as before) 31.37 kN (7,055 lb), cruise sfc 14.72 mg/Ns (0.520 lb/h/lb).

Brief details of PS-9:

Type

Two-shaft turbofan.

Fan

Single stage.

LP Compressor

Four stages.

HP Compressor

Five stages, first stage having variable inlet guide vanes.

Combustion Chamber

Annular.

HP Turbine

Single stage, with aircooled blades.

LP Turbine

Three stages.

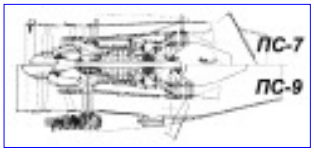
Performance Rating

T-O, S/L	100.0 kN (22,487 lb st)
Cruise, conditions as above	21.08 kN (4,740 lb)

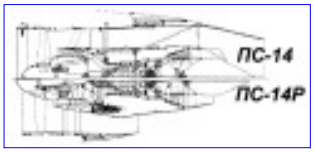
Specific Fuel Consumption

Cruise, conditions as above	16.96 mg/Ns (0.599 lb/h/lb)

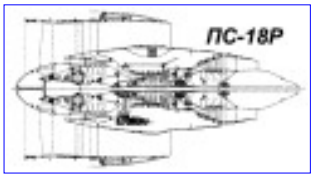
NEW ENTRY



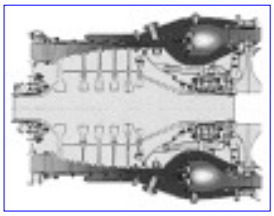
Cross-sections of PS-7 (upper) and PS-9 (2002)



Cross-sections of PS-14 (upper) and PS-14R (2002)



Cross-section of PS-18R (2002)



Cross-section of common core (2002)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 18 April 2002

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`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

PS-90A

This high-bypass ratio turbofan was not derived from any existing engine. It is assembled from 11 modules, and is designed for long life, high reliability and low fuel burn. Originally known as the D-90, this engine is the first to have a designation reflecting the name of the General Designer, Pavel Solovyov.

PS-90A

Baseline version, to which detailed description below applies. Bench testing began in 1984. Flight testing began in 1987 with an engine replacing the starboard inner D-30KP in an Il-76. At this time the D-90 was competing against the NK-93; though less powerful, it had better fuel economy and appeared to be cheaper to maintain, and it was chosen to power the Il-96.

Certification was completed in 1991, the certificate 16D being received in April 1992. The PS-90A powers the Il-96-300, first flown on 28 September 1988, and the Tu-204, first flown on 2 January 1989. It has also been selected for the prototype Tu-330 airlift transport, in which application it has an OEI emergency rating of 171.6 (38,580 lb st). The Il-96-300 was certified in December 1992 and the Tu-204 in December 1994.

Modifications effected during development included modifying blade profiles and adding a third stage to the LP compressor, a new combustor, an improved HP turbine without an intershaft bearing, and a heated nose spinner. For reliability reasons the variable HP stators have been changed from hydraulic to pneumatic operation. Many parts, including the nose spinner and titanium honeycomb nozzle, have been changed to composite material.

The high profile of the PS-90, and continued problems, led to a decree by President Boris Yeltsin in May 1995 to guarantee investment for a recovery programme by Aviadvigatel, Perm Motors, Tupolev, Aviastar (Ulyanovsk) aircraft plant, Universal (owner of Orel-Avia) and Promstroi Bank. Palliatives include reducing TET, reducing coking and debugging the software. GE/SNECMA have had a long association with the engine, which may be ended by a major investment (reported as being up to US$150 million) by Pratt & Whitney which, with MTU, almost redesigned the entire LP system. The redesign features a one-piece (instead of three-part) shaft, new fan, new LP core booster with four stages instead of two, and rebladed HP and LP turbines. This effort must inevitably impact on Pratt's own PW2337, also flying on the Il-96M.

In 1998-99, Reshetnikov's team at Aviadvigatel was working with Perm Motors and Pratt & Whitney in a US$30 million joint programme to upgrade the PS-90A family of engines. Although these have a satisfactory record of six years in service, the number delivered by late-1999 being just over 200, they are financially uncompetitive with Western engines because of their need for time-dependent maintenance and parts replacement. In 1999, overhaul and parts lives were still in the range of 1,000 to 5,000 hours, and when one engine remained on-wing for 5,370 hours this set a Russian record.

In early 2001 it was flatly stated that ``the PS-90A is the only Russian engine fully certificated to ICAO noise and emissions standards''. It was explained that Russian certification stipulates TBOs based on achieved results. In the ``first year of use''. PS-90A engines were changed ``every 500 to 700 hours''. In November 1995 TBO was 3,000 hours, in January 1996 it was raised to 5,000, and since 1997 the engine has been operated on an on-condition basis. In February 2001 the average on-wing life was ``nearly 5,000 hours, but one engine has operated for 7,640 hours and another for almost 7,000''.

In February 2001 Viktor Samokhin, Deputy Manager for Civil Aviation Engineering and Development at the Federal Ministry of Transport, said: ``A programme for improving PS-90A reliability and further development has been approved by the Ministry of Defence and Rosaviakosmos''. At that time 180 PS-90A engines of all types had been actually delivered, and these were then installed on 14 Il-96-300 (of which ten were flying), the Il-76MF prototype, 21 Tu-204 (six flying), and the Tu-214 prototype. Total flight time had reached 511,000 hours in October 2000, but has climbed fairly slowly since.

The Perm 'Aviadvigatel' enterprise is using ``adaptable forms of sale for PS-90A engines. In the majority of cases delivery is paid for over time, often in accordance with the number of operating hours''.

The PS-90A core is the basis of engines for surface applications. The first of these are the GTU-12P and GTU-16P gas-pumping engines, for which Perm Motors in 1998 signed a contract with RAO Gazprom.

Type

Two-shaft turbofan with mixer and fan reverser.

Fan

Single-stage, with 33 titanium blades, with snubbers. Hub/tip ratio 0.34. Mass flow 470 kg (1,036 lb)/s. Pressure ratio 1.747. Bypass ratio (cruise) 4.27.

LP Compressor

Two-stage booster bolted to rear of fan.

HP Compressor

Thirteen-stage spool with variable inlet guide vanes and first two stators. Overall pressure ratio (cruise) 35.55. Speed (maximum) 12,100 rpm.

Combustion Chamber

Can-annular with 12 flame tubes with duplex burners and two igniters.

HP Turbine

Two stages, with advanced blades cooled by air passed through cold heat exchanger. Entry gas temperature 1,640K (1,367ºC).

LP Turbine

Four stages with radial clearance control. Maximum gas temperature 898 K (625ºC).

Jetpipe

Mixer combines core and bypass flows to single nozzle.

Reverser

Multiple blocker doors close off fan duct as translating mid-section of cowl moves to rear, to uncover all-round reverser cascades. No core reverser.

Control System

Two-channel electronic, with hydromechanical back-up.

Lubrication

Closed Loop.

Starting

Pneumatic, air starter from ground supply or cross-bleed.

Dimensions

Fan diameter	1,900 mm (74.8 in)
Length overall	4,964 mm (195.4 in)

Weight, Dry

	2,950 kg (6,503 lb)

Performance Ratings

(ISA)

T-O, S/L	156.9 kN (35,275 lb st) to 30ºC

Cruise at 11,000 m (36,090 ft) and M0.8	34.32 kN (7,716 lb)

Specific Fuel Consumption

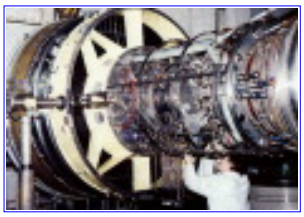
Cruise, as above	17.1 mg/Ns (0.604 lb/h/lb)

Contract Price

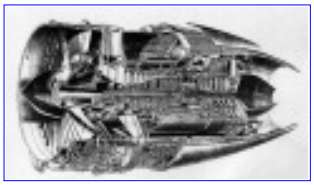
Said in February 2001 to be ``nearly US$2 million".

VERIFIED

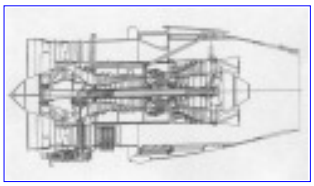
PS-90A



PS-90A in service (2001)



Cutaway drawing of PS-90A (upper half showing reverser in operation)



Longitudinal section through PS-90A (upper half showing reverser in operation)

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SALUT - MMPP (MOSCOW MACHINE-BUILDING PRODUCTION PLANT) SALUT

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General Director: Yury S Yeliseev
Chief of Marketing Bureau: Semenov Alexander

In 1912, this large production plant was the first in Russia to undertake series production of aircraft engines. In the Soviet Union it became Factory No 24, named for M V Frunze. It was allocated the production of air-cooled radial engines designed by A D Shvetsov, and from 1932, the large water-cooled V-12 engines of the AM family designed by A A Mikulin. After the Second World War the factory was renumbered No 45, and as well as completing piston-engine production contracts it also undertook the manufacture of the early experimental turbojets designed by A M Lyul'ka. This established a link with 'AL' designs which has grown over the years. In 1947 V Ya Klimov and N G Metskhvarishvili arrived to oversee mass-production of the Rolls-Royce Nene, which was designated RD-45 after the factory. They went on to manage the production of the derived VK-1 family, described under Soyuz. In 1957 this work was complete, and No 45 OKB was headed by Eduard Eduardovich Luss, formerly First Deputy to Lyul'ka, and since then Salut has mainly produced Lyul'ka engines (though it made an important contribution to Tumanskiy's R-15, see under Soyuz).

In 2000, Salut became commercially linked with Ivchenko Progress, Motor Sich and UMPO. It is in

production with Lyul'ka AL-31F and AL-31FP and tooling to produce the AL-55. It is also producing major parts of the D-436T and D-27 (for these see Ivchenko Progress, Ukraine) and the GTE-25U gas turbine for surface applications. Salut also produces spares for all these engines and for the AL-21 and the Soyuz R-15B-300 family.

In 2001 it was building up its own design office. The immediate objective of Salut designers is to improve the AL-31F and create derivatives with increased thrust. The name means `Salute'. The Granit Machine-Building Design Bureau is adjacent.

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 22 March 2002

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UMPO - UFA ENGINE INDUSTRIAL ASSOCIATION JSC

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General Director: Valeri P Lesunov

This factory, which UMPO claims is the largest in Russia, was established in 1925. Named *Rossiya* (Russia), it has mass-produced the AL-31F and AL-31FP, R-13, R-25, R-29B, R-35F, R-95Sh and R-195 designed by Lyul'ka Saturn and Soyuz. It has also produced gearboxes and transmissions for the Mi-26 and all Kamov turbine-engined helicopters, and a wide range of other items. In 1999, it had completed tooling to participate in series production of the D-436T1 and D-436TP (see under Ivchenko Progress, Ukraine). UMPO is participating in the development of the ZMKB progress D-27 propfan, and in May 2001 it was announced that UMPO will also share in manufacture of the Lyul'ka Saturn AL-41F. UMPO is a partner in a group which, in addition to Ivchenko Progress, includes Motor Sich, Klimov and ZMKB.

UMPO also produces the AL-31ST gas-pumping engine and a wide range of other products including engines for cars, snowmobiles and water scooters.

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

LYUL'KA SATURN INC

AL-41

Development of this completely new fighter engine was launched in 1985. At the 1991 Paris air show, the Editor was given an off-the-record briefing on it by Dr Viktor Chepkin, who was most enthusiastic about its potential. At that time, the prototype had run on the testbed, more than fulfilling expectations. Dr Chepkin said the Saturn design bureau intended to introduce `SAT' designations, the new engine being the SAT-41. In the event, the traditional 'AL' designation has been retained.

The collapse of the Soviet Union virtually halted funding, and work on the initial application, the twin-engined Mikoyan MFI (1-44), which was essentially complete in 1991, likewise ground to a stop. Little could be done until in 1997 a trickle of funds became available from the export of Su-27 and related fighters. Then, after years of discussion, on 17 September 1998 the engine, by now designated AL-41F, was promised firm development funding.

Part of the money will be found by the Russian Ministry of Defence, but - in what appears to be a new development - the major part, said to be approximately Rb6 billion (US$354 million), has been promised by the Moscow regional government. The agreement was signed by the Mayor of Moscow, Yurii Luzhkov, and the Governor of Yaroslavl region, Anatoly Lisitsyn. The 1-44 was at last publicly revealed on 12 January 1999. Whether or not full funding is made available, the AL-41F is reported to have a firm application in the series version of Su-32FN. Sukhoi, long reliant on Lyul'ka engines, also originally selected a version of this engine for the S-37 (not to be confused with the Su-37), but the only example of this aircraft so far built has D-30F6M engines.

By 1999, despite the lack of funding, AL-41 and augmented AL-41F prototype and development engines had run many hundreds of hours at the design bureau and at CIAM. Flight-cleared engines have

been air-tested in aircraft of the LII flight-test institute. Subsonic testing has been performed with the engine in a nacelle under a Tu-16, while testing up to Mach numbers in excess of 2.5 have been carried out with a MiG-25. In mid-1999 little more could be done until the start of MFI flight testing. At last, the flight-test programme of this aircraft opened on 29 February 2000.

On 20 April 2001 Rybinsk Motors stockholders voted to merge with Lyul'ka Saturn, and a week later 95 per cent of the Lyul'ka Saturn stockholders voted to endorse such a merger. The principal reason for the merger was to provide an organization strong enough to put the AL-41 into production. As it has been estimated this will require 'US$1 billion' the prospects still look bleak (in 2001 Lyul'ka Saturn's electricity was cut off for non-payment). Sources of income were listed as Russian State payments for defence items, plus income from sales to other countries, notably China and India.

In June 2001 Mr Chepkin, together with Yuri Lastochkin of Rybinsk and Valeri Lesunov of UMPO, signed a protocol of the intention to create a corporation to complete the development of 'fifth-generation engines' (the AL-41 family) and put them into production. The three organizations planned to unite in every aspect of the effort through to after-sales service. It was also announced that they would work on two versions of the fifth-generation engine, the **AL-41F-1** with vectored thrust, and a later engine (presumably with a fixed nozzle), the **AL-41F-2**.

The AL-41F has from the outset been designed for dry supersonic cruise for more than 1 hour, with augmentation reserved for acceleration or extreme manoeuvres. In 1999 the Editor was told that, if funding was maintained, the engine was planned to be ready for production in 2003-04, but the latest (and probably optimistic) estimate is 2006. Apart from the Su-32FN and a possible derivative of the MFI prototype, other applications for this outstanding engine might include the Su-27IB (Su-34), Su-30, Su-33 and Su-37.

For the longer term it is hoped to use it to power a single-engined light multirole fighter to compete with the American JSF. Such an aircraft is (again optimistically) planned to be built in 2004-05. To match the requirements of such an aircraft it was unofficially believed in Moscow in late 2000 that an uprated AL-41F would be needed, with an augmented thrust of no less than 254 kN (57,095 lb st). This would make possible a single-engined aircraft with a take-off weight of 19 tonnes (normal, with AAMs and a tank) or 27 tonnes (maximum).

In September 1998 the Defence Ministry announced that production AL-41F engines would be made by Rybinsk Motors. Later Salyut argued that the contract should go to their factory, or at least be shared, partly because `Rybinsk has no experience of vectored nozzles, which are crucial components.

From the outset, the AL-41 family of engines has been designed to be fully modular and to make the fullest use of the most advanced available materials. It also incorporates the most advanced aerodynamics, with turbomachinery significantly surpassing any previous fighter engine of which details were known in the mid-1980s. Particular attention has been paid to using the largest possible rotating blading and the lowest possible parts count. Numerical details were still classified in early 2000, but Dr Chepkin has said that, compared with the AL-31F, the AL-41F has `significantly greater thrust, a specific weight 20 per cent less, significantly reduced RCS (Radar Cross Section) and life-cycle costs estimated to be 25 per cent lower'.

The same turbomachinery is being developed as the core of a commercial turbofan family (thrust `16 to 40 tonnes', 156.8 to 392 kN, 35,270 to 88,180 lb st) and shaft-drive versions for gas pumping and possibly marine propulsion. The agreement with Rolls-Royce on gas-pumping engines is to be extended to cover the AL-41 shaft derivatives.

The following refers to the AL-41F:

Type

Two-shaft variable-bypass augmented turbofan.

LP Compressor

No inlet guide vanes. Minimum number of stages with maximum work per stage. FPR 25 per cent higher than that of the AL-31F. Designed for sustained flight with AOA (Angle Of Attack) from 0-360º.

HP Compressor

Fewer than eight stages, with special provisions to reduce leakage and perfect aerodynamics.

Combustion Chamber

Advanced annular `with effective mixture formation and cooling'. No visible smoke.

HP Turbine

Highly loaded single stage with single-crystal blades and `incorporating a new cooling concept'. Active clearance control. TGT `12 per cent higher than in fourth-generation fighter engines'. It was later said to be `155 to 200ºC higher'.

LP Turbine

Single stage with advanced cooling.

Afterburner

High-intensity radial combustion. Engine designed for dry supercruise.

Nozzle

Multimode variable con/di type with unique TVC system. Nozzles for single- and twin-engine installations are quite different.

Accessories

In Su-32FN (as in MFI), both engines drive a central *agregat* (multifunction gearbox) on which the starter and accessories are mounted. The lubrication system is designed for sustained negative-g flight.

Control System

FADEC, with health monitoring, with hydraulic backup.

Dimensions

Similar to AL-37FU.

Weight, Dry

	about 1,850 kg (4,078 lb)

Performance Rating

Max T-O	175 kN (39,336 lb st)
Max dry	about 113.9 kN (25,600 lb st)

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AERO-ENGINES - PROPFAN, RUSSIA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-93

The Kuznetsov bureau is a pioneer not only of cryogenic fuels but also of propfans. Studies in this field began in 1974, following the `fuel crisis'. The impressive open-rotor NK-62 demonstrator (245 kN, 55,155 lb st) was first tested in 1983, recording the unprecedented sfc of 7.92 mg/Ns (0.28 lb/h/lb st). Several other designs were studied, including engines with the gearbox and fans at the rear. Under chief designer Valentin Anisimov, development focused in 1986 on the smaller NK-93 (initially partnered by the military NK-92). This has gear-driven shrouded contrarotating front fans, with variable-pitch blades.

The first core was tested in February 1988, and the first complete engine ran in August 1991. Prototype NK-93 engines have been manufactured by JSC KMPO at Kazan in co-operation with JSC Motorostroitel and JSC Metallist at Samara. By mid-1992 five engines were on test, and by 1997 over 10,000 hours had been logged and several engines had been air tested with an Il-76LL testbed at the Gromov Flight Research Institute. Thrust levels at S/L routinely exceeded 200 kN (44,967 lb st) for forward propulsion, and 139 kN (31,000 lb st) braking with the fan blades in reverse pitch. Funds have not permitted flight testing in the initial intended applications, the Il-96MK and Tu-204-230, and though the intended certification date was early 1997, work was nowhere near completion in 2001.

Despite this, in February 2001 it was reported that CIAM had completed testing, and that production was ``now planned to begin at KMPO''. In October 2001 further details of progress were given by Stanislav Svatenko, chief of thermo gas-dynamics at Samara Kuznetsov SNTK (a new title which

suggests a formal merger of NK with SMPO, though the Editor cannot yet confirm this). Svatenko said that at that time KMPO, Motorostroitel and Metallist had almost completed the tenth NK-93, of a planned total of 15. Testing on the ground and in the Il-76LL would continue through 2002, for planned certification in late 2003 and the start of production in 2004.

Originally other applications envisaged included the Il-96-500, Tu-214, Tu-230, Tu-304 and Tu-330. By 2002 the Tu-330 large twin-engined airlifter had come to the fore as the most immediate application. Long shelved through lack of funds, this potentially important rival to Ukraine's An-70 (and, indeed, the proposed A400M) received enough Russian funds in 2001 for AO Tupolev to resume work on it. It was originally intended to be powered by PS-90A turbofans.

Several turbofan variants are being studied. According to Svatenko, these would have direct-drive fans, and ratings of 11, 12 and 16 tonnes (107.9, 117.7 and 156.9 kN; 24,250, 26,455 and 35,275 lb st).

NK-92

In 1996 under development for military use. Performance the same as NK-93. At that time selected for four-engined Il-106 strategic airlifter and Il-90-200.

NK-94

Under development with cryogenic fuels, especially LNG. Performance the same as NK-93. Selected for several projected aircraft including Il-96M derivative, Tu-156M2, Tu-216 and Tu-338.

Type

Three-shaft geared contrarotating shrouded propfan.

Fan

Two stages, contrarotating, same directions as Tu-95 (`Bear') propellers. Front fan (40 per cent power) eight blades, rear (60 per cent power) 10 blades. Blades swept 30º, pitch range 110º. Prototype blades solid magnesium, production blades (by Stupino propeller factory) solid sparless graphite-epoxy composite retained by short steel root slotted into disc. Blade length 1,050 mm (41.34 in). Mass flow (cruise rpm, S/L) 1,000 kg (2,205 lb)/s. Bypass ratio 16.6.

Output

Planetary gearbox, transmitting 22,370 kW (30,000 shp) through seven satellite pinions. Designed for service life of 20,000 hours.

LP Compressor

Seven-stage, titanium discs and blades.

HP Compressor

Eight-stages, the first five titanium, last three steel. Overall pressure ratio (T-O) 28.85, (Cruise, as below) 37.0.

Combustion Chamber

Fully annular with vaporising burners. Being studied for use of LNG (liquefied natural gas) fuel.

HP Turbine

Single-stage with cooled single-crystal blades drives HP compressor.

IP Turbine

Single-stage IP drives LP compressor.

LP Turbine

Three-stage LP, drives propfan gearbox.

Dimensions

Length	5,972 mm (235 in)
Inlet diameter	1,455 mm (57.28 in)
Propfan diameter	2,900 mm (114.2 in)
Shroud external diameter	3,150 mm (124.0 in)

Weight, Dry

	3,650 kg (8,047 lb)

Performance Ratings

T-O (ISA, S/L)	176.5 kN (39,683 lb st)
Cruise (11,000 m; 36,090 ft, M 0.75)	31.38 kN (7,055 lb)

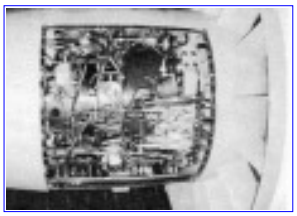
Specific Fuel Consumption

T-O, as above	6.52 mg/Ns (0.23 lb/h/lb st)
Cruise, as above	13.89 mg/Ns (0.49 lb/h/lb)

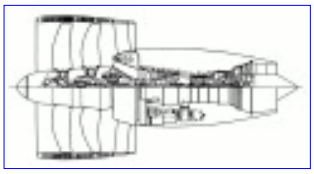
UPDATED

Close-up of NK-93 propfan blades

NK-93 in configuration for Il-96-300 and Tu-204



NK-93 core seen through cowl door



Longitudinal section through NK-93 (2001)

NK-93A in assembly 1998 with fan case removed (2000)

The NK-93A engine on outdoor test (2000)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

PS-90A76

This engine has been derived from the PS-90A to replace the D-30KP-II as the power plant of the Il-76TD and Il-76TF. Compared with the D-30KP the sfc is reduced by 26 per cent at T-O and 17 per cent in cruise. The engine differs from the original PS-90 mainly in the anti-icing bleed air system and in the fact that the reverser hydraulics operate on AMG-10 fluid instead of NGZh-5U.

Fan

Mass flow 500 kg (1,102 lb)/s. Pressure ratio 1.722. Bypass ratio 4.29.

HP Compressor

Overall pressure ratio 26.8.

HP Turbine

Improved materials, entry temperature 1,590K (1,317ºC).

Performance Ratings

T-O (flat rated to 730 mm, 30ºC)	142.2 kN (31,967 lb st)
Max available	156.9 kN (35,275 lb st)

Specific Fuel Consumption

Cruise (11,000 m, 36,090 ft, M0.8)	16.79 mg/Ns (0.593 lb/h/lb)

UPDATED

PS-90A76
(2002)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

PS-90A2

This engine replaced a joint effort with Pratt & Whitney designated PS-90P. It was designed to replace the PS-90A on the Il-96-300 and the PW2337 on the IL-96M/T. It is interchangeable with the PS-90A, but is upgraded in the following items: from USA, improved HP turbine and bearing support (Pratt & Whitney), FADEC control system (Hamilton Standard), pneumatic valves (AlliedSignal) and oil filters (Pall); from France, optical pyrometer (Auxitrol) and electrical connector (Deutsch); from Germany, on-board monitoring system (BGT), fuel/oil heat exchanger (Behr) and bearings (FAG); from Sweden, bearings (SKF), and from Russia, design refinement and certification (AVI) and production (PECo [Perm Engine Co]). Its description differs from the basic PS-90A in the following:

Fan

Mass flow 500 kg (1,102 lb)/s. Pressure ratio 1.746. Bypass ratio 4.29.

HP Compressor

Overall pressure ratio 35.2.

HP Turbine

Entry gas temperature 1,750 K (1,477ºC).

LP Turbine

Max gas temperature 933 K (660ºC).

Control System

FADEC.

Specific Fuel Consumption

Cruise (11,000 m, 36,090 ft, M0.8)	16.99 mg/Ns (0.600 lb/h/lb)

UPDATED

PS-90P, overtaken by the PS-90A2

PS-90A2
(2002)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

D-20P

In 1955 P A Solovyov began the design of the D-20 low-ratio turbofan engine (at the time often called a bypass turbojet) for long-range bombers. The complete D-20 engine first ran in late 1956, and several underwent prolonged testing to clear the production **D-20P** for airline service as the engine of the Tu-124 short-haul passenger aircraft in ambient conditions between ±40ºC.

The D-20P entered service with Aeroflot in October 1962. It was the first turbofan to go into regular operation in the Soviet Union. It established an outstanding record for reliability. About 3,000 were delivered, and the Tu-124 remained in service until 1981.

Type

Two-shaft turbofan (bypass turbojet).

Air Intake

Eight radial struts and central bullet fairing, de-iced by hot bleed air from fourth HP stage (from final stage at low rpm).

Fan

Three-stage axial, with supersonic blading in first stage. Mass flow 113 kg (249 lb)/s at 8,550 rpm. Pressure ratio (S/L, static at max continuous 7,900 rpm) 2.4:1. Bypass ratio 1.

Compressor

Eight-stage axial. Automatically controlled flap valves downstream of the third and fourth stages bleed air into the fan duct to stabilise behaviour. Pressure ratio (at max continuous, 11,170 rpm) 5; overall pressure ratio 13.

Combustion Chamber

Can-annular, with 12 flame tubes each fitted with duplex burner.

Fuel Grade

T-1, Ts-1 To Gost 10227-62 (Avtur-50 to DERD.2494, MIL-F-5616).

Turbine

Single-stage HP turbine with cast blades; stator blades and both sides of disc cooled by bleed air. Two-stage LP turbine with forged blades. Max gas temperature downstream of turbine 650ºC.

Jetpipe

Concentric pipes for fan airflow and core gas, terminating in supersonic nozzles of fixed-area type.

Lubrication

Open type, with oil returned to tank. Consumption in flight, not over 1 kg (2.2 lb)/h. Typical pressure 3.45-4.41 kg/cm^2 (50-64 lb/sq in).

Oil Grade

Mineral Oil Mk-8 Or Mk-8P To GOST 6457-66 (DERD.2490 or MIL-O-6081B).

Accessories

Two gearboxes provide drives for starter/generator, tachometer, air compressors, hydraulic pump, oil pump and other controls and instruments. For restarting in flight, an altitude sensing device meters fuel flow appropriate to height. An automatic fire extinguishing system is fitted. De-icing of the air intake and inlet guide vanes is controlled automatically. The engine also has oil chip detectors, vibration monitors and turbine gas temperature limiters.

Starting

Electric (DC) system, incorporating STG-18TM starter/generator.

Dimensions

Length overall	3,304 mm (130 in)
Diameter, bare	976 mm (38.3 in)

Weight, Dry

	1,468 kg (3,236 lb)

Performance Ratings

Max T-O	52.96 kN (11,905 lb st)
Long-range cruise, M0.75, 11,000 m (36,000 ft)	10.79 kN (2,425 lb st)

Specific Fuel Consumption

Max T-O	20.4 mg/Ns (0.72 lb/h/lb st)
Long-range cruise, as above	25.5 mg/Ns (0.90 lb/h/lb st)

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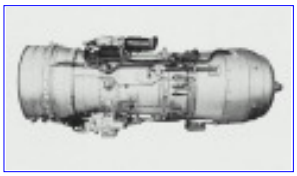
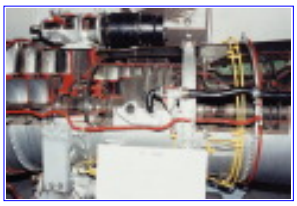
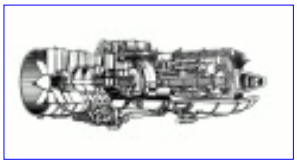


D-20P
(2000)



Cutaway D-20P display engine
(2000)



Cutaway drawing of D-20P
(2000)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

PS-90A10

The PS-90A10 was developed to power short-range aircraft with 100 to 140 passengers, such as the Tu-334.

Type

Two-shaft turbofan with mixer and clamshell reverser.

Fan

Single-stage; 33 titanium blades with snubbers. Mass flow 258.3 kg (569.4 lb)/s. Bypass ratio (T-O) 3.02, (cruise) 2.85. No intershaft bearing in the fan drive, and no booster stages.

HP Compressor

As PS-90A, three rows of variable inlet guide vanes; active clearance control. Overall pressure ratio (cruise) 26.0.

Combustion Chamber

Can-annular.

HP Turbine

Two-stage, with controlled air cooling and active clearance control. TET 1,450 K (1,177ºC).

LP Turbine

Two-stage, with active clearance control.

Jetpipe

Lobe-type mixer, integrated nozzle.

Reverser

Clamshell type, reverse factor 0.4.

Control System

Full-authority digital electronic.

Dimensions

Fan diameter	1,400 mm (55.13 in)
Total length	4,280 mm (168.5 in)

Weight, Dry

	1,900 kg (4,180 lb)

Performance Ratings

T-O, S/L	90.2 kN (20,283 lb st) flat rated to +30ºC, 730 mm Hg
Cruise at 11,000 m (36,090 ft) and M0.8	20.1 kN (4,519 lb)

Specific Fuel Consumption

(installed)

Cruise, as above	17.84 mg/Ns (0.63 lb/h/lb)

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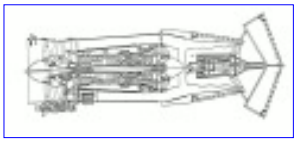

Longitudinal section through PS-90A10 (showing reverser in operation) (2000)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

D-30F6

This large supersonic engine was designed from 1972 expressly for the MiG-31. Requirements included M2.83 cruise at 11,000 to 21,000 m (36,100 to 68,900 ft) with the lowest possible fuel consumption, and M1.25 at S/L. The engine comprises seven interchangeable modules. It has attained an outstanding standard of reliability in several hundred thousand flight hours.

D-30-10V

This is an unaugmented derivative rated at 88.2 kN (19,840 lb st) which powers the high-altitude subsonic Myasishchev M-55. Another unaugmented derivative, in this case for supersonic aircraft, is the D-21A1, described separately.

D-30F6 modified

This engine powers the Sukhoi S-37 FSW (forward-swept wing) research fighter. Ratings are unchanged, and the nozzles are similar to those of the MiG-31, with no provision for vectoring.

Type

Two-shaft augmented turbofan (bypass turbojet).

LP Compressor

Five stages, fixed inlet guide vanes. Mass flow 150 kg (331 lb)/s. Pressure ratio 3. Bypass ratio 0.57.

HP Compressor

Ten stages, first row variable stators and bypass doors behind stages 4 and 5. Pressure ratio 7.05. Overall pressure ratio 21.15.

Combustion Chamber

Can-annular with 12 interlinked flame tubes.

HP Turbine

Two-stage, with max TET 1,387ºC. Cooling air bled from HP stages 5, 10, cooled in heat exchanger in bypass duct.

LP Turbine

Two-stage.

Afterburner

High volume, with four flameholder rings.

Nozzle

Multiflap type with large variable area and cooling flows. Flow stabilised by auxiliary valve plates in divergent petals.

Accessories

Independent gas-turbine APU under compressor used for starting (one per engine). Independent lubrication system.

Control System

FADEC mounted on airframe.

Dimensions

Inlet diameter	1,020 mm (40.2 in)
Length	7,040 mm (277.2 in)

Weight, Dry

	2,416 kg (5,326 lb)

Performance Ratings

(S/L static)

Dry	93.2 kN (20,944 lb st)
Augmented	152.1 kN (34,215 lb st)
Frontal (max)	186.1 kN (41,843 lb st)
Power/frontal area	18,900 kg/m^2 (3,871 lb/sq ft)

Specific Fuel Consumption

Dry	20.4 mg/Ns (0.72 lb/h/lb st)

Augmented	53.8 mg/Ns (1.9 lb/h/lb st)

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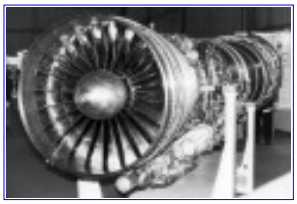

D-30F6

D-30F6

Longitudinal section through D-30F6

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

PS-90A12

This engine was originally developed to power the projected Yak-242. It is also offered for the Myasishchev VGP-60 and as an alternative engine for the Il-76MD, Il-62M and Tu-154M. Booster (LP) stages are eliminated, and the moderate cycle parameters make gas generator operating conditions easier, in comparison with the PS-90A, and meet environmental requirements. The developed gas generator provides low life-cycle cost. Service life is set at 20,000 hours.

Type

Two-shaft turbofan, with mixed bypass and core airflows; reverser in bypass duct.

Fan

Single-stage. Mass flow 370.0 kg (816.0 lb)/s. By-pass ratio (T-O) 5.01; (cruise) 4.9. FPR 1.6. No intershaft bearing.

HP Compressor

Identical to PS-90A, with three variable guide vanes and active control of radial clearance. OPR (max cruise) 23.3.

Combustion Chamber

Can-annular.

HP Turbine

Two stages, with air cooling and active clearance control. TET (T-O) 1,263ºC, (cruise, as below) 1,239 K (966ºC).

LP Turbine

Three stages, with active clearance control.

Exhaust System

Lobe-type mixer and integrated nozzle.

Reverser

Cascade vanes in bypass duct. No core reverser.

Control System

Full-authority digital electronic.

Dimensions

Fan diameter	1,670 mm (65.76 in)
Length	4,795 mm (188.82 in)

Weight

Dry	2,300 kg (5,071 lb)

Performance

T-O, S/L	117.7 kN (26,455 lb st) to 30ºC, 730 mm Hg
Cruise at 11,000 m (36,090 ft) and M0.8	22.55 kN (5,071 lb)

Specific Fuel Consumption (installed)

T-O	10.11 mg/Ns (0.357 lb/h/lb)
Cruise, as above	16.48 mg/Ns (0.582 lb/h/lb)

Modifications of PS-90A:

ENGINE	PS-90A76		D-30KU-90		PS-90A10		PS-90A12		PS-90A		PS-90A2	
Operating conditions and performance	**H=0 M=0 +30º/730 T-O**	**H=11* M=0.8 ISA cruise**	**H=0 M=0 +30º/730 T-O**	**H=11* M=0.8 ISA cruise**	**H=0 M=0 +30º/730 T-O**	**H=11* M=0.8 ISA max cruise**	**H=0 M=0 +30º/730 T-O**	**H=11* M=0.8 ISA cruise**	**H=0 M=0 ISA**	**H=11* M=0.8 ISA max cruise**	**H=0 M=0 ISA**	**H=11* M=0.8 ISA cruise**

Thrust (lb)	31,967 35,273	7,429	23,148	6,063	20,282	4,519 flat rated to ISA +10ºC	26,455	5,071 flat rated to ISA +10ºC	35,273	7,716	35,273	7,716
Specific fuel consumption (lb/h/lb)	0.387	0.593	0.471	0.664	0.427	0.63	0.372	0.582	0.382	0.604	0.378	0.600
Airflow (corrected) (kg/s)	457.7	497.2	258	286.7	265.1	274.1	375.7	381.0	470.5	500	472	499.4
Turbine inlet temperature (ºK)	1,590	1,345	1,520	1,347	1,450	1,200	1,558	1,239	1,560	1,383	1,571	1,393
Bypass ratio	4.48	4.29	2.44	2.36	3.0	2.85	5.01	4.91	4.382	4.272	4.417	4.29
Overall pressure ratio	28.5	35.83	26.7	31.0	23.3	25.9	22.1	23.3	31.32	35.55	30.92	35.19
Fan pressure ratio	1.63	1.72	2.02	2.196	1.71	1.75	1.62	1.61	1.674	1.747	1.673	1.746
Fan speed (rpm)	4,347	4,184	4,834	4,685	6,212	5,765	5,072	4,595	4,335	4,215	4,335	4,210
HP compressor speed (rpm)	11,970	11,165	11,815	11,055	11,537	11,687	11,300	10,316	11,796	11,130	11,750	11,110
Nozzle exhaust velocity (m/s)	330		397		362		333		330	350	329	348
Fuel flow, idle (kg/h)	630	630	500	500	400	400	460	460	630	630	550	550
Engine length (L) (mm)	4,964		5,700		4,732		4,795		4,964		4,964	
Fan diameter (mm)	1,900		1,455		1,400		1,670		1,900		1,900	
Engine dry mass (kg)	2,950		2,400		1,900		2,300		2,950		2,950	
Compressor stages	3+13		3+13		1+13		1+13		3+13		3+13	
Turbine stages	2+4		2+3		2+2		2+3		2+4		2+4	
NOTE	Flight testing on Il-76MF		Passed through the testing. Interchangeable with D-30KU-154		Intended to power Tu-334, etc		Working project for Yak-242 and GP-60		Base engine is operated on Il-96-300 and Tu-204		Joint development of OJSC. `Aviadvigatel' and PW for Il-96-300. Tu-204. Il-96M/T	

***** = kilometres (36,089 ft)

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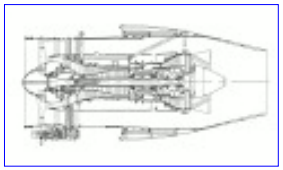
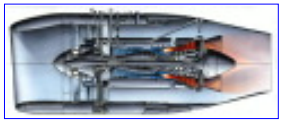


Longitudinal section through PS-90A12 (2000)



Longitudinal section through installed PS-90A12

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

D-110

Work on the complex D-112 four-shaft propfan (described in previous editions) has been terminated, and in late 2001 work had been transferred to the D-110. This is a major redesign of the PS-90A, with the objective of 40 per cent greater thrust and 13 per cent lower specific fuel consumption.

Type

Two-shaft geared ducted turbofan.

Fan

Single stage, with large inserted blades, driven through gearbox of 2.66 ratio. Diameter 2,710 mm (106.7 in). BPR (T-O, S/L) 11.0.

LP Compressor

Two-stage core booster.

HP Compressor

13 stages, with variable IGVs and first two stators. OPR (cruise, as below) 38.0.

Combustion Chamber

Can-annular, becoming annular.

HP Turbine

Two stages, entry temperature 1,366ºC.

LP Turbine

Four stages driving reduction gear.

Jetpipe

Separate fan and core flows.

Reverser

Cascade type, in fan duct.

Weight, Dry

	About 3,600 kg (7,930 lb)

Performance rating

T-O, S/L	215.7 kN (48,460 lb st)

Specific fuel consumption

Cruise (11,000 m; 36,090 ft, Mach 0.8)	14.64 mg/Ns (0.518 lb/h/lb)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

D-100

Aviadvigatel has designed this engine as a next-generation turbofan derived from the PS-90A. It is regarded as the ideal propulsion system for such aircraft as the Tu-204M, Il-96M, Il-106 and A340, pending the arrival of later designs (see D-110 which follows).

Type

Two-shaft turbofan.

Fan

Single stage with very large inserted blades with advanced aerodynamics. Diameter 2,350 mm (95.92 in). Performance (T-O, S/L, 30ºC, static) mass flow 699 kg (1,541 lb)/s, FPR 1.42, BPR 8.1; (cruise, 11,000 m, 36,090 ft at M0.8) mass flow 296 kg (653 lb)/s, FPR 1.5, BPR 7.8.

LP Compressor

Four-stage core booster, rotating with the fan.

HP Compressor

12 stages, with variable IGVs and next two stator rows, part-span snubbers on first two rotor stages. Active clearance control. Pressure ratio (T-O) 14.3, (cruise) 16.8. OPR (T-O) 28.7, (cruise) 36.7.

Combustion Chamber

Either fully annular or as PS-90A with 12 flame tubes (the drawing shows both types).

HP Turbine

Two-stage, with air-cooled monocrystal blades. TGT 1,311ºC.

LP Turbine

Six stages with narrow ring rotor stages.

Jetpipe

Plain core jetpipe without mixer (option of lobe-type mixer and integrated nozzle).

Reverser

Four blocker doors pivoted to rear of short fan duct.

Dimensions

Not published, but rather larger than PS-90A.

Weight, Dry

	3,500 kg (7,716 lb)

Performance Ratings

T-O, as above	186.3 kN (41,887 lb st)
Cruise, as above	37.26 kN (8,377 lb)

Specific Fuel Consumption

T-O, as above	8.1 mg/Ns (0.286 lb/h/lb st)
Cruise, as above	15.405 mg/Ns (0.544 lb/h/lb)

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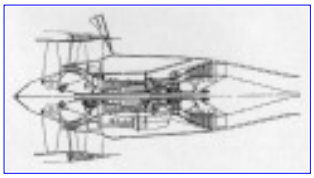

D-100 turbofan, showing reverser operation; annular chamber in lower half

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

D-30KU-90

The D-30KU-90 is a derivative of the D-30KU equipped with the core of the PS-90A engine. The main objective of this modification was to reduce fuel consumption and enable the engine to meet ICAO 2004 emissions and the noise legislation of Chapter 3 Annex 16. The engine is designed to be interchangeable with all variants of the D-30KU, and may be used on the Il-62M and Il-76 and the Tu-154M. Serial production of the engine is scheduled for mid-1997.

Type

Two-shaft bypass turbofan, equipped with a mixer and reverser.

LP Compressor

Three-stage, mainly made of titanium alloys. The first-stage blades have vibration-damping mid-span shrouds. Mass flow 264.94 kg (540 lb)/s at 4,715 rpm. Bypass ratio 2.44.

HP Compressor

13-stage with variable inlet guide vanes and the first two stators. Overall pressure ratio 31.0.

Combustion Chamber

Can-annular type with 12 flame tubes, with a two-stage axial-radial swirler. Two flame tubes are equipped with igniter plugs.

HP Turbine

Two-stage, with a system for cooling rotor discs and all the blades and guide vanes. The first-stage blades are shroudless. The second-stage blades have tip shrouds.

LP Turbine

Three stage. All rotor blades have tip shrouds.

Accessories

Two accessory gearboxes, at front and rear, drive fuel and oil pumps, control units, a constant-speed drive, hydraulic pumps and generator.

Starting

Pneumatic starter energised from an APU, ground power supply or engine cross-bleed.

Control System

Hydromechanical, responsible for the engine start, operation and compressor control.

Oil System

Closed-loop type, with a fuel-oil heat exchanger and centrifugal separator.

Dimensions

Length, with reverser	5,700 mm (224.4 in)
Diameter:	
Inlet	1,455 mm (57.3 in)
Max	1,560 mm (61.4 in)

Weight, Dry

With reverser	2,770 kg (6,107 lb)
Without reverser	2,400 kg (5,291 lb)

Performance Ratings

T-O, S/L:	
For Il-76	117.7 kN (26,455 lb st) to 30ºC
For Tu-154M	103.0 kN (23,148 lb st) to 30ºC
Cruise (11,000 m; 36,090 ft, M0.8)	26.97 kN (6,063 lb)

Specific Fuel Consumption

Cruise, as above	18.81 mg/Ns (0.664 lb/h/lb)

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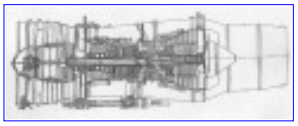

Longitudinal section through D-30KU-90

D-30KU-90

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

D-30KU

Despite its designation, this engine has no part in common with the D-30, and is considerably larger and more powerful. It was first certificated in 1971, and several versions are now in service:

D-30KU

Fitted to all current versions of Il-62M. T-O rating 107.9 kN (24,250 lb st) up to 21ºC. Outer engines fitted with airframe-mounted up/down reversers.

D-30KU II

Also designated **KU-154**, this version is configured to suit the Tu-154M and immediately related versions. T-O rating flat rated at 104 kN (23,830 lb st) to ISA+15ºC. The two side-mounted engines have engine-mounted up/down reversers. Note: the D-30KU-154-111 is described under Rybinsk Motors.

D-30KP

The original engine of all versions of Il-76. Mass flow 280 kg (617.3 lb)/s. TET 1,152ºC. T-O rating 117.7 kN (26,460 lb st) up to ISA+15ºC. All engines fitted with airframe-mounted lateral reversers.

D-30KP-II

Current engine of the Il-76T, (all versions), Il-78 tanker and A-50 `AWACS'. T-O rating 117.7 kN (26,460 lb st) up to ISA+23ºC. All are manufactured by **Rybinsk Motors jsc**, which by June 2000 had delivered 2,812.

D-30KPV

Fitted to A-40. Generally similar to KP, and similarly rated, but without reverser.

The following refers to the D-30KU:

Type

Two-shaft turbofan, with mixer and reverser.

Intake

Fixed ring with multiple fixed-incidence aerofoil struts carrying front LP bearing. Bleed air and oil anti-icing.

Fan (LP Compressor)

Three stages, mainly of titanium alloy. First-stage rotor blades with part-span snubbers. Mass flow, 269 kg (593 lb)/s at 4,730 rpm (87.9 per cent), with bypass ratio of 2.42.

HP Compressor

11 stages, first two having part-span snubbers. Guide vanes pivot 30º over 7,900 to 9,600 rpm, while air is bled from fifth and sixth stages. Overall pressure ratio (S/L, static) 20 at HP speed of 10,460 rpm (96 per cent).

Combustion Chamber

Can-annular type with 12 flame tubes. Each tube comprises hemispherical head and eight short sections welded with gaps for dilution air. Single-swirl type main/pilot burner centred in each tube. Igniter plugs in two tubes.

HP Turbine

Two-stage with cooled blades in both stages. Second-stage rotor blades tip shrouded. TET 1,127ºC.

LP Turbine

Four-stage, with shrouded blades.

Jetpipe

Forced mixer and common pipe for core and bypass flows. Reverser and nozzle varies with installation.

Accessories

Front and rear drive boxes under engine carry all shaft-driven accessories. Differential constant-speed drive to alternator and air turbine starter.

Starting

Pneumatic starter fed by ground supply, APU or cross-bleed.

Fuel Specification

T-1, TS-1, GOST-10227-86, A-1 (D1655/63t), DERD.2494 or 2498, Air 3405/B or 3-GP-23e.

Oil System

Closed type. Fuel/oil heat exchanger and centrifugal air separator with particle warning.

Oil Specification

MK-8 or MK-8P to GOST 6457-66 (mineral) or BNII NP-50-1-4F to GOST 13076-67 (synthetic).

Dimensions

Length with reverser	5,700 mm (224 in)
Inlet diameter	1,455 mm (57.28 in)
Max diameter of casing	1,560 mm (61.4 in)

Weight, Dry

With reverser	2,668 kg (5,882 lb)
Without reverser	2,318 kg (5,110 lb)

Performance Ratings

T-O	107.9 kN (24,250 lb st) to 21ºC
Cruise at 11,000 m (36,090 ft) and M0.8	26.97 kN (6,063 lb)

Specific Fuel Consumption

At T-O rating	13.83 mg/Ns (0.49 lb/h/lb st)
Cruise, as above	19.83 mg/Ns (0.70 lb/h/lb)

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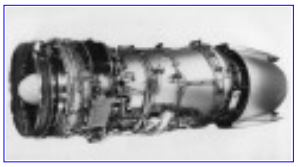
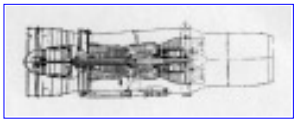


D-30KP-II



Longitudinal section through D-30K with plain jetpipe

Russian cutaway drawing of the D-30KU II

D-30KU II

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

D-30

In 1961 the design bureau of P A Solovyov received a contract for the development of a more powerful engine than the D-20P to meet the requirements of the Tu-134. The resulting D-30 featured zero stages added to the fan and compressor, to increase both the airflow and pressure ratio.

D-30

Initial version, state-tested in 1967.

D-30 II

Fitted with reverser. Powered Tu-134A from 1972.

D-30 III

Fitted with zero-zero stage on fan, to maintain existing ratings to 25ºC, and with reduced TET. Powers Tu-134A-3. The data below refer to this version.

Type

Two-shaft turbofan (bypass turbojet).

Intake

Fabricated from titanium alloy. Air-bleed anti-icing of central bullet and radial struts.

Fan

Five-stage axial (LP compressor) (II, four-stage). First stage has shrouded titanium blades held in disc by pinned joints. Pressure ratio (T-O rating, 7,700 rpm, S/L, static), 2.65. Mass flow 126.8 kg (279.5 lb)/s. Bypass ratio 1.

Compressor

Ten-stage axial (HP compressor). Drum and disc construction, largely of titanium. Pressure ratio (T-O rating, 11,600 rpm, S/L, static), 7.1. Overall pressure ratio, 17.65.

Combustion Chamber

Can-annular, with 12 flame tubes fitted with duplex burners.

HP Turbine

Two-stage HP turbine. First stage has cooled blades in both stator and rotor. TET 1,087ºC.

LP Turbine

Two stages. All blades shrouded and bearings shock-mounted.

Jetpipe

Main and bypass mixer with curvilinear ducts. D-30-II engine of Tu-134A fitted with twin-clamshell reverser.

Accessories

Automatic ice protection system, fire extinguishing for core and bypass flows, vibration detectors on casings, oil chip detectors and automatic limitation of exhaust gas temperature to 620ºC at take-off or when starting and to 630ºC in flight (5 minute limit). Shaft-driven accessories driven via radial bevel gear shafts in centre casing, mainly off HP spool, with gearboxes above and below centre casing and fan duct. D-30-II and III have constant-speed drives for alternators.

Starting

STM-10 pneumatic starter fed by ground supply. Series II, STG-12TVMO starter/generators.

Fuel Specification

T-1 and TS-1 to GOST 10227-62 (equivalent to DERD.2494 or MIL-F-5616).

Oil System

Open type, with oil returned to tank. Consumption in flight not over 1 kg (2.2 lb)/h.

Oil Specification

Mineral oil MK-8 or MK-8P to GOST 6457-66 (equivalent to DERD.2490 or MIL-O-6081B).

Dimensions

Length overall	3,983 mm (156.8 in)
Base diameter of inlet casing	1,050 mm (41.3 in)

Weight, Dry

	1,550 kg (3,417 lb)

Performance Ratings

T-O	66.68 kN (14,990 lb st)
Long-range cruise rating, 11,000 m (36,000 ft) and M0.75	12.75 kN (2,866 lb)

Specific Fuel Consumption

T-O	17.22 mg/Ns (0.608 lb/h/lb st)
Cruise, as above	22.38 mg/Ns (0.79 lb/h/lb)

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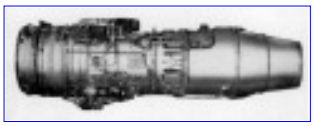

D-30 initial series

D-30 III two-shaft turbofan with reverser

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AERO-ENGINES - TURBOSHAFT, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

D-25V

The engine KB (design team) at Perm, headed by Solovyov from 1953, worked not only on the D-20 but also on two turboprops, the D-19 and TV-2M. From the latter was derived the **TV-2BM** helicopter engine, first tested in 1955. This enabled Mil' to produce the Mi-6, by far the largest and most capable helicopter ever produced at that time (or until the later Mi-26). The production engine has the designation **D-25V.**

The complete helicopter power plant comprises two D-25V engines, identical except for handed jetpipes, and an R-7 gearbox. The latter has four stages of large gearwheels providing an overall ratio of 69.2:1. The R-7 is 2,795 mm (110.04 in) high, 1,551 mm (61.06 in) wide and 1,852 mm (72.91 in) long. Its dry weight is 3,200 kg (7,054 lb), more than the weight of the pair of engines which drive it.

The **D-25VF** turboshafts fitted to the Mi-10 and Mi-10K crane helicopters are uprated to 4,847 kW (6,500 shp). These engines have increased mass flow and operate at higher turbine gas temperatures.

Total production of D-25 engines exceeded 2,200. In 1999, most were still in use in various Mi-6 versions.

The following details apply to the basic D-25V:

Type

Single-shaft turboshaft with free power turbine.

Intake

Six hollow radial struts, the two vertical struts housing splined shafts driving upper and lower accessory drive boxes. Vertical struts de-iced by oil drained from upper drive box; four inclined struts and bullet fairing

de-iced by hot oil returned from engine to tank.

Compressor

Nine-stage axial. Comprises fixed inlet guide vane assembly, first-stage stator ring, upper and lower casings with dovetailed stator blades, ninth-stage stator ring and exit vanes, rotor, and air blow-off valves. Mass flow 26.2 kg (57.8 lb)/s. Pressure ratio 5.6.

Combustion Chamber

Can-annular. Assembled from diffuser (the structural basis of the engine), inner shroud, 12 flame tubes with transition liners, diaphragm and compressor-shaft shroud.

Compressor Turbine

Single-stage compressor turbine, overhung behind rear roller bearing. TET 967ºC.

Power Turbine

Two-stage, overhung on end of rear output shaft. Both turbines rotate counter-clockwise, seen from the rear. Normal power turbine rpm, 7,800 to 8,300; maximum 9,000. Transmission shaft in three universally jointed sections, allowing for 10 mm (0.4 in) misalignment between engine and gearbox.

Jetpipe

Large fabricated assembly in heat-resistant steel, curved out to side to allow rotor transmission to pass through duct wall in air-cooled protecting pipe.

Accessories

SP3-12TV electric supply and starting system; fuel supply to separate LP and HP systems; airframe accessories driven off upper and lower gearboxes on inlet casing.

Starting

The SP3-12TV system starts both engines and also generates electric current. It comprises an STG-12TM starter/generator on each engine, igniter unit, two spark plugs with cooling shrouds, two switch-over contactors, solenoid air valve, pressure warning, PSG-12V control panel and electrohydraulic cutout switch of the TsP-23A centrifugal governor. In the starter mode the system draws current from a ground supply receptacle or from batteries.

Fuel Specification

T-1, TS-1 to GOST 10227-62 (DERD.2494, MIL-F-5616).

Oil System

Pressure circulation at 3.45 to 4.41 kg/cm^2 (50 to 64 lb/sq in). Separate systems for gas generator and for power turbine, transmission and gearbox.

Oil Specification

Gas generator, MK-8 to GOST 6457-66 or transformer oil to GOST 982-56. Power turbine and gearbox, mixture (75-25 Summer, 50-50 Winter) of MK-22 or MS-20 to GOST 1013-49 with MK-8 or transformer oil. Hourly oil consumption, gas generator not over 1 kg (2.2 lb), power turbine and transmission not over 2 kg (4.4 lb).

Dimensions

Length overall, bare	2,737 mm (107.75 in)

Length overall with transmission shaft	5,537 mm (218.0 in)
Inlet diameter	572 mm (22.5 in)
Width	1,086 mm (42.76 in)
Height	1,158 mm (45.59 in)

Weight, Dry

With engine-mounted accessories	1,325 kg (2,921 lb)

Performance Ratings

T-O	4,101 kW (5,500 shp)
Rated power	3,504 kW (4,700 shp)
Cruise (1,000 m; 3,280 ft, 135 knots; 250 km/h; 155 mph)	2,983 kW (4,000 shp)

Specific Fuel Consumption

T-O, as above	108 µg/J (0.639 lb/h/shp)
Cruise, as above	118.1 µg/J (0.699 lb/h/shp)

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D-25V

Simplified cutaway drawing of D-25V

D-25V with left-hand jetpipe

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 22 November 2001

Jane's Aero-Engines 11

`AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

D-21A1

The D-21A1 is a bypass engine equipped with a mixer and an axially symmetric supersonic nozzle. It is designed to power the Sukhoi S-21 high-altitude supersonic business jet.

The D-21A1 is a non-augmented modification of the D-30F6. It conforms to requirements concerning emission and noise levels (Chapter 3). Its effect upon the ozone layer is minimal. One of the main advantages of the design is a short-term period of engineering, and minimum cost of development.

Type

Two-shaft, bypass turbofan.

LP Compressor

Five-stage, fixed inlet guide vanes. Mass flow 153 kg (337.3 lb)/s. Overall pressure ratio 2.99 (climb, 12,000 m; 39,370 ft, M1.2).

HP Compressor

Ten-stage spool, variable inlet guide vanes, 4th and 5th-stage air bleeds. Overall pressure ratio 20.15. Bypass ratio 0.83 (maximum cruise, 15,800 m; 51,850 ft, M2, ISA+10ºC).

Combustion Chamber

Can-annular with 12 flame tubes.

HP Turbine

Two-stage, TET 1,266ºC at maximum cruise.

LP Turbine

Two-stage.

Nozzle

Multiflap, supersonic.

Dimensions

Length	4,837 mm (190.4 in)
Diameter	1,020 mm (40.16 in)

Weight, Dry

	2,100 kg (4,630 lb)

Performance Ratings

T-O, S/L	52.27 kN (11,750 lb st)
Subsonic cruise, 11,000 m (36,000 ft) M0.9, ISA+10ºC	10.10 kN (2,270 lb)
Max acceleration, 12,000 m (39,370 ft) M1.2, ISA+10ºC	28.44 kN (6,393 lb)
Max mode, 15,800 m (51,850 ft), M2, ISA+10ºC	23.34 kN (5,247 lb)

Specific Fuel Consumption

(engine installed)

Subsonic cruise	26.9 mg/Ns (0.95 lb/h/lb)
Max mode	33.7 mg/Ns (1.19 lb/h/lb)

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Longitudinal section of D-21A1

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 22 November 2001

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`AVIADVIGATEL' - `AIRCRAFT ENGINES' OPEN STOCK COMPANY (AVIADVIGATEL)

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This design bureau, one of the largest and most famous in the former USSR, was founded in 1934 as the KB (design bureau) of No 19 aero-engine GAZ (State Aviation Factory) in Perm. From the outset the General Constructor (Chief Designer) was Arkadii Dmitriyevich Shvetsov, whose M-11 radial of 81 kW (100 hp) was already well on its way to its final production total of over 130,000. He later created the ASh series of more powerful engines, of which the ASh-21, ASh-62 and ASh-82 were to rival the M-11 in importance. The TK-19, the most powerful turbosupercharger ever put into production, gave him a feel for gas turbines, but he died in 1953 and was succeeded by Pavel Aleksandrovich Solov'yov (often written Soloviev). He was handed Kuznetsov's TV-2 shaft-drive engine, then rated at 3,680 kW (5,005 shp) and told to develop it further. In 1956 Solov'yov developed the D-20P turbofan and then used this as the basis for the D-25V turboshaft, by far the most powerful helicopter engine of that era.

Interchangeability of elements between different engine types was one of Solov'yov's guiding principles. His engines eventually flew over 59 million hours in 42 countries, powering such aircraft as the Mi-6 and Mi-10 helicopters and the Tu-124, Tu-134, Tu-154M, Il-62M, Il-76 and MiG-31.

The PS-90A, Solovyov's last design, is fitted to the Tu-204, Il-76MF and Il-96-300. The engines designed by `Aviadvigatel' are manufactured by Perm Motors, except for the D-30KU family, which are the responsibility of both Perm and Rybinsk. Many of the new designs are based on the PS-90A core: the PS-90A10, PS-90A12, PS-90A76, D-30KU-90, D-100, D-110, and D-112. Attention has also been paid to propfans with contrarotating fans, with a bypass ratio of about 15. Another area of design is industrial gas-turbine plant for gas pumping and power generation. Aviadvigatel has been able to preserve its personnel, and is developing international links.

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 24 October 2001

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LYUL'KA SATURN INC

AL-37FU

The designation of this fighter engine development prototype stems from *Forsazh* (boosted, that is, with augmentor) and *Upravlyaemye soplo* (controlled, that is, with vectoring nozzle). Though derived from the AL-31 and AL-35 families, this engine incorporates not only a vectoring nozzle but also numerous other advanced features, including a new fan (LP compressor). The engine has been cleared for series production with the designation AL-31FP (which see).

In September 1996 chief designer Anatoliy Andreyev said that one of the most difficult design problems was sealing the joint between the jetpipe and nozzle. At maximum power at sea level the gas pressure at this joint reaches 7 atm (103 lb/sq in) and temperature is around 2,000ºC (3,632ºF).

At the same time Dr Chepkin said that, because of lack of government funds, no orders had yet been placed for the AL-37FU, and that the design bureau has been able to afford only three prototype engines. The first has been on bench testing since 1994, and has been instrumental in proving numerous design features and further improvements. Thanks to a staged cash payment for Su-27s exported to China, Sukhoi Design Bureau has been able to fund a flight-test programme for the other two engines in aircraft No 711, the T-10M-11 (Su-35 demonstrator, see AL-35).

This programme, based at the NII-VVS (air force flight test centre) at Zhukovskiy, was split into two stages. The first stage of 46 (of a planned 50) flights was completed between 2 April and 28 August 1996. On 14 June, during the *Engine `96* show, a public demonstration was planned, but it was cancelled because of Russian elections. In September, No 711 was impressively demonstrated at the Farnborough airshow, making `Cobra' manoeuvres through angles in excess of 100º. The second test stage, also of 50 flights, was completed by the end of 1996. The intention was then to hand the AL-37FU over to a serial plant for production, with the designation AL-31FP. The latter is the engine of both the Su-30M and export Su-30MK two-seat multirole aircraft, and also the Su-37 single-seat fighter (not to be confused with the S-37 *Berkut*) and the planned lightweight S-55. At the

time of writing, no series contract had been signed; the problem is purely shortage of funds.

A Sukhoi chief designer, Vladimir Konokhov, has stated that the entire TVC (thrust-vectoring control) system can be retrofitted to existing Su-27 and related aircraft. No decision has yet been obtained from the Russian government regarding whether it can be exported, though it has been announced that the 40 Su-30MKIs being built at Irkutsk for the Indian Air Force, with deliveries from late 1997, ``will have thrust vectoring available from 1999". From 2002 India plans to build the Su-30MKI under licence, with the AL-31FP engine.

A the 1999 Paris airshow an upgraded Su-30MK took part in the flying display (it was unfortunately lost through no fault of the engines, the crew ejecting). Among its new features were AL-37FU engines with the nozzles rotated to vector the nozzles on an axis inclined at 60º, symmetrically on the left/right engines. Thus, on this aircraft it was possible to impart vectored thrust in yaw as well as pitch and roll. According to test pilot Vyacheslav Averyanov ``In addition, the TVC programming can command the nozzles to act independently and apply differential thrust and differential vectoring".

Dr Chepkin has revealed that ``A package of measures has been developed to reduce infra-red signature in the non-afterburning mode. These could be integrated into serial engines at a customer's wish". At present the TBO is 1,000 hours, except for the nozzle which is currently lifed at 250 hours, with the intention that this should be doubled to 500 when current testing is completed.

At the MAKS airshow in August 2001 Yuri Koptev, Director of Aviakosmos, said that in the first quarter of 2002 a crucial decision would be taken on Russia's next-generation fighter, the PAK-FA. Despite the existence of the all-new AL-41F, Lyul'ka Saturn confirmed that engine proposals in the AL-37F family were being considered, competing against the smaller engines by Klimov. Though not confirmed officially, it is widely believed that the PAK-FA is being considered in two sizes, an 18-tonne MiG or a 23/25-tonne Sukhoi. This would obviously affect the choice and number of engines.

It should be emphasised that AL-37 was a designation for development engines, used in defining the AL-31FP. The following were basic features:

Type

Two-shaft augmented turbofan with fully variable vectoring nozzle.

Fan

Four stages with wide-chord blades handling significantly greater mass flow than the AL-31F.
Computer-designed blades `minimising vibration problems'. First-stage blades slotted into disc and readily replaceable, and the complete fan module is also replaceable. New variable IGVs (0 to -30º) to avoid surge even in tail-first flight.

Compressor

Nine stages, with first three vanes variable (IGVs plus next two stators). All stages of blading readily replaceable. OPR 25. BPR about 0.65.

Combustion Chamber

Annular with 28 downstream burners fed from inner manifold.

HP Turbine

Single stage, cooled blades. Air/air heat exchanger controlling active tip clearance system.

LP Turbine

Single stage, replaceable as a module.

Jetpipe

Short mixer section upstream of augmentor, replaceable as a module.

Afterburner

Configuration generally similar to AL-31F, replaceable as a module. Acoustic linkage with fan almost eliminated to avoid transfer of pressure fluctuations. This is achieved mainly by the air/air heat exchanger in the bypass duct.

Nozzle

Axisymmetric convergent/divergent with secondary flow between 16 pairs of inner and outer flaps. The entire nozzle is mounted on a machined forging of refractory steel (planned to be replaced by titanium alloy to reduce weight). This ring combines the forces from two pairs of external actuating cylinders. These are linked to the aircraft's hydraulic system (in a series installation the working fluid would be fuel). The rams can pivot the entire nozzle in any direction, lateral limits being ±8º, but in the Sukhoi installation the proximity of the tailcone limits movement to the vertical plane, the limits being ±15º, but see earlier comments on diagonal-axis vectoring. According to Dr Chepkin, ``Differential operation on a twin-engined aircraft has the same effect as a 3-D multi-axis nozzle". In series engines an automatic back-up system would lock the nozzle horizontal in the event of any failure of any actuator or the supply system.

Control System; FADEC

Integrated with the aircraft fly-by-wire flight-control system. Nozzle movement is commanded by finger pressure on the force-sensing engine throttle levers. Nozzle movement can be made synchronous with that of the aircraft horizontal tails.

Oil System

Totally new, with tank provided with negative overload compartment, flexible scoop, vent valves and cyclone air separator with lock for sustained inverted flight.

Accessories

Mounted on `banana' gearbox above the fan case, driven off the HP spool.

Dimensions

Inlet diameter	932 mm (36.7 in)
Overall length	approx 5 m (197 in)

Weight, Dry

	1,660 kg (3,660 lb)

Performance Ratings

Max augmented	142.2 kN (31,967 lb st)
Max dry	83.36 kN (18,740 lb st)

Specific Fuel Consumption

Max dry	19.18 mg/Ns (0.677 lb/h/lb st)

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Operating linkage of the AL-37FU nozzle

AL-37FU

Nozzle (deflected down) of the AL-37FU (1999)

Nozzles of the 1999 upgraded Su-30MK (1999)

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 04 July 2001

Jane's Aero-Engines 10

SOYUZ - TMKB (TUSHINSKOYE {TUSHINO} ENGINE DESIGN BUREAU) `SOYUZ'

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Chief of Development: Valeriy V Putskov
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This major engine KB (design bureau) was established as OKB-500 on 13 July 1942, the General Designer being Alekseiy D Charomskiy, a pioneer of high-power aircraft diesel engines. He was succeeded by Vladimir M Yakovlev (1946-53), Nikolaiy G Metskvarishvili (1957-65), Konstantin R Khachaturov (1965-82) and Yuriy Ye Shvetsov (1982-87). Nusberg was appointed in 1987.

Following the important ACh family of diesels, V M Yakovlev failed to develop the impressive M501 to the point where it could be cleared for production. It had seven banks each of six cylinders, and

was rated at 6,000 hp. Instead, Factory 500 was introduced to gas turbines by contracts to mass-produce the RD-500, based on the Rolls-Royce Derwent 5. In addition to the engines described (R29, R35 and RD-1700) TMKB 'Soyuz' was from 1962 an important source of small rocket engines of the R210 and RD-210 families for attitude correction of satellites and spacecraft. Today the factory is named after V V Chernyshev (which see).

The third 'Soyuz' company, OAO TMKB (Turayevskoye MKB) produces rockets and missiles, notably the 3D80 and Kh-31 families of air-launched cruise missiles. It also produces afterburners for the RD-33 (see under Klimov).

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AERO-ENGINES - TURBOSHAFT, RUSSIA

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Jane's Aero-Engines 10

ST PETERSBURG NPO IM KLIMOV

VK-2500

Also known as the TV3-117VMA-SBZ, this derivative from the TV3-117 is to be produced jointly with Motor Sich, of Ukraine. In February 2001 it was reported that plans for VK-2500 production were being finalised between the two partners. Klimov's share was ``said by observers to be 20 to 25 per cent'' and to involve assembly of engines for military customers from modules produced at Zaporozhye. Motor Sich is to produce engines for civil helicopters. The types of helicopter were said to be the Ka-32, Ka-50, Ka-52, Mi-8/-17, Mi-24 and Mi-28. In early 2001 an Mi-24VM was flight testing VK-2500 engines.

So far the VK-2500 has been referred to as a turboshaft only, and the example displayed in Moscow in summer 2000 generally resembled previous TV3-117 helicopter engines, apart from the following.

Compressor

Mass flow 9.3 kg (20.5 lb)/s. OPR 10.

Compressor Turbine

TGT 1,020ºC.

Control System

FADEC, type BARK-78.

Dimensions

Length	2,055 mm (80.91 in)

Width	660 mm (25.98 in)
Height	728 mm (28.66 in)

Weight

Dry	295 kg (650 lb)

Performance Ratings

(S/L, static)

Emergency OEI	2,013 kW (2,700 shp)
T-O	1,790 kW (2,400 shp)
Cruise	1,305 kW (1,750 shp)

Specific Fuel Consumption

T-O, as above	78.2 µg/J (0.463 lb/h/shp)
Cruise, as above	85.63 µg/J (0.507 lb/h/shp)

Contract Price

Cheaper than Western engines, at approximately US$600,000.

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VK-2500 turboshaft version (2001)

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 04 July 2001

Jane's Aero-Engines 10

SOYUZ - AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

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Chief Designer: Evgeniy A Fomin
Tel: (+7 095) 242 66 05
Deputy Chief Designer: Aleksandr F Zhirnov
Factory Director: Mikhail N Bychkov

Note: AMNTK 'Soyuz' should not be confused with TMKB 'Soyuz' (which follows the AMNTK 'Soyuz' entries), nor with OAO TMKB 'Soyuz', mentioned under TMKB 'Soyuz'.

In late 1940 senior engine designers Aleksandr Aleksandrovich Mikulin, V Ya Klimov and S K Tumanskiy lobbied Stalin to have engine OKBs (design bureaux) separated from production factories and established as independent entities. Stalin agreed, but the German invasion was a distraction, and it was not until 1942 that Mikulin gained authorisation to take over the almost empty premises of the former OrgAviaProm works. Here, on 18 February 1943, Mikulin opened GAZ (aviation factory) No 300, charged with the development of completely new engines, which took the suffix -300 in their

designation. Later the design bureau was named 'Soyuz', meaning Alliance.

Previously famous for high-power piston engines, Mikulin chose as his deputies Tumanskiy and Boris Sergeyevich Stechkin. From 1945 they directed a team made up mainly of captured German engineers in the design of the AM-TKRD-01, rated at 32.33 kN (7,275 lb st), which powered the forward-swept EF-140 jet bomber of 1948, and the AM-TRD-02, rated at 41.64 kN (9.369 lb st), which powered the Tupolev 82. His greatest engine was the AM-3, described later. With the help of his deputy Tumanskiy he produced the AM-5, which powered the Yak-25 night fighter and KSShCh cruise missile. This engine led to the RD-9 described later.

Mikulin died in 1955 and was succeeded by Tumanskiy, who directed the design of a succession of simple two-shaft fighter turbojets which were made in enormous quantities. These retained the designation suffix 300, and in 1966, the factory was given the name Soyuz. In 1973 Tumanskiy died; he was succeeded by Oleg N Favorskiy, who in turn was succeeded in 1987 by Kobchenko. The -300 family of turbojets have powered aircraft to altitudes exceeding 30 km (98,425 ft) and 3,000 km/h (1,864 mph).

These engines were mass-produced by several factories, notably Ufa which assigned its own system of Product numbers. By 2001 deliveries of these engines exceeded 26,500, with aggregate flight time exceeding 60 million hours.

Among other products of this establishment are rocket engines for large cruise missiles. Notable examples are the R201-300, engine of the Kh-22 (NATO name `AS-4 Kitchen'), and the R209-300, which powers the KSR-2 (NATO name `AS-5 Kelt').

AMNTK `Soyuz' tried to survive as a separate corporation eager to preserve its tremendous breadth of technical capabilities. In addition to engines described below, it has produced the RDK-300 turbofan in the 3.43 kN (772 lb st) class and the KR7-300 rated at 21.37 kN (4,806 lb st) for unmanned aircraft, several types of liquid-propellant rocket engines, a thermionic energy unit for spacecraft and a new type of turbo-refrigerator. It held discussions with Textron Lycoming (now part of Honeywell), with a view to mutual co-operation in engines for regional transports. It is now engaged upon its own range of turbofans of 3.92 to 39.2 kN (882 to 8,820 lb st) and related turboshaft engines of 300 to 5,000 kW (402 to 6,700 shp).

Wisely, in 1998 Soyuz became part of the MAPO-MiG conglomerate. It thus partners the Klimov team in St Petersburg as this group's engine design team.

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 04 July 2001

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RYBINSK MOTORS - RYBINSK MOTORS JSC

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In 1939 the USSR authorised a new engine KB (design bureau) at the Moscow Aeronautical Institute, formed from MAI staff and students, charged with developing the 24-cylinder 2,500 hp M-250. Supervisor was Glib S Skubachevsky, the head of the MAI engine-design department, and his deputy was Vladimir A Dobrynin, who previously had been deputy to Mikulin (see under Soyuz). In 1941 the need to evacuate Moscow disrupted M-250 work, as the KB moved first to Voronezh, then to Ufa, and in 1943 to GAZ No 36 at Rybinsk. Here Dobrynin was appointed Chief Constructor (that is, chief designer), and in 1949 he led the design of the M-253K for the Tu-85. This 4,300 hp compound piston engine almost went into production as the VD-4K, but the Tu-85 was overtaken by the much more powerful turboprop-engined Tu-95. Accordingly, from 1953 the OKB switched its effort to turbojets, its principal work being the VD-7 series of engines (which see).

Though he lived to 1978, Dobrynin retired in 1960 and was succeeded by Pyotr Alekseyevich Kolesov. Under his direction what had in 1966 been renamed RKBM (Rybinsk engine-building design office) became the Soviet centre for lift engines for STOVL (short takeoff, vertical landing) aircraft. His successor, Vladimir I Galiguzov, was chief designer of the RD-38, and he in turn was succeeded by Aleksandr S Novikov by the time the RD-41 was needed (for these engines, see other entries).

In 1997, Rybinsk Motors and Rybinsk Engine-Building Design Bureau production factory merged to become a stronger integrated entity. Today the main activities are production, overhaul and repair of the D-30KU, D-30KU-154, D-30KP-II and -III turbofans, the RMZ-320MR and -640 for microlights, and various non-aero products including large gas turbines for power generation and gas pumping. The factory is tooling up to make the TVD-1500 and RD-600V described on later pages. It would probably share in any future production of the Lyul'ka AL-41F.

Several of the engines described on following pages are no longer active programmes. They are included to make the record more complete.

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 04 July 2001

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KMPO - KAZAN MOTOR-BUILDING PRODUCTION ASSOCIATION JSC

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Director: Aleksandr Pavlov

This joint-stock company was recently formed out of an organisation which has produced aero engines since 1931. It has most recently produced versions of the NK-8 and NK-86. In 1997, in a move towards rationalisation, this production plant became part of the NK group.

Note: KMPO is unrelated to GAZ No 16 at Kazan which in 1954, headed by P F Zubets, was selected to put the Mikulin AM-3 into production and develop further RD-3 versions. When the M-50 supersonic bomber was cancelled, the OKB at GAZ-16 turned to non-aero work.

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Date Posted: 04 July 2001

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KLIMOV CORPORATION - ST PETERSBURG NPO IM KLIMOV

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This large private company has a complex history. Its head office is on the site of the Russian Renault factory, the first factory making car and aero engines in Russia, established in 1912. The title above means Scientific Production Union named for Klimov. Vladimir Yakovlevich Klimov was appointed to head GAZ No 26 in Rybinsk, which in 1935 was assigned to mass-produce the French Hispano-Suiza HS12, with the Soviet designation M-100. From this were derived high-power piston engines such as the VK-103 and VK-105, production of which exceeded 129,000, despite the fact that in October 1941 the office and factory were evacuated to Ufa.

In 1945 Klimov managed conversion of the German Jumo 004B axial turbojet into the RD-10,

produced in modest numbers. In 1946 General Klimov was appointed to head GAZ No 117, once the Russian Renault factory. However, unexpectedly, Britain sold the USSR Rolls-Royce Nene turbojets, and additionally Klimov was put in charge of a ``crash programme'' at GAZ No 45 in Moscow to produce this engine at the maximum rate (the nicety of getting a licence was ignored). Several thousand, designated RD-45 after the factory, were delivered, while Klimov improved it into the VK-1 (which see), production of which exceeded 20,000. In 1952 the VK-3 bypass jet (low-ratio turbofan) was on test with afterburner. Klimov's last engines were the VK-5 and VK-7 for large supersonic MiGs, and the initial design of TV2-117 turboshaft (which see).

Klimov died in 1962 and was succeeded by his First Deputy, Sergei Pietrovich Isotov, who developed gas turbines mainly for helicopters but who in 1968 moved into the field of fighters. Isotov died in 1983 and was succeeded by Vladimir Styepanov (who retired early) and Alexander Sarkisov, but today the bureau has been renamed after its founder, and the `117' is continued in its engine designations.

Engines designed by NPO Klimov are manufactured at Perm (Sverdlov) and Zaporozhye (Motorostroitel), except for the TV7-117 which is in production at the Chernishov factory (Moscow) and Baranov (Omsk), and the RD-33, made by Chernishov. Total production of the TV2-117 and TV3-117 exceeds 27,000. In June 1993 a design and manufacturing link with Pratt & Whitney Canada on turboshaft and turboprop engines has resulted in licence-production of several PT6A turboprop versions, designated PT6K, by Pratt & Whitney (Rus).

The enterprise also developed the helicopter main gearboxes VR-2, VR-8A, VR-14, VR-24, VR-252 and VR-80. These are in production at factories in Russia and Ukraine. The combined total of gas-turbine engines and helicopter gearboxes exceeds 150,000.

By 1998, in a move towards consolidation of Russian industry, Klimov became part of the MAPO-MiG conglomerate. Klimov brought with it the large St Petersburg production plant previously known as No 117 ``Red October.''

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 04 July 2001

Jane's Aero-Engines 10

LYUL'KA SATURN INC

AL-55

This turbofan was revealed at the Moscow ``Engines-98'' exhibition, bearing several company logos, one being that of the UMPO (which see). It was described as being a scaled-down AL-31F, though the HP spool had only six stages and both turbines were single-stage. Four versions were planned:

AL-55

Basic turbofan, for trainers and business jets. Initial TBO target 2,000 h.

AL-55

With vector-controlled nozzle, for fighter trainers. In data below called AL-55V.

AL-55F

With augmenter (afterburner), for light twin-engined fighters.

AL-55F

With vector-controlled nozzle, here called AL-55FV. Described as suitable for modernised MiG-21 (but this would need four or five engines!).

The AL-55 is a company venture, being developed with money received for exported engines. In the first

instance it is intended to replace the Larzac in the MiG-AT. The engine on display at the exhibition was a mockup but, by the end of 1998, Lyul'ka-Saturn had intended to build five engines, two for bench test, two for flight test and one reserve. The following details were disclosed in late 1998, for the engine as it was planned at that time:

Type

Two-shaft bypass turbojet (F and FV versions, with afterburner).

Inlet

Fixed, 18 fixed radial struts, diameter 452 mm (17.8 in).

LP Compressor

Four stages. Mass flow (all versions) 29.8 kg (65.7 lb)/s. Pressure ratio 3.12. BPR 0.552.

HP Compressor

Six stages. OPR 21.0.

Combustion Chamber

Annular, with downstream burners.

HP Turbine

Single stage. Maximum TET 1,798 K (1,525ºC).

LP Turbine

Single stage.

Nozzle

Fixed, mixed core/bypass flows, in military applications fitted with thrust-vector control (F and FV versions) variable, multi-flap.

Dimensions

Length:	
AL-55	1,210 mm (47.64 in)
AL-55V	1,340 mm (52.76 in)
AL-55F	2,520 mm (99.21 in)
AL-55FV	2,570 mm (101.18 in)
Diameter:	
AL-55, AL-55F	560 mm (22.05 in)
AL-55V	630 mm (24.80 in)
AL-55FV	655 mm (25.79 in)

Weight, Dry

AL-55	355 kg (783 lb)
AL-55V	365 kg (804.7 lb)
AL-55F	425 kg (936.95 lb)

AL-55FV	445 kg (981.04 lb)

Performance Rating

Max T-O, S/L, static:	
AL-55, AL-55V	8.9 kN (2,000 lb st)
AL-55F, AL-55FV	13.35 kN (3,000 lb st)
Cruise, 11,000 m; 36,090 ft; M0.8:	
AL-55	1.96 kN (441 lb)

Specific Fuel Consumption

Max T-O as above:	
AL-55, AL-55V	21.51 mg/Ns (0.71 lb/h/lb st)
AL-55F, AL-55FV	46.73 mg/Ns (1.65 lb/h/lb st)
Cruise 11,000 m; 36,090 ft; M0.8:	
AL-55	25.77 mg/Ns (0.91 lb/h/lb st)

Contract Price

Predicted, for production AL-55	US$500,000-600,000

At the MAKS-99 (Moscow airshow) a far more convincing AL-55 was displayed, again bearing the logos of the three partners. It was at once evident that the engine has been redesigned, to give greater power. The AL-55 now has the same number of compressor and turbine stages as the AL-31, and is described as being a 50 per cent linear scale of the fighter engine.

Described as a `multi-functional gas turbine', the AL-55 is still to be planned in the same four versions as described above, but the thrust ratings were updated to:

AL-55, AL-55V

T-O rating 19.61 kN (4,409 lb st)

AL-55F, AL-55FV

T-O rating 29.42 kN (6,614 lb st)

At the 2000 Farnborough airshow these figures were further modified, and a new mockup image published (reproduced here). It was also made clear the Lyul'ka Saturn is the developing agency and Ufa (UMPO) the manufacturer. New figures are:

Mass flow (all) 29.5 kg (65.04 lb)/s. Specific fuel consumption, as given above for the four versions.

Dimensions

Length:	
AL-55	1,210 mm (47.64 in)
AL-55V	1,340 mm (52.76 in)
AL-55F	2,520 mm (99.21 in)
AL-55FV	2,590 mm (101.97 in)
Diameter:	

AL-55, AL-55F	590 mm (23.23 in)
AL-55V, AL-55FV	620 mm (24.41 in)

Weight, Dry

AL-55	315 kg (694 lb)
AL-55V	325 kg (716.5 lb)
AL-55F	385 kg (848.8 lb)
AL-55FV	405 kg (892.9 lb)

Performance Ratings

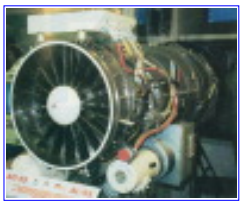
Max T-O, S/L, static:	
AL-55	21.78 kN (4,894 lb st)
AL-55V	21.57 kN (4,850 lb st)
AL-55F, AL-55FV	34.34 kN (7,716 lb st)

Further details are awaited. Lyul'ka Saturn and its partners have set a target price of US$600,000 for a baseline production AL-55 configured for installation in the MiG-AT. The engine has been specifically designed to fit in the engine bay of that aircraft, which was originally tailored to the French Larzac engine. The target price is, of course, significantly below the price quoted to MIG-MAPO for the French engine. The first complete AL-55 was expected to go on test in the first quarter of 2000.

UPDATED

Two views of the AL-55F mockup (Yefim Gordon) (1999)

Two views of the AL-55F mockup (Yefim Gordon) (1999)



The definitive AL-55 on display at MAKS-99 (2000)

AL-55 (2000)

AL-55 at MAKS-2000 (Yefim Gordon) (2001)

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 04 July 2001

Jane's Aero-Engines 10

LYUL'KA SATURN INC

AL-34

This turboprop was planned by the former A M Lyul'ka design bureau to be the most economical in the world. The first brochure was prepared in 1988. The AL-34 is a major project of today's Lyul'ka Saturn Inc.

Intended for light multipurpose aeroplanes and helicopters, its original form featured two independent power sections (called modules) mounted on a common reduction gear to drive tandem coaxial propellers. Each module incorporated a regenerator, which could be bypassed at take-off and landing. The AL-34 can also be supplied as a single power module driving a simplified gearbox to a single propeller.

Lyul'ka Saturn Inc describe this engine as reliable, simple, suitable for hot/high operation and with great capacity for growth. Published life-cycle times are: initial guaranteed, to first overhaul, 2,000 hours; subsequent, to first overhaul, 5,000 hours; full designated life cycle, 10,000 hours.

In 2000 this engine was still under development. It is to be certificated to FAR-33 and JAR-E. It is being considered for several Russian aircraft, including the T-108, M-101T Gzhel, Molniya-100 and -200, Vityaz and Geraklit. It has been announced as the selected engine for the Sukhoi S-86, which requires a pusher installation.

In January 2001 it was stated that Lyul'ka Saturn required US$22 million to complete development, and was ``continuing work, with a foreign investor''. Versions for surface application were being studied with Gazprom and others.

Type

Single or twinned free-turbine turboprop or turboshaft.

Intake

Each power section has an annular (radially inwards) intake protected by a mesh screen, at the back (at the front in a pusher installation). In a tractor installation the engine inlets would be in a plenum chamber fed by a ram

inlet underneath the front of the cowling.

Compressor

Axial straightener followed by single centrifugal stage.

Regenerator

Close-matrix drum with air flow radially inwards and hot gas flow radially outwards. Valves bypass regenerator on pilot command.

Combustion Chamber

Reverse-flow annular, with downstream burners in the front of the flame tube which curves inwards to reverse flow a second time into the turbine nozzle.

Turbines

Single-stage compressor turbine. Single-stage free power turbine. `New high-temperature materials for blades and discs, and optimum cooling system with controlled air supply`.

Jetpipe

From the regenerator the hot gas is collected in a surrounding drum and ejected downwards through a short pipe.

Output

A tubular output shaft from each power turbine carries a pinion at its front end driving a two-stage reduction gear with coaxial output shafts. Additional reduction gear required for turboprop versions.

Accessories

Mounted on the rear faces of the main gearbox.

Control System

Hydromechanical, with FADEC under development.

AL-34-1 single version

Dimensions

Length	1,609 mm (63.35 in)
Width	640 mm (25.20 in)
Height	676 mm (26.61 in)

Weight, Dry

Turboprop	178 kg (392.04 lb)

Performance Ratings

T-O	745.7 kW (1,000 shp)
Cruise, 5,180 m, 17,000 ft; M0.87	410 kW (550 shp)

Specific Fuel Consumption

Cruise, as above	59.1 µg/J (0.350 lb/h/shp)

AL-34-2 twinned version

Dimensions

Length, overall	2,330 mm (91.7 in)
Length, ignoring propeller shafts	1,700 mm (66.9 in)
Width	1,270 mm (50.0 in)
Diameter of each power section	600 mm (23.6 in)

Weight, Dry

Without aircraft accessories	400 kg (882 lb)
Single power section	150 kg (331 lb)

Performance Ratings

T-O, S/L, static	809 kW (1,085 shp)
T-O, OEI	514.5 kW (690 shp)
Climb (4,600 m; 15,090 ft)	809 kW (1,085 shp)
Cruise (5,200 m; 17,060 ft, 500 km/h; 311 mph)	809 kW (1,085 shp)
Cruise (11,000 m; 36,090 ft, 465 km/h; 289mph)	447.4 kW (600 shp)

Specific Fuel Consumption (as above)

Cruise (5,200 m)	57.9 µg/J (0.343 lb/h/shp)
Cruise (11,000 m)	53.6 µg/J (0.317 lb/h/shp)

Contract Price

2001 estimate for production AL-34, US$200,000-240,000.

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Simplified section through AL-34

Mockup of AL-34 single power section (2001)

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 04 July 2001

Jane's Aero-Engines 10

LYUL'KA SATURN - LYUL'KA SATURN INC

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First Deputy, Managing Director: Valeriy A Lebedev
First Deputy, Chief Designer: Yevgeniy Yu Marchukov
Chief Designer: Mikhail Goykhenberg
Head of External Relations: Viktor G Nesterov
(*Tel:* (+7 095) 283 13 74)
Deputy, External Relations: Yevgeniy A Andreyarkin

Little known in the West, Arkhip Mikhailovich Lyul'ka was one of the great pioneers of the turbojet. From the start, in 1936 at the Kharkov Aviation Institute, he did calculations and component tests only on axial-compressor designs. In 1938, in Leningrad (St Petersburg), he began work on the VRD-1 turbojet, but the first actually to run, during the long siege of the city in 1941-43, was the VRD-2, of 6.86 kN (1,543 lb st). In 1944 the TR-1 ran at 12.75 kN (2,866 lb st), and in 1947 this engine passed its State Acceptance and powered the Il-22 four-engined bomber. In 1945 Lyul'ka was appointed Director of the OKB attached to aero-engine GAZ (State Aviation Factory) No 165, at the address above. By this time the design had been completed of the VRD-3, or TR-3, which later ran at 45.11 kN (10,140 lb st). In 1950 Lyul'ka became a General Constructor, and the TR-3 became the AL-5. His subsequent engines appear in separate entries.

After Lyul'ka's death in 1984, GAZ No 165 was named after him, and the General Director, Dr

Chepkin, also named the establishment Saturn. In addition to the engines listed below, this bureau is responsible for the RTWD-14 auxiliary power unit of the Buran spacecraft, the D-57 reusable LH_2/LO_2 rocket engine for space vehicles (40 t thrust, engine life 800 s) and the TP-22 engine, which drives hydraulic pumps on the Energiya vehicle, running on GH_2 from main engine cooling (20 to 150 kW, 26.8 to 201 shp). Total production of Lyul'ka Saturn military turbojet and turbofan engines exceeds 12,000.

In 1982 NPO Saturn was linked with MKB Granit (which see). On 8 April 1997, the company signed an agreement with Rolls-Royce for the joint conversion of its large jet engines (AL-31 derived) for power generation, mainly to drive gas pipelines. Dr Chepkin said the result would be ``a globally competitive engine''. The resulting AL-31ST now also involves partners in Italy and the Czech Republic.

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 01 May 2001

Jane's Aero-Engines 10

RYBINSK MOTORS JSC

RD-36-41

Just as the Sukhoi T-4 supersonic strategic bomber was inspired by the USAF's B-70, so was its engine inspired by the General Electric J93, the engine of the B-70. It was to `mislead the enemy' that this engine was given a designation in the prolific `RD-36' family, though the RD-36-41 was totally unrelated to the RD-36-35, -51A and 51V.

As the engine of an aircraft intended to fly at Mach 3 (3,200 km/h, 1,988 mph), the RD-36-41 posed enormous challenges. Partly because the entire T-4 project had many critics, as well as rival Tupolev, the opportunity was taken to take as many short-cuts as possible, without compromising safety, and P A Kolesov decided that several J93 features were simply copied. Among these were the design of the compressor, the use of brazed pipe joints and the packaging of accessories in a fireproof box.

Thanks to massive support from CIAM, the engine ran on the testbed in 1965, and a modified version began air testing under a Tu-16LL in 1968. The first engine was delivered for the T-4 in 1969, but the first flight of that four-engined aircraft did not take place until 22 August 1972. The T-4 programme was abandoned in March 1974. A total of 29 engines was delivered.

Type

Single-shaft turbojet afterburner.

Compressor

Ten-stage axial with first four and last four stages variable. Mass flow 165 kg (363.76 lb)/s. Pressure ratio 9.7. Mass flow in M3 cruise (see below) 95.8 kg (211.2 lb)/s.

Combustion Chamber

Can-annular with 16 vaporising burners. Cleared to use special RG-1 Natfil fuel, thermostable after prolonged coking.

Turbine

Two axial stages with inserted aircooled blades. Maximum TGT 1,057ºC.

Nozzle

Short afterburner with three spray rings and variable 16-flap convergent/divergent nozzle.

Accessories

Grouped in fireproof box under engine inlet, driven by shaft in 6 o'clock inlet strut.

Dimensions

Length	4,690 mm (184.65 in)
Inlet diameter	1,220 mm (48.03 in)

Weight

Dry	2,850 kg (6,283 lb)

Performance Ratings

T-O, S/L:	
Max dry	120.6 kN (27,116 lb st)
Max afterburner	166.7 kN (37,478 lb st)
20 km, M3:	
Max afterburner cruise	75.0 kN (16,865 lb)

Specific Fuel Consumption

Max dry, as above	26.88 mg/Ns (0.95 lb/h/lb st)
Max afterburner	53.76 mg/Ns (1.9 lb/h/lb st)

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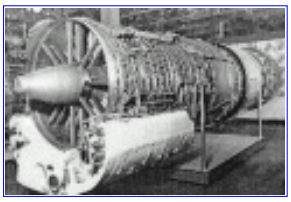

RD-36-41 (2001)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 01 May 2001

Jane's Aero-Engines 10

ST PETERSBURG NPO IM KLIMOV

RD-43, VKS

Derived from the RD-33, this two-shaft turbofan has been described as a `fourth-generation engine'. Designed for retrofitting to all versions of MiG-29, it is also to be marketed in an unaugmented version. The designation VKS stands for Vladimir Klimov, Sarkisov.

VKS-5

Unaugmented version, with T-O rating of 51.46 kN (11,574 lb st).

VKS-10

Augmented version, for MiG-29 retrofit and other supersonic applications.

Full details are not yet (August 2000) available, but Klimov states that funding of the afterburning engine is assured. All versions have FADEC control, and are cleared to the very high maximum TET of 1,527ºC. The VKS-10 nozzle is similar to that of the RD-133. The requirement for the VKS-10 arose because of growth in weight of successive MiG-29 versions. According to an unofficial account the VKS-10 version was to be ready for service before the end of 2000, but this was denied by a Klimov spokesman, who said that much testbed running remained to be done.

In February 2001 an unofficial report (by *Jane's* correspondent Piotr Butowski) suggested that maximum thrust might be as high as 118 kN (26,556 lb st). It called the engine the VK-10.

Dimensions

VKS-5 length	2,900 mm (114 in)
VKS-10	similar to RD-33 and RD-133

Weight, Dry

VKS-5	885 kg (1,951 lb)
VKS-10	lighter than RD-33

Performance Ratings (T-O, S/L)

VKS-5	51.46 kN (11,574 lb st)
VKS-10	98.01 kN (22,057 lb st)

Specific Fuel Consumption (T-O, S/L)

VKS-5	20.96 mg/Ns (0.74 lb/h/lb st)
VKS-10	50.98 mg/Ns (1.08 lb/h/lb st)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 01 May 2001

Jane's Aero-Engines 10

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

R28V

Originally designated R-27VM-300, this lift/cruise engine was developed at AMNTK `Soyuz' under Favorski to power the Yak-38M. The mass flow was increased. See under R27V-300.

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

OMSK ENGINE DESIGN BUREAU

TVD-10

Glushenkov directed the development of this turboprop in the 1960s, to power the Beriev Be-30 multirole light transport. It was based on the GTD-3 turboshaft, which by that time was in production. As the power turbine was at the rear, the drive had to be taken through a rear gearbox to a high-speed shaft passing across the top of the engine (Glushenkov studied the way this was done in de Havilland's turboprop version of the Gnome). An epicyclic gearbox at the front, above the intake, then provided an output at 1,590 rpm for the 3.0 m (118 in) VISh AV-24B/M three-blade propeller.

Development was protracted, and Be-30 prototypes flew with the ASh-21 piston engine and imported French Astazou turboprops. The TVD-10 was eventually qualified in 1970 at 686 kW (920 shp), but the Be-30 did not go into production.

TVD-10M

T-O power 810 kW (1,086 shp). Fitted to small batch of Be-32 local-service transports. The Be-32 was rejected by Aeroflot in favour of the Antonov An-28, but this took so long to develop that Aeroflot switched to the Czech L-410. Development of the An-28 continued. In 1978, for political reasons, this was transferred to Poland, complete with its power plant. Accordingly the TVD-10 programme was transferred, together with the AV-24AN and AV-25B propellers. See TWD-10 under PZL Rzeszów.

TVD-10A

OMKB never entirely stopped working on the TVD-10, and this Arctic version was an active programme in the early 1990s for the An-28A for Polar service. There has been no recent news of this, nor confirmation of the report that the An-28 would be put into production in Russia (Siberia) by the Novosibirsk Industrial Aviation Association.

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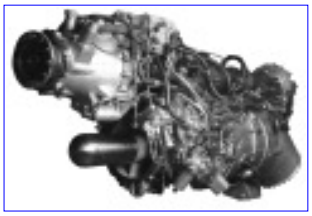

TVD-10 (2001)

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

MOTOR - MOTOR, GNPP (STATE SCIENTIFIC PRODUCTION ENTERPRISE)

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General Designer: Aleksandr F Ivakh

This design organisation was formed by the MAP in September 1955 as OKB-26. It is co-located at the enormous factory now known as UMPO (which see). Most of OKB-26's early work was concerned with the production of different versions of RD-9 turbojet (see under AMNTK 'Soyuz'), followed by the development of derived engines. The first sub-family assigned to OKB-26 was the R11K series of engines for all versions of pilotless La-17 (see under Soyuz R11-300).

Full design capability was established in 1962 under Sergei A Gavrilov, who directed the development of all the engines described hereafter. He was succeeded by Aleksey A Ryzhkov, who was followed by Ivakh in 2000. GNPP 'Motor' claims to have worked on "70 types of engine, of which 21 have been produced in series".

The design office was given the name 'Motor' in 1992. In addition to the turbojets described hereafter GNPP 'Motor' also produces gas turbines rated at up to 10 MW for surface-power applications, as well as the UMZ family of small piston engines.

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

MACHINE-BUILDING DESIGN BUREAU 'GRANIT'

GRANIT TVD-150

The TVD-150 is the turboprop version of a new (2000-era) family of simple gas turbines which are also being developed as jet and turboshaft engines, the latter also including ground power units in the 100 kW category. Output shaft speeds vary from 2,000 to 8,000 rpm. The following refers to the turboprop, which Granit states has been selected to power the D-12 Lark (an aircraft not known to Editor). The specific fuel consumption is as reported, but appears unbelievably high.

Type

Free-turbine turboprop.

Compressor

Single-stage centrifugal. Air enters from rear and flows forward to reach compressor.

Combustion Chamber

Straight-flow annular. Cleared for all hydrocarbon fuels and LNG.

Compressor Turbine

Single stage.

Power Turbine

Single stage.

Jetpipe

Vertically downwards.

Output

Epicyclic reduction gear at front to drive four-blade propeller.

Weight

Dry	50 kg (110 lb)

Performance Rating (S/L)

T-O	118.5 kW (150 shp)

Specific Fuel Consumption

T-O, as above	186.2 µg/J (1.1 lb/h/shp)

TVD-150 (2001)

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AERO-ENGINES - TURBOSHAFT, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

OMSK ENGINE DESIGN BUREAU

GTD-3

This engine was the first to be assigned to the new Omsk design bureau in the late 1950s. The requirement was to produce an engine for the Kamov Ka-25 helicopter for the AV-MF (Soviet naval aviation). Led by V A Glushenkov (whose name is omitted from all today's OMKB literature), the engine ran in 1959, the Ka-25 prototype appeared at the Tushino display in 1961, and flight testing was completed in 1964. Series production began at Omsk Baranov (which see) in 1966, and in 20 years the factory delivered just over 1,000 engines. Subsequently the core was used as the basis for the TVD-10 turboprop, described in a separate entry.

GTD-3

Initial version, T-O rating 559 kW (750 shp).

GTD-3F

Main series version. T-O rating (5 min) 671 kW (900 shp). Fitted to almost all Ka-25 helicopters as built. Details as given below.

GTD-3M

Fitted with RV-3M reduction gear, enabling T-O power to be increased to 746 kW (1,000 shp).

GTD-3BM

Improved long-life engine. T-O rating 738 kW (990 shp). Retrofitted to most surviving Ka-25 helicopters.

PZL-10W

Designation of version produced under licence in Poland (see under PZL-Rzeszów).

Type

Free-turbine turboshaft.

Compressor

Six axial stages followed by one centrifugal. Even the centrifugal impeller has inserted blades. Mass flow 4.5 kg (9.92 lb)/s at 28,800 rpm. GTD-3M, 4.65 kg (10.25 lb)/s. OPR 6.5.

Combustion Chamber

Annular, with auxiliary burners in two starting units.

Compressor Turbine

Two stages. TGT 869ºC.

Power Turbine

Single stage, with rear drive.

Reduction Gear

Spur gear at rear, type RV-3F.

Dimensions

Length	2,295 mm (90.35 in)
Width	900 mm (35.43 in)
Height	580 mm (22.83 in)

Weight

Dry	240 kg (529 lb)

Performance Ratings

(S/L, ISA)

T-O	671 kW (900 shp)
Cruise	357 kW (479 shp)

Specific Fuel Consumption

T-O, as above	50.7 µg/J (0.300 lb/h/shp)

Cruise, as above	63.88 µg/J (0.378 lb/h/shp)

NEW ENTRY

Cutaway GTD-3 (2001)

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AERO-ENGINES - TURBOSHAFT, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

ST PETERSBURG NPO IM KLIMOV

TV7-117V (VK-3000)

This turboshaft engine is based on the core of the TV7-117S turboprop described in a separate entry. The following are distinguishing features.

Compressor

Mass flow 9.2 kg (20.28 lb)/s. OPR 17.0.

Compressor Turbine

TGT 1,237ºC.

Jetpipe

Twin pipes angled out at 60º on each side.

Output

Primary gearbox giving front drive for Mil Mi-38 and Mi-38Z and rear drive for Kamov Ka-52.

Control System

FADEC, type BARK-65.

Dimensions

Length	1,780 mm (70.08 in)
Width	635 mm (25.0 in)

Height	727 mm (28.62 in)

Weight

Dry	360 kg (793.7 lb)

Performance Ratings

(S/L, static)

30 s	2,796 kW (3,750 shp)
2.5 min	2,610 kW (3,500 shp)
30 min	2,237 kW (3,000 shp)
Max continuous	2,088 kW (2,800 shp)
Cruise	1,342 kW (1,800 shp)

Specific Fuel Consumption

Max continuous	74.13 µg/J (0.439 lb/h/shp)
Cruise, as above	83.83 µg/J (0.496 lb/h/shp)

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 12 April 2001

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TUMENSKIE - OAO (JOINT STOCK COMPANY) TUMENSKIE MOTORSTROITELY

14 Tumen, 625014 Russia
Tel: (+7 3452) 22 83 73
Fax: (+7 3452) 21 48 91

General Director: Viktor G Kulchihin

This is one of the large production factories, known in Russia as a `machine-building enterprise'. Among a wide range of non-aeronautical products - such as automotive transmissions, cross-country vehicles and high-pressure pumps - it was the production source of the RU-19-300 and R27V-300 (both, see under AMNTK Soyuz).
Note: Do not confuse with designer S K Tumanskiy.

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

RYBINSK MOTORS JSC

RD-36-35

P A Kolesov's team at Rybinsk was picked, without competition, to produce the Soviet Union's specialised lift jets, for near-vertical installation. This engine was produced in several versions.

RD-36-35

Baseline engine, first tested in 1964. Total production by 1967, 62. Installed in MiG-21PD Type 23-31 (two engines in tandem) and Sukhoi T-58VD (row of three). TGT 957ºC. Mass flow 40.5 kg (89.3 lb)/s. Weight 176 kg (388 lb). S/L rating 23.49 kN (5,180 lb st). Sfc 37.65 mg/Ns (1.33 lb/h/lb st).

RD-36-35K

Booster for space vehicles, notably MiG 'Spiral'. Tested 1966. S/L rating 23.49 kN (5,180 lb st). Sfc 39.06 mg/Ns (1.38 lb/h/lb st).

RD-36-35T

Take-off booster for Tu-22 supersonic bomber (retrofit). Horizontal installation. In production 1966. S/L rating 24.21 kN (5,511 lb st). Sfc 39.06 mg/Ns (1.38 lb/h/lb st).

RD-36-35BF

Refined lift jet for Yak-36M, Yak-38 and MiG-21PD. S/L rating 29.0 kN (6,393 lb st).

RD-36-38FV(R)

Fully rated lift engine for Yak-38M. Total of 575 produced at Rybinsk 1972-89. Described below.

RD-36-35PR

Lift engine for Beriev VVA-14 and for Ekranoplans.

Type

Single-shaft turbojet for vertical or horizontal installation.

Compressor

Six stages. Mass flow 45.3 kg (99.87 lb)/s. Pressure ratio 4.4.

Combustion Chamber

Annular.

Turbine

Single stage. TGT 1,067ºC.

Weight

Dry	201.5 kg (444 lb)

Performance Rating

S/L	30.5 kN (6,725 lb st)

Specific Fuel Consumption

	25.0 mg/Ns (0.883 lb/h/lb st)

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RD-36-35 (2001)

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-110

In 1980 the Kuznetsov technical staff, led by V S Osipov, began detail design of a turboprop with a new pusher configuration. The basic gas turbine was treated as a generator of a flow of gas to drive an LP turbine and pusher propeller system mounted behind it. The elements of the gas generator were almost entirely new. The gas generator went on test in December 1988. So far not enough money has been made available to take testing of the complete engine to certification.

Type

Three-shaft turboprop, with gas generator driving independent pusher propellers.

LP Compressor

Seven stages, with variable inlet guide vanes.

HP Compressor

Nine stages, with variable inlet guide vanes.

Combustion Chamber

Annular, with main and cruise rings of burners.

HP Turbine

Single stage, with single-crystal blades.

IP Turbine

Single stage.

LP Turbine

Three stages.

Jetpipe

Large diameter annulus surrounding reduction gear and propeller hubs.

Propeller

Contrarotating four-blade propellers of 4.7 m (15 ft 5 in) diameter, with special extended blade roots with aerodynamic-profile cuffs of refractory material.

Weight, Dry

Without propeller	2,300 kg (5,071 lb

Performance Ratings

T-O, S/L, static:	
Thrust	176.5 kN (39,683 lb st)
Power	15,883 kW (21,300 shp)
Cruise (11,000 m 36,090 ft; M 0.75):	47.64 kN (6,614 lb)

Specific Fuel Consumption

T-O, as above	5.384 mg/Ns (0.19 lb/h/lb st)
Cruise, as above	12.469 mg/Ns (0.44 lb/h/lb)

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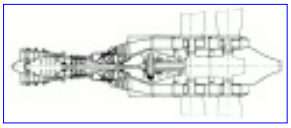

Longitudinal section through the NK-110 (2001)

The NK-110 gas generator, seen from the gearbox end (top) and from the inlet end. Note the extended nose bullet (Samara)
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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

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ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-56, NK-64

In 1979 the Kuznetsov bureau decided to begin work in the field of high-BPR turbofans for heavy transport aircraft. Funding was provided for the NK-56 to power the Il-96, and the smaller but more efficient NK-64 to power the IL-96-300 and Tu-204. In each case the PS-90 was the competitor engine.

The NK-56 first ran on the testbed in July 1980, and the NK-64 followed in April 1984. Director of both programmes was A I Yelizarov, and chief designer V S Osipov. There was every indication that these would have been good engines, but talent and money had to be shared with the NK-62 and NK-110 turboprops, which in some ways were more promising and had no competitor. In the end the immediate large turbofan market was left to the PS-90. In May 1983, the turbofan design teams were transferred to the even more promising NK-93, which remains an active (if under-funded) programme to this day. Brief details of the NK-56 and NK-64 are given here for the record.

Type

Three-shaft turbofans.

Fan

Single stage. BPR, (NK-56) 4.9, (NK-64) 4.1.

Compressor

HP and LP compressors with total of 15 stages. OPR (NK-56) 25.5, (NK-64) 27.6.

Combustion Chamber

Annular with single burners.

Turbine

Single-stage HP, single-stage IP, three-stage LP. TGT (NK-56) 1,298ºC.

Reverser

Cascade and blocker-door type.

Dimensions

Fan diameter:	
NK-56	2,050 mm (80.71 in)
NK-64	1,860 mm (73.23 in)

Weight, Dry

NK-56	3,340 kg (7,363 lb)
NK-64	2,850 kg (6,283 lb)

Performance Ratings

T-O, S/L:	
NK-56	176.5 kN (39,683 lb st)
NK-64	156.9 kN (35,273 lb st)
Cruise (11,000 m, 36,089 ft, M 0.8):	
NK-56	35.3 kN (7,936.5 lb)
NK-64	34.32 kN (7,716 lb)

Specific Fuel Consumption

T-O, as above:	
NK-56	10.86 mg/Ns (0.383 lb/h/lb st)
NK-64	10.49 mg/Ns (0.370 lb/h/lb st)
Cruise, as above:	
NK-56	20.98 mg/Ns (0.74 lb/h/lb)
NK-64	16.44 mg/Ns (0.58 lb/h/lb)

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NK-64 (Samara) (2001)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

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ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-25

The NK-23, rated at 215.76 kN (48,500 lb st) with maximum afterburner, was tested in July 1976, and actually powered at least one Tu-22M (according to the Samara OKB, designated Tu-22MYe), but it was discontinued in 1977 in favour of the markedly superior NK-25. This engine had been designed in 1972-74, and was instrumental in enabling the Tupolev design bureau to develop the Tu-22M-3, with significantly greater capability.

Supervisor of the NK-25 programme was A A Ovcharov, and Lead Designer A I Lotsman. Among the big team of departmental chief designers, that responsible for the new control system was A P Anisimov.

The NK-25 first ran in 1974, was flight-tested in a pod carried under a Tu-142LL and entered production at OAO Samara in 1976. The Tu-22M-3 began flight testing, with NK-25 engines, on 22 June 1977. Production was limited by the first SALT treaty to 30 aircraft per year, and in any case funding ran out and manufacture tapered off in the early 1990s with a total of 268 of this version delivered, plus a small number of Tu-22MR reconnaissance aircraft and flying testbeds. Production of new NK-25 engines was about 680. Efforts are being made to update the aircraft, and to keep as many as possible airworthy, especially for foreign customers.

Type

Three-shaft augmented turbofan.

Intake

Fixed geometry, fabricated in steel, with 18 narrow-chord radial struts, all with hot-air anti-icing. No interspersed guide vanes.

Fan

Three stages, first stage with straight rotor blades with part-span snubbers. Mass flow 339 kg (747 lb)/s. Bypass ratio 1.45.

IP Compressor

Five stages, with variable inlet guide vanes.

HP Compressor

Six stages, based on that of NK-22. OPR 25.9.

Combustion Chamber

Annular, similar to that of NK-22.

HP Turbine

Single stage, with single-crystal blades. TGT 1,324ºC (1,597ºK).

IP Turbine

Single stage.

LP Turbine

Two stages

Afterburner

Geometrically similar to that of NK-22, with ceramic-coated inner liner. Improved aerodynamics and new nozzle with 18 reprofiled flaps.

Control Systems

FADEC, with duplicated digital electronics in fireproof boxes.

Accessories

Grouped on underside of compressor case, driven from front end of LP shaft.

Dimensions

Length	5,200 mm (205 in)
Fan diameter (1st stage)	1,355 mm (53.35 in)
Envelope diameter	about 1,500 mm (59 in)

Weight

Dry	about 3,000 kg (6,614 lb)

Performance

(S/L, Static)

Max T-O, with afterburner	245.2 kN (55,155 lb st)
Normal (military, max dry)	186.4 kN (41,900 lb st)

Specific Fuel Consumption

Max T-O	58.94 mg/Ns (2.08 lb/h/lb st)

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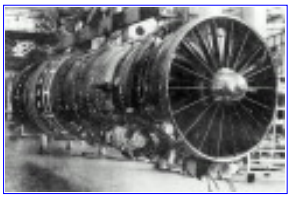

NK-25 (Samara)
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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-62

In the 1970s the N D Kuznetsov design bureau worked on numerous projects for large engines offering high propulsive efficiency. One of the few to reach the stage of complete engine testing was this turboprop. It remains the most powerful turboprop ever to have run. The complete prototype engine was on outdoor test from December 1982 until 1990, and completed two 100-hour tests. Figures for shaft power have not been published. The propeller comprised two counter-rotating four-blade units with a diameter of 4.7 m (15 ft 5 in).

NK-62M

The NK-62 was intended to be modified to this standard, designed in 1985. Take-off thrust would have been 285.2 kN (64,080 lb st) and cruise sfc 12.75 mg/Ns (0.45 lb/h/lb). Dry weight was estimated at 4,850 kg (10,692 lb). It was estimated that by 1993 T-O thrust could have been developed to 313.9 kN (70,547 lb st). In 1989 the NK-62M was abandoned. Work was transferred to the NK-93 and NK-110.

Type

Three-shaft turboprop.

Intake

Fixed-geometry annular surrounding reduction gear.

LP Compressor

Two-stage fan and single-stage core supercharger.

IP Compressor

Five stages

HP Compressor

Seven stages.

Combustion Chamber

Annular

HP Turbine

Single stage.

IP Turbine

Single stage.

LP Turbine

Three stages.

Weight, Dry

Without propeller	4,200 kg (9,259 lb)

Performance Ratings

T-O, S/L:	245 kN (55,067 lb st)
Cruise (11,000 m, 36,090 ft; M 0.75)	44.1 kN (9,920 lb)

Specific Fuel Consumption

T-O	8.16 mg/Ns (0.288 lb/h/lb st)
Cruise, as above	13.6 mg/Ns (0.48 lb/h/lb)

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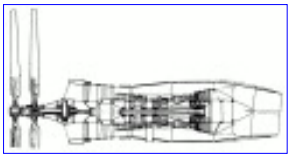

NK-62 (2001)

NK-62 on outdoor test (2001)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-86, NK-87

In 1973, a new engine was derived at Kuibyshyev from the NK-8 family to power the Ilyushin Il-86 wide-body transport. As was often the case in the Soviet Union the engine designation was the same as that of the aircraft. The design was supervised by N D Pechenkin, and Leading Designer was O F Filippov. The engine first ran in July 1974, and the first four flight-cleared NK-86 engines powered the Il-86 prototype on its first flight on 22 December 1976. The engine was certificated in April 1979 and was subsequently produced by KMPO (which see).

The aircraft had a protracted development, mainly not due to the engine, and though 20 aircraft had flown by 1981 it was not until that year that limited Aeroflot service began. The design engine life was 10,000 hours, with major overhauls at 4,000 hours. Production engines passed through several modification programmes, one change being the introduction of an analogue electronic control system with diagnostic functions.

NK-86A

Improved engine, cleared to slightly higher rotational speeds and temperatures. First run in August 1983 and certificated in August 1985. Entered service in 1987, with target life of 10,000 hours. Guaranteed life to first overhaul 3,000 hours (1,500 cycles). Modified with single-crystal HP turbine blades made of ZhS-30 material. This engine remains in service in all versions of Il-86.

NK-87

Derivative engine for *Lun* and *Spasatel* Ekranoplans. Greatly simplified, but fitted with tilting propulsive nozzle.

First tested January 1983, certificated July 1986. Design life 20,000 hours (7,500 cycles).

Type

Two-shaft bypass turbojet (low-BPR turbofan).

Intake

Fabricated with 12 radial struts; no intermediate short guide vanes as in NK-8 engines.

Fan

Two stages, both with solid blades fitted with part-span clappers. Pressure ratio 2.23. Mass flow 288 kg (635 lb)/s. BPR 1.18 (NK-87, 1.17).

IP Compressor

Three stages on LP shaft to supercharge core.

HP Compressor

Six stages. Pressure ratio 12.93 (NK-87, 13). OPR 28.8.

Combustion Chamber

Annular, improved life and aerodynamics.

HP Turbine

Single stage. Entry gas temperature (86) 899ºC (1,172ºK) normal, 987ºC maximum, (86A, 87) 1,007ºC (1,280ºK).

LP Turbine

Two stages.

Jetpipe

Core mixer upstream of blocker/cascade reverser; NK-87 has plain jetpipe pivoted about horizontal axis.

Control Systems

Analogue electronic; NK-87 has single control system governing eight engines providing 100 tonnes thrust.

Dimensions

Length:	
NK-86, NK-86A, basic engine	3,638 mm (143.2 in)
NK-86, NK-86A, with reverser	5,278 mm (207.8 in)
Diameter of fan (all)	1,455 mm (57.3 in)
Inlet diameter (all)	1,600 mm (63.0 in)

Weight, Dry

NK-86, NK-86A, with reverser	2,750 kg (6,063 lb)
NK-87	2,200 kg (4,850 lb)

Performance Rating

ISA, S/L, static:	
NK-86, NK-87	127.5 kN (28,660 lb st)
NK-86A	130.5 kN (29,343 lb st)
Cruise (11,000 m, 36,090 ft, M0.8):	
NK-86	31.6 kN (7,073 lb)

Specific Fuel Consumption

T-O, S/L:	
NK-86, NK-86A	14.74 mg/Ns (0.52 lb/h/lb)
NK-87	15.02 mg/Ns (0.53 lb/h/lb)
Cruise, as above:	
NK-86, NK-86A	20.97 mg/Ns (0.74 lb/h/lb)

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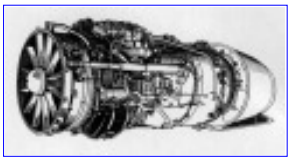
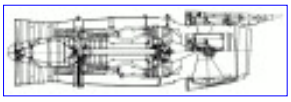


Russian drawing of NK-86A



NK-86 with reverser (2001)

NK-86 (2001)

NK-87 (2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

MOTOR, GNPP (STATE SCIENTIFIC PRODUCTION ENTERPRISE)

R95

This engine was developed from 1975 to power low-level ground-attack aircraft, notably the Su-25. A design requirement was that the entire power plant installation (but not necessarily the aircraft fuel system) should continue to operate in the face of rifle-calibre fire, and to resist ingestion of large birds.

The engine was based on the mature R11F and R13-300. The entire design was modified in detail to make it more robust, with minimal increase in weight. The afterburner was replaced by a plain jetpipe, and operating parameters were modified, with rotational speeds and temperatures slightly reduced. The accessory group, beneath the compressors, was very similar to that of the R-13 family.

Production at Ufa began in 1980. With designation **R95Sh**, this engine powers the Su-25, Su-25UB, Su-25UTG and Su-25BM. From 1987 it was replaced in production by the R195. The GT3-10/95 is a derivative rated at 10 MW for ground electrical power.

Type

Two-shaft turbojet.

LP Compressor

Three stages. Mass flow 64.8 kg (142.9 lb)/s.

HP Compressor

Five stages. Overall pressure ratio 8.7

Combustion Chamber

Can-annular, identical with R-13. Cleared to operate on automotive fuels.

HP Turbine

Single stage. TGT 875ºC (1,148ºK).

LP Turbine

Single stage.

Jetpipe

Fixed-area nozzle, with central tube to suck out cooling air.

Dimensions

Length	3,300 mm (130 in)
Inlet diameter	805 mm (31.7 in)
Overall diameter	914 mm (36 in)

Weight

Dry	990 kg (2,182.5 lb)

Performance Rating

(S/L, static)

Max T-O	40.21 kN (9,039 lb st)

Specific Fuel Consumption

T-O, as above	24.36 mg/Ns (0.86 lb/h/lb st)

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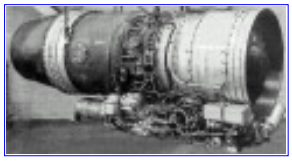

R95Sh (2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

RYBINSK MOTORS JSC

RD-41

This second-generation lift jet was designed for the Yak-41, and was also fitted to the Yak-41M (these aircraft later became known erroneously as the `Yak-141'). The engine was designed in 1982, first ran in 1984, and was delivered to Yakovlev in 1988.

Type

Single-shaft turbojet for near-vertical installation.

Compressor

Seven-stage axial. Mass flow 53.5 kg (117.95 lb)/s. Pressure ratio 6.3.

Combustion Chamber

Annular.

Turbine

Single-stage with aircooled blades. TGT 1,207ºC. Speed 12,500 rpm.

Contol System

Electronic.

Dimensions

Height	1,594 mm (62.76 in)

Diameter	635 mm (25.0 in)

Weight

Dry	290 kg (639.3 lb)

Performance Rating

T-O, S/L	41.00 kN (9,039 lb st)

Specific Fuel Consumption

	39.63 mg/Ns (1.4 lb/h/lb st)

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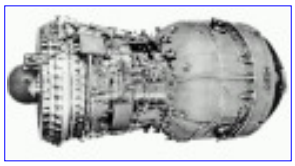

RD-41 (2001)

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

ST PETERSBURG NPO IM KLIMOV

VK-1500

Having successfully collaborated with Motor Sich on the development of all versions of the TV3, Klimov decided in 1999 to invite the Zaporozhe-based partner to collaborate on this derived engine with lateral inlets, a free-turbine drive through the core spool and a centreline primary reduction gearbox at the front. As the designation suggests, this new engine family is to be derated to little more than half the power of the TV3-117 family.

VK-1500

The baseline turboprop version. As the illustration shows, this version has lateral inlets. The engine is being developed as a candidate for the Antonov An-3 and An-38-300 and Beriev Be-32. Certification is targeted for the end of 2001.

VK-1500V

Turboshaft version, intended for Kamov Ka-60 and Ka-62. Specification being refined. The same engine core is being developed for ground electric-power generation.

The VK-1500 turboprop differs from the TV3-117 turboprop version in the following respects.

Intake

Laterally symmetric vertical semi-rectangular inlets feeding air diagonally in on each side.

Compressor

First two stages redesigned. Spool redesigned to permit output shaft to pass along centre. Mass flow 7.3 kg (16.09 lb)/s. OPR 7.4.

Combustion Chamber

Completely new design.

Compressor Turbine

Entry temperature reduced to 914ºC.

Power Turbine

Drives output shaft passing forward along engine centreline.

Output

Epicyclic reduction gear on centreline of engine at front.

Control System

FADEC, type SAU-2000, supplied by OAO 'Star' and NPP 'Ega'.

Dimensions

Length	1,714 mm (67.48 in)
Width	708 mm (27.87 in)
Height	847 mm (33.35 in)

Weight

Dry	340 kg (749.6 lb)

Performance Ratings

T-O (S/L, static)	1,118 kW (1,500 shp)
Cruise (3,000 m; 9,842 ft, 400 km/h; 248 mph)	783 kW (1,050 shp)

Specific Fuel Consumption

T-O, cruise, both as above	85.56 µg/J (0.507 lb/h/shp)

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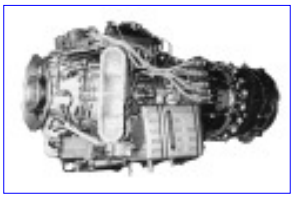

VK-1500 turboprop version (2001)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

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ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-118

In 1998, the Kuznetsov management at Samara authorised a Technical Proposal for an engine to power any future LFI (light front-line fighter) or SZLBS (supersonic light combat aircraft). The only immediate potential application is the Yak-130, though there are other Russian projects in these categories, such as a twin-engined Sukhoi S-54. In late 1999 brief particulars of the resulting NK-118 were published. By 2000 there was no indication that there was funding for the testing of NK-118 hardware.

Type

Two-shaft turbofan, with vectoring nozzle and/or (NK-118F) afterburner.

Fan

Probably two stages. Mass flow 52.3 kg (115.3 lb)/s. BPR 0.813.

Compressor

Probably six stages. OPR 27.5

Combustion Chamber

Annular.

HP Turbine

Single stage with cooled blades. TET 1,507ºC.

LP Turbine

Two stages.

Jetpipe

Incorporates mixer and either afterburner (NK-118F) or vectoring nozzle.

Control system

FADEC.

Dimensions

Not disclosed.

Weight, Dry

Not disclosed

Performance Rating

(S/L, T-O)

NK-118	27.41 kN (6,173 lb st)
NK-118F	41.22 kN (9,259 lb st)

Specific fuel consumption

(S/L, T-O)

NK-118	17.59 mg/Ns (0.62 lb/h/lb st)

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

TV128-300

This entirely new pusher propfan, with small contra-rotating propellers, has been designed for the most efficient propulsion of business and light regional aircraft. It has been developed with a group entitled FPG Aviko-M (unknown to the Editor). No details of its design have yet been made available beyond the following figures:

Dimension

Length overall	1,200-1,750 mm (47.24-68.9 in) depending on installation

Weight

Dry	200 kg (441 lb)

Performance Ratings

T-O (S/L, static, ISA)	969 kW (1,300 hp)
Cruise (12,000 m; 39,370 ft, M0.7)	373 kW (500 hp)

Specific Fuel Consumption

T-O (as above)	78.93 µg/J (0.467 lb/h/shp)
Cruise (as above)	66.25 µg/J (0.392 lb/h/shp)

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TV128-300

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

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AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

R79

This engine was built to a new design as the lift/cruise power plant of the Yak-141. Though development of this supersonic carrier-based VTOL aircraft was suspended in 1991, development of the R79 continued for several further years - according to Mr Kobchyenko, `with funds from sources outside the Russian government'.

By 1993 a total of 12 engines had run about 3,500 hours, including over 500 hours in flight. The engine fitted to the Yak-141 was designated **R79V-300**. Work was also proceeding on the upgraded **R79M**, rated at about 176.2 kN (39,600 lb st), with FADEC control, new combustion chamber, and (for STOL applications) a fixed axi-symmetric nozzle limited to vector angles of 20º up or down for flight-control augmentation.

VK-21

This is the designation of two experimental engines derived from the R79 for propulsion of conventional aircraft. With a fixed noise-suppressing nozzle, the T-O thrust would be 117.66 kN (26,455 lb st). This engine was selected by Sukhoi for the projected S-21 supersonic bizjet.

Type

Two-shaft augmented turbofan with vectoring nozzle.

LP Compressor

Three stages. Mass flow 180 kg (396.8 lb)/s.

HP Compressor

Eleven stages. OPR 22. Bypass ratio 1.0. HP bleed from two stages to provide aircraft hover control power.

Combustion Chamber

Annular, with vaporising burners fed from an inner manifold, giving very low emissions. Non-traditional double-zone design.

HP Turbine

Two stages with air-cooled single-crystal blades. Maximum TGT can briefly exceed the normal limit of 1,347ºC.

LP Turbine

Two-stage, rotating in opposite direction to HP.

Afterburner

Fuel burner rings just behind LP turbine light up in sequence to give fully modulated variable augmentation. Can be used with the nozzle in the 95º position for hovering flight.

Jetpipe

Convergent nozzle with variable primary area. Connected to bypass duct periphery by three tapering-wedge pipe sections which rotate in opposite directions to vector nozzle from 0º (forward flight) to 63º (STO) and 95º (VL and hovering).

Control System

Three-channel electronic, with duplicated hydromechanical units as back-up. Automatically varies engine thrust to trim aircraft in pitch, supplies modulated air to roll and yaw control jets, supplies bleed air to start lift engines, and controls main engine fuel flow and drives to main hydraulic and electric power.

Dimensions

Inlet (fan) diameter	1,100 mm (43.31 in)
Max diameter (external)	1,716 mm (67.56 in)
Length	5,229 mm (205.87 in)

Weight

Dry	2,750 kg (6,063 lb)

Performance Ratings

(ISA, S/L)

Max dry	107.63 kN (24,200 lb st)
Max afterburner	152.0 kN (34,170 lb st)
Max (full aircraft control bleed)	137.3 kN (30,864 lb st)

Specific Fuel Consumption

Max dry, as above	18.70 mg/Ns (0.66 lb/h/lb st)
Max afterburner, as above	45.3 mg/Ns (1.60 lb/h/lb st)

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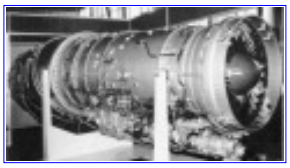

R79V-300

R79V-300 nozzle in 95º position

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

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TMKB (TUSHINSKOYE [TUSHINO] ENGINE DESIGN BUREAU) `SOYUZ'

R29-300

This augmented turbojet was developed from the R-27F2M-300 at the Tushino MKB `Soyuz' under the leadership of Khachaturov. One major difference was to increase the diameter of the first two compressor stages. Production and rebuild/overhaul were handled by the MMZ Red October factory at Ufa named for V V Chernyshov.

It is simpler than the corresponding American F100-100, with fewer compressor stages and a lower pressure ratio; but it is more powerful and costs much less. Different subtypes were fitted to former MiG-23 and MiG-27 versions for former Warsaw Pact front-line use, and to the Su-22. In all these aircraft water injection is used on take-off, the MiG-23MF water tank having a capacity of 28 litres (7.4 US gallons; 6.2 Imp gallons).

The following versions have been identified:

R29-300

Original fully rated production engine for MiG-23MF and related versions. Produced at Ufa as Product 55. In 1999, the arms-export organisation Rosvoorouzheniye was setting up a support facility for MiG-23 engines in Ethiopia.

R29B-300

Simplified engine with small afterburner and short two-position nozzle for subsonic low-level operation. Fitted

to all MiG-27 versions, with fixed or variable inlet. Produced at Ufa as Product 55B.

R29PN

This replaced the R29B-300 as the standard engine of non-export MiG-23 aircraft.

R29BS-300

Different accessory gearbox. Produced from 1974 to 1986 as Product 55BS to power the Su-22 (export Su-17 versions).

LP Compressor

Five-stage, no inlet vanes or variable stators.

HP Compressor

Six-stage. Overall pressure ratio (29B) 12.4, (29-300) 13.1. Mass flow (29B) 105 kg (235 lb)/s, (29-300) 110 kg (242.5 lb)/s.

Combustion Chamber

Annular, vaporising burners.

HP Turbine

Single-stage with air-cooled blades; maximum 8,800 rpm. TET (R29-300) 1,150ºC, (R29BS-300) 1,135ºC.

LP Turbine

Single-stage, maximum 8,500 rpm.

Afterburner

Fuel rings with separate light-up give modulated fully variable augmentation. Fully variable nozzles differ in different installations (see variants).

Dimensions

Length:	
R29-300	4,960 mm (195.3 in)
R29B-300	4,992 mm (196.5 in)
Max diameter:	
R29-300	912 mm (35.9 in)
R29B-300	986 mm (38.8 in)

Weight, Dry

R29-300	1,880 kg (4,145 lb)
R29B-300	1,782 kg (3,929 lb)

Performance Ratings

(S/L)

Max afterburner:	

R29-300	117.65 kN (26,455 lb st)
R29B-300, BS-300	112.8 kN (25,353 lb st)
Max non-afterburner:	
R29-300	81.40 kN (18,298 lb st)
R29B-300, BS-300	78.45 kN (17,635 lb st)

Specific Fuel Consumption

Max T-O, afterburner:	
R29-300	56.45 mg/Ns (2.0 lb/h/lb st)
R29B-300	50.81 mg/Ns (1.8 lb/h/lb st)
Max non-afterburning:	
R29-300	26.81 mg/Ns (0.95 lb/h/lb st)
R29B-300	26.53 mg/Ns (0.94 lb/h/lb st)

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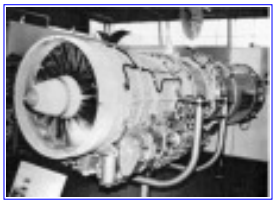
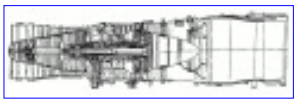
R29-300



R29B-300

R29B-300 with afterburner removed

Inverted assembly of R29B-300 at HAL



Longitudinal section through R29-300 (2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

R27V-300

While TMKB 'Soyuz' developed the R27 into the R29, the original design team at AMNTK 'Soyuz' developed this special version for the Yak-36M and Yak-38 V/STOL aircraft. Production engines were manufactured in 1974-91 at JSC (OAO) Tumenskie (which see) at Tumen.

R27AF-300

This conventional version was studied at AMNTK 'Soyuz' as the main engine of the Yak-28VV, a projected VTOL derivative of the Yak-28 tactical bomber which would also have been fitted with four R39P-300 vertical lift engines.

R28-300

Previously known as the R27VM-300 and R28V-300, this engine differed only in details from the R27V-300. Maximum T-O rating was reduced to 59.82 kN (13,448 lb st). It was fitted to the Yak-38M and to the Yak-38U trainer.

The description of the R27F2-300 applies, with the following differences:

Type

Vectored-thrust turbojet for V/STOL aircraft.

LP Compressor

Mass flow 100 kg (220.5 lb)/s.

HP Compressor

OPR 10.5.

HP Turbine

TGT 1,167ºC.

Jetpipe

Plenum chamber downstream of turbines divides gas flow into two and directs it through two jetpipes angled diagonally downwards and outwards. These terminate in rings to which are joined curved nozzles power-rotated in unison (in opposite directions) over a maximum arc of 100º to direct the efflux down, or to the rear, or slightly ahead.

Dimensions

Length	3,706 mm (145.9 in)
Diameter	1,012 mm (39.84 in)

Weight

Dry	1,350 kg (2,976 lb)

Performance Rating (S/L, static)

Max T-O:	67.66 kN (15,212 lb st)

Specific Fuel Consumption

Max T-O	24.92 mg/Ns (0.883 lb/h/lb st)

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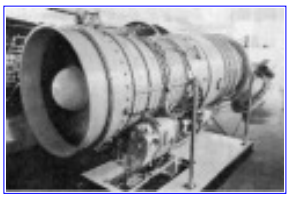

Two views of the R27V-300 (1996)

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TMKB (TUSHINSKOYE [TUSHINO] ENGINE DESIGN BUREAU) `SOYUZ'

R27-300

The R27 was designed at Factory No 300 (now AMNTK 'Soyuz') under Tumanskiy in 1966. It was transferred to TMKB 'Soyuz' where it was developed under Khachaturov, before being produced in large numbers at the Ufa factory. AMNTK 'Soyuz' retained the R27V programme, which is described under that heading.

R27F2-300

This entered production at Ufa in 1970, being called Product 47. It was fitted to the MiG-23MF and MS, a few of which remain in service (not in Russia).

R27F2M-300

This upgraded engine featured variable stators in the first two compressor stages and increased TGT. It was produced at Ufa as Product 47M for the MiG-23UB and MiG-23S. Except where indicated, the following refers to this version.

Type

Two-shaft turbojet.

LP Compressor

Five stages. No inlet guide vanes, but first two stators variable, circulation bleed around first-stage rotor. Max rpm 8,500. Mass flow (F2-300) 89 kg (196.2 lb)/s, (F2M-300) 95 kg (209.4 lb)/s.

HP Compressor

Six stages. No variable stators. Max rpm 8,800. OPR (F2-300) 10.0, (F2M-300) 10.9.

Combustion Chamber

Annular, with burners fed from inner manifold. In max afterburner, automatic water injection.

HP Turbine

Single stage with air-cooled rotor blades. Max TGT 1,100ºC.

LP Turbine

Single stage.

Jetpipe

Detachable afterburner with geometry similar to R25.

Dimensions

Length	4,850 mm (190.94 in)
Diameter	1,060 mm (41.73 in)

Weight

Dry	1,725 kg (3,803 lb)

Performance Ratings

(S/L, static, ISA)

Max afterburner:	
R27F2-300	95.0 kN (21,368 lb st)
R27F2M-300	100 kN (22,481 lb st)
Max dry:	
R27F2-300	63.725 kN (14,320 lb st)
R27F2M-300	67.6 kN (15,200 lb st)

Specific Fuel Consumption

Max afterburner	59.27 mg/Ns (2.1 lb/h/lb st)
Max dry	27.66 mg/Ns (0.98 lb/h/lb st)

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AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

R15-300

This turbojet was designed specifically for high-altitude cruising at over M2.5. Originally planned in 1956 for a cruise missile, it was found to be suitable for modification as the engine of a highly supersonic interceptor.

The original missile engine had no variable features and was virtually a turbo-ramjet. Like all such high-Mach engines it was notable for having a modest inbuilt pressure ratio (because of the far greater compression by ram effect in the inlet and duct), a convergent/divergent nozzle significantly larger in diameter than the inlet, and construction almost entirely of titanium and steel.

In 1965-75 various Mikoyan Ye-155 and Ye-155M prototypes set 30 world records for speed, climb and altitude. The aircraft were reported to the FAI as the Ye-266 (often translated in Western publications as E-266) and the engines as the R-266, with a thrust of 11,000 kg (24,250 lb st). All this was disinformation, though the records were real enough.

Production deliveries began in 1969. Gorkii and MMPP Salyut jointly delivered about 3,000 R15 engines. The following versions were produced.

R15-300

Also sometimes rendered as R15K or KR15. Engine of Tupolev 123 (service designation DBR and DR-2) *Jastreb* (Hawk) long-range cruise missile. Engine cruised in dry thrust, with maximum afterburner near target to give S/L rating of 137.3 kN (30,864 lb st).

R15B-300

Ratings (max dry) 73.5 kN (16,525 lb st), (a/b) 100.1 kN (22,500 lb st). Service life (TBO is implied) 150 hours only. Powered MiG-25P/PU/RB/RBK/RBT/RBV and RU and retained in modified MiG-25BM, RBF and RBSh.

R15BD-300

Ratings (max dry) 86.24 kN (19,387 lb st), (a/b) 110.0 kN (24,700 lb st). Despite increased ratings has shorter projecting multiflap nozzle. TBO extended to 1,000 hours. Powers MiG-25PD and PDS.

R15BF2-300

Zero stage, increased rpm and temperature. Propelling nozzle surrounded by fixed-geometry cylinder. Ratings (max dry) 107.2 kN (24,071 lb st), (a/b) 132.3 kN (29,740 lb st). Powered Ye-155M.

The following refers to the basic R15-300 except where otherwise stated:

Type

Single-shaft turbojet with afterburner.

Compressor

Five stages with fixed inlet and stator vanes. Construction titanium, steel and high-nickel alloy. Mass flow (B-300) 144 kg (317.5 lb)/s. Pressure ratio 4.75 (BF2, 4.95).

Combustion Chamber

Annular, with 18 vaporising burners. T-6 anti-coking fuel used exclusively, able to soak at high temperature in tank.

Turbine

Single stage, with solid rotor blades but special forced air cooling. TGT (B-300) 942ºC, (BD-300) 957ºC.

Afterburner

Large-volume jetpipe with coated and perforated corrugated liner, three spray rings with flameholder gutters fed with fuel and ignited in succession for `soft' light-up. Con-di nozzle with 12 fuel-actuated flaps. Overhaul life 50 hours, with each operation limited to 3 minutes.

Accessories

Radial shaft from front of compressor drives external gearbox with 10 drive pads. Gearbox is handed, on outer side under compressor casing.

Starting

Self-contained gas-turbine starter (usually S3 of 112 kW; 150 shp) in front of accessory gearbox. Dual high-energy ignition.

Control System

Hydromechanical, with separate subsystem for afterburner and nozzle.

Fuel Specifications

T-6, special fuel with freezing point -62ºC, flash point 54.4ºC.

Dimensions

Length:	
R15B-300	6,264 mm (246.6 in)
R15BD and BF versions	6,655 mm (262.0 in)
Diameter of inlet	996 mm (39.2 in)
Max diameter over nozzle:	
R15-300	1,512 mm (59.5 in)
R15BD and BF versions	1,640 mm (64.57 in)

Weight, Dry

R15B-300	2,625 kg (5,787 lb)
R15BD and BF versions	2,590 kg (5,710 lb)

Performance Ratings

See model listing

Specific Fuel Consumption

R15B-300:	
Max dry	35.42 mg/Ns (1.25 lb/h/lb st)
Max a/b	76.50 mg/Ns (2.70 lb/h/lb st)

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Two views of a cutaway R15B-300

Two views of a cutaway R15B-300

Simplified cutaway of R15BF2-300

R15B-300

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Date Posted: 12 April 2001

Jane's Aero-Engines 10

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

R11

This simple turbojet was designed by A A Mikulin and his deputies S K Tumanskiy and B S Stechkin. It was originally designated AM-11. It was one of the most important military aero-engines in history. Substantial numbers are still flying, some made in the Soviet Union and others in China.

The R11 was a pioneer two-spool engine, but with few compressor stages and the entire design biased in favour of toughness and simplicity rather than performance or economy. It pioneered the use of a first stage with transonic blades overhung ahead of the front bearing, without inlet guide vanes. It was also slim, to match its supersonic applications. There were many versions, the following being produced in series:

R11V-300

Initial production version, thought not qualified until 1958. The suffix -300 denotes the Ufa design and production bureau (see Soyuz introduction). Tailored to flight at ultra-high altitudes. Powered several special aircraft including Yak-25RV.

R11-300

First version to enter service, qualified 1954.

R11F-300

First mass-produced version, qualified 1956. Fitted with improved afterburner. Powers MiG-21F, P and U

subtypes. Service designation R-37F.

R11AF-300

Modified mountings and accessories to suit nacelle installation in Yak-28B, L and U.

R11F2-300

Fitted with improved compressor to increase mass flow, and new afterburner and nozzle. Powers MiG-21P, PF and FL.

R11AF2-300

Similar to F2-300 but with mountings and accessories tailored to Yak-28R, I and all P variants.

R11F2S-300

Similar to F2-300 but equipped to supply large flow of bleed air for flap blowing. Powers MiG-21PFM, PFS, S, U and UM, and Su-15, 15TM, UT and UM.

In total, about 20,900 of these versions were produced in the then Soviet Union, ending in 1972. A considerable further number have been produced in China, see under LM and LMC with designation WP-7.

Note: These fighter engines should not be confused with the simplified single-shaft version developed by the bureau at Ufa to power all versions of the Lavochkin La-17M target and cruise missile. These were designated R11K1, R11K2 and R11KA. Launch thrust was 5,400 lb (2,450 kg). Production R11K engines were delivered in 1960-79, see under GNPP 'Motor'.

Type

Two-shaft turbojet, (except V-300) with afterburner.

Intake

No separate structure, plain circular entry to compressors.

LP Compressor

Three-stage, with first stage overhung ahead of the front bearing with hemispherical rotating spinner and 31 snubberless titanium blades. Pressure ratio (typical) 2.74. Mass flow (300, V-300, F-300) 63.7 kg (140.4 lb)/s, (F2-300, AF2-300) 65 kg (143 lb)/s, (F2S-300) 66 kg (145.5 lb)/s.

HP Compressor

Three-stage, like LP with blades dovetailed into discs carried on short tubular shafts. Overall pressure ratio (300, V-300, F-300) 8.05, (F2-300, AF2-300) 8.72, (F2S-300) 8.9.

Combustion Chamber

Can-annular, with outer casing housing 10 flame tubes. Nos 1 and 6 of special shape, fitted with torch igniters. Air-film liners with ceramic coating on both sides. F2S engines have surrounding bleed-air manifold with a pipe connection on each side.

HP Turbine

Single stage, with 96 solid blades cast with integral tip shrouds held in fir-tree roots. TET (300, F2-300) 902ºC, (F-300) 897ºC, (F2S-300) 952ºC.

LP Turbine

Single stage, with solid inserted blades. Outlet temperature (typical) 810ºC.

Afterburner

Not fitted to R11V-300. Other versions, very long jetpipe with double-wall liner and three spray rings in Vee-gutter flameholders, fed from separate centrifugal fuel pump. Single full-power level only, available at max rpm. Multiflap variable nozzle actuated by (usually three) hydraulic rams.

Accessories

Most versions, gearbox above compressor casing with drives for fuel and oil pumps, starter/generator, aircraft alternator and hydraulic pump. On left side is a separate starting tank of petrol (gasoline) to ensure a ready light-up.

Control System

Hydromechanical, with single-lever throttle control and separate switch to engage afterburner when at 100 per cent rpm.

Dimensions

Length overall:	
R11V-300	2,953 mm (116.3 in)
R11F-300	4,660 mm (183.46 in)
Others (typical)	4,600 mm (181.1 in)
Diameter of inlet	825 mm (32.5 in)
Max diameter	906 mm (35.7 in)

Weight, Dry

R11V-300	895 kg (1,973 lb)
R11-300	1,040 kg (2,293 lb)
R11F-300	1,182 kg (2,606 lb)
R11F2-300, AF2-300	1,117 kg (2,462.5 lb)
R11F2S-300	1,126 kg (2,482 lb)

Performance Ratings

(S/L, ISA)

Max dry:	
R11V-300, F2S-300	38.24 kN (8,598 lb st)
R11F-300	38.04 kN (8,554 lb st)
R11F2-300, AF2-300	38.72 kN (8,708 lb st)
With afterburner:	
R11-300	43.62 kN (11,265 lb st)
R11F-300	49.00 kN (12,654 lb st)
R11F2-300, AF2-300	52.92 kN (13,668 lb st)
R11F2S-300	57.44 kN (14,832 lb st)

Specific Fuel Consumption

Max dry:	

R11V-300	26.34 mg/Ns (0.93 lb/h/lb st)
R11F-300, F2-300, AF2-300	26.62 mg/Ns (0.94 lb/h/lb st)
R11F2S-300	26.91 mg/Ns (0.95 lb/h/lb st)
With afterburner:	
R11-300	55.51 mg /Ns (1.96 lb/h/lb st)
R11F-300	61.74 mg/Ns (2.18 lb/h/lb st)
R11F2-300, AF2-300	62.03 mg/Ns (2.19 lb/h/lb st)
R11F2S-300	67.13 mg/Ns (2.37 lb/h/lb st)

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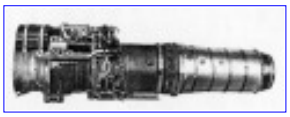
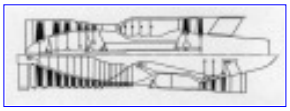
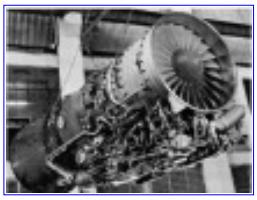


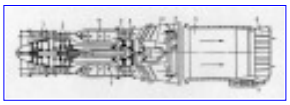
R11V-300



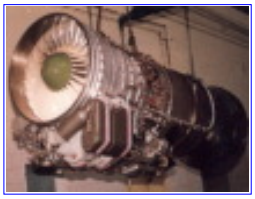
R11-300 (upper half) compared with the General Electric J79



R11F2S-300 with afterburner removed



A simplified section through an R11 afterburning turbojet. 1, LP compressor; 2, HP compressor; 3, HP turbine; 4, LP turbine; 5, afterburner; 6, variable nozzle; 7, main frame; 8, nozzle actuator; 9, afterburner gutters; 10, afterburner vaporiser; 11, combustion chamber flame tube



HAL-built R11F2S-300

Cutaway R11F (2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

RD-9

The first turbojet of wholly Soviet design to go into production was the AM-5. When this was certificated in 1953 the Mikulin OKB had already run the prototype of an improved engine, the AM-9, with a rebladed compressor with a zero stage with transonic air flow, increasing mass flow from 37.5 to 43.3 kg/s. From the outset the AM-9 was held to the same restricted diameter as its predecessor, because it was intended for supersonic aircraft. It passed the State bench test in 1954. The AM-9 was the key to the twin-engined MiG-19 and Yak-25, and it was produced in large numbers for both, the designation changing in 1956 to RD-9 to reflect Mikulin's replacement by Tumanskiy. Other versions followed:

RD-9A, RD-9AK

Versions without afterburner. Powered Yak-25M and Yak-26.

RD-9AF-300, RD-9AF2-300

Fitted with afterburner. Powers Yak-27 and Yak-28. Produced at Ufa 1957-74 and supported to 1986.

RD-9B

Different configuration. Fitted with afterburner. Powers MiG-19.

RD-9BF-811

Principal engine of MiG-19 and Chinese J-6 versions. Made in China with small modifications as WP6A (see LM, China).

RD-9BK, BKR

Powered La-17, -17MM and -17R pilotless aircraft 1959-85.

M-9A

Powered Tupolev 141 cruise missile.

M-9FK

Powered K-10 missile.

Type

Single-shaft turbojet, with or without afterburner.

Intake

Cast assembly with four de-iced radial struts, one housing drive to accessory section above and projecting ahead of inlet. Central fixed bullet and front bearing.

Compressor

Nine-stage. Welded ring construction. Mass flow (9A, 9B) 43.3 kg (95.46 lb)/s. (9BF-811) 46.2 kg (101.85 lb)/s. Pressure ratio (B) 7.44, (AF) 7.8.

Combustion Chamber

Can-annular type with 12 flame tubes, each terminating in a section of turbine inlet periphery. Spill-type burners. Two igniters fed from starting tank.

Turbine

Two-stage type with uncooled blades inserted into large flat discs, driving compressor via tubular shaft. TET (9B) 877ºC, (9BF-811) 1,027ºC.

Afterburner

Constant diameter type with main starting burner in turbine rear cone and single ring of fuel nozzles and gutter flameholders around rear of cone. Three stages of reheat. Ten adjustable nozzle flaps positioned by four rams.

Accessories

Vertical tower shaft off front of compressor drives gearbox above compressor casing. Two-speed starter/generator usually projects ahead of intake.

Control System

Autonomous hydromechanical starting and acceleration control.

Dimensions

Length, with afterburner	5,483-5,560 mm (215.9-218.9 in)
Diameter	668 mm (26.3 in)

Max height (RD-9BF-811)	950 mm (37.4 in)

Weight, Dry

RD-9B	695 kg (1,532 lb)
RD-9AF2	701 kg (1,545 lb)
RD-9BF-811	725 kg (1,598 lb)

Performance Ratings

(max rpm, S/L)

RD-9A, RD-9AK	27.46 kN (6,173 lb st)
RD-9B, dry	25.49 kN (5,732 lb st)
RD-9B, afterburner	31.87 kN (7,165 lb st)
RD-9B-811, dry	29.42 kN (6,614 lb st)
RD-9B-811, afterburner	36.78 kN (8,267 lb st)

Specific Fuel Consumption

RD-9BF-811, dry	27.76 mg/Ns (0.98 lb/h/lb st)
RD-9BF-811, afterburner	45.32 mg/Ns (1.6 lb/h/lb st)

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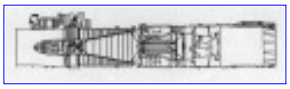

Section through RD-9AF-300

RD-9B with afterburner removed

Longitudinal section through RD-9BF-811 (shortened) (2001)

RD-9B (2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

AM-3, RD-3M

This turbojet, designed just after the Second World War, had little to commend it except size and simplicity. These assets were enough to find it several important applications over a period of well over 40 years.

After the War the engine design bureau of Gen A A Mikulin was the largest in the Soviet Union. Mikulin had the priceless advantage of S K Tumanskiy as his deputy, and Tumanskiy quickly became highly proficient in the technology of gas turbines. Under his direction the AM-2, -3, -5 and -9 were launched as major programmes. The AM-5 and AM-9 led to the RD-9 described next. The AM-3 led to the following:

AM-3

Also known as the M-209, the preseries engines were designed in 1947-49 by a team led by P F Zubets. First tested in 1950, and later air-tested in a retractable nacelle carried under a Tu-4LL. T-O rating 66.19 kN (14,880 lb st). Similar engines powered the Tu-104.

AM-3M

Service designation RD-3M. T-O rating 80.42 kN (18,078 lb st). Powered Tupolev 88 prototypes of the Tu-16.

RD-3M-200

First production version, assigned to the giant factory at Kazan (see KMPO). T-O rating 85.32 kN (19,180 lb st). Powered Tu-16 and derived aircraft converted from this original version, Tu-104A and Article 103 (M-4 prototype).

RD-3D

T-O rating 85.8 kN (19,290 lb st). Powered M-4. Also known as AM-3D.

RD-3M-500

T-O rating 93.2 kN (20,950 lb st). Powered nearly all later versions of Tu-16 and derivatives, and Tu-104B. About 4,100 delivered.

RD-3M-500A

T-O rating 103.0 kN (23,150 lb st), with 104.0 kN (23,386 lb st) available in emergency. Powered M-4A and 3MS.

WP8

RD-3M-500 made under licence in China, see XAE (China).

Type

Single-shaft turbojet.

Intake

Fabricated light alloy with front bearing held in six long-chord struts. Large bullet fairing over starter.

Compressor

Eight stages, with fixed IGVs and stators. Casing made in front, centre and rear sections. Mass flow (RD-3M-500) 150 kg (331 lb)/s at 4,700 rpm. Pressure ratio 6.4.

Combustion Chamber

Can-annular, with 14 flame tubes.

Turbine

Two stages, overhung behind rear bearing. TGT 857ºC. Gas temperature behind turbine at T-O rating 720ºC.

Jetpipe

Short, fixed area, exit diameter (RD-3M) 840 mm, (RD-3M-500A) 861 mm.

Starting

In most versions, S-300M gas-turbine starter (75 kW, 100 hp at 35,000 rpm) in nose bullet, driving through gearbox.

Dimensions

Length	5,340 mm (210.2 in)
Diameter	1,400 mm (55.1 in)

Weight

Dry	3,090-3,133 kg (6,812-6,907 lb)

Performance Ratings

See model listing.

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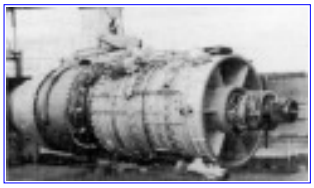
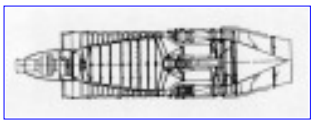
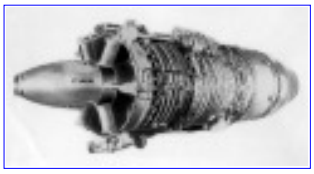


RD-3M-500



Longitudinal section through RD-3M



RD-3M-500A

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AERO-ENGINES - TURBOPROP/TURBOSHAFT, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

RYBINSK MOTORS JSC

TVD-1500 (RD-600)

The gas generator for the TVD-1500 was designed at Rybinsk in the late 1980s to provide a core suitable for turboprop, turboshaft and, ultimately, turbofan engines, as well as industrial and marine power units. Features include modular construction with the fewest possible number of parts, the use where justified of the very latest titanium alloys, new high-temperature materials and advanced composites, and FADEC control. By 1994, prototypes had run of turboshaft and turboprop versions, and the following models had been identified:

TVD-1500B

Baseline turboprop, with T-O rating of 970 kW (1,300 shp) and cruise rating (7,500 m; 24,600 ft at M0.65) of 559 kW (750 shp). Total of 22 produced 1990-99. On offer for Myasishchev M-102 and M-112, Beriev Be-32K and Sukhoi S-80. Antonov has abandoned a version of An-38 with this engine.

TVD-1500SKh

Tractor turboprop for Antonov An-102 (SKhS) agricultural aircraft. T-O rating 970 kW (1,300 shp).

TVD-1500T

Pusher propfan version. Selected for Ilyushin Il-Kh.

TVD-1500A

Also designated **RD-600S**. Turboshaft version. Selected as primary choice for twin-engined Aviaspetstrans Yamal, both engines driving common remote gearbox for AV-34 pusher propeller. T-O rating 970 kW (1,300 shp).

TVD-1500V

Also designated **RD-600V**. Turboshaft version, with contingency rating of 1,156 kW (1,550 shp) and T-O rating of 956 kW (1,282 shp). Total of 22 test engines produced 1989-99. Selected for twin-engined Ka-60 and Ka-62 helicopters. Expected eventually to power Ka-52 attack helicopter.

Type

Free-turbine turboshaft, turbofan or turboprop.

Intake

In the RD-600S, the air enters from a surrounding collector, the exhaust being at the output end next to the drive gearbox. In the tractor turboprop versions there is a ram inlet above the propeller with a duct leading to a remote power section.

Compressor

Three axial stages, with variable inlet guide vanes and first two stators, followed by one centrifugal stage. ECM-machined blades, EB-welded rotors and precision-cast casing. Mass flow 4.0 kg (8.8 lb)/s. Pressure ratio (1500B) 14.4, (RD-600V) 12.7.

Combustion Chamber

Annular folded reverse-flow type, with ring of vaporising burners round the rear face and two radially mounted high-energy igniters.

Compressor Turbine

Two stages, with solid monocrystal rotor blades. TGT 1,267ºC.

Power Turbine

Two stages, with DS rotor blades in first stage. In front-drive engines the discs are connected to a long quill shaft passing through the gas generator. In reversed engines, as in the section drawing, they are connected to a large-diameter tubular drive shaft pointing away from the gas generator.

Jetpipe

In front-drive tractor engines there is a simple curved pipe from the rear. In reversed engines the power turbines discharge into a surrounding scroll around the output shaft from which a curved jetpipe extends from either side or both sides.

Output

Turboprop versions have a two-stage spur gear followed by a single-stage planetary, but the location of the gearbox depends on the overall configuration. In the photographs it is carried remote from the gas generator (power section) and mounted on two sloping struts and the load-carrying tube surrounding the primary drive shaft. Output 1,700 rpm. Usual propeller AV-36, six blades, diameter 2,650 mm (104.3 in). In reversed engines the output is beyond the jetpipe. Most turboshaft versions have a single stage of spur gears.

Accessories

On front-drive engines most accessories are mounted on the rear face of the reduction gear at the front of the engine. In reversed engines a separate accessory gearbox is provided at the end opposite to the output, driven by the HP shaft.

Fuel Specifications

Engine described as having multifuel capability.

Dimensions

Length:	
Turboprop, front drive	1,965 mm (77.4 in)

Turboshaft	1,250 mm (49.2 in)
Width (typical)	620 mm (24.4 in)
Height (front-drive turboprop)	760 mm (29.9 in)

Weight, Dry

TVD-1500B	240 kg (529 lb)
RD-600V	220 kg (485 lb)

Performance Ratings

See model listing and table.

Specific Fuel Consumption

See table.

TVD-1500S ratings

	T-O		Max	Max cruise		Cruise	
Altitude	S/L	340 m	340 m	3,000 m	6,000 m	3,000 m	6,000 m
		1,115 ft	1,115 ft	9,843 ft	19,685 ft	9,843 ft	19,685 ft
Speed (km/h)	0	0	0	400	475	350	360
(mph)	0	0	0	249	295	217.5	224
Ambient (ºK)	303	288	303	269	249	269	249
Atmospheric pressure (mm Hg)	730	760	730	526	354	526	354
Power (kW)	969	1,044	838	746	646	559	514.5
(shp)	1,300	1,400	1,124	1,000	866	750	690
SFC (µg/J)	76.72	-	80.10	71.82	66.75	80.10	69.96
(lb/h/shp)	0.454	-	0.481	0.432	0.399	0.481	0.419

RD-600V ratings (first four columns S/L static, ISA 288ºK)

	Contingency	Max continuous	Max	Cruise	Cruise (5,000 m,
					16,404 ft)
Power (kW)	1,156	969	828	746	559
(shp)	1,550	1,300	1,110	1,000	675
SFC (µg/J)	-	77.73	81.45	83.73	83.73
(lb/h/shp)	-	0.461	0.483	0.4955	0.4955

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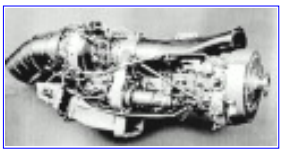


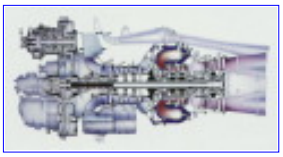
(2001)

Two views of TVD-1500S front-drive turboprop

TVD-1500 turboprop with Sich six-blade propeller

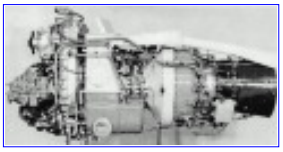
Longitudinal section through TVD-1500A reversed-configuration pusher turboshaft

Longitudinal section through TVD-1500B



Longitudinal section through RD-600V (1998)

RD-600V (Yefim Gordon) (1998)



RD-600V (2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

RU-19

Service designation: TRD-29
This simple turbojet was developed in 1958-59 to power the Soviet light trainer competition, the entrant being the Yak-30. The winner was the Czech L-29.

RU-19-300

Flight-cleared in 1961 for Yak-30 and Yak-32. T-O rating 8.83 kN (1,985 lb st). From 1966, installed in right nacelle of An-24RV and An-24RT.

RU-19A-300

Combined propulsion engine and APU. Used during hot/high T-O to relieve main engines of electrical load and provide small residual thrust; available in flight if necessary. T-O rating (no APU load) 7.85 kN (1,765 lb st), (maximum electrical load) 2.16 kN (485 lb st). Installed in right nacelle of An-26 and An-30. Produced from 1969 by OAO Tumenskie (which see). Also made under licence by Aerostar, Romania.

Details below refer to the RU-19-300:

Type

Single-shaft turbojet.

Compressor

Seven-stage axial, no variable features. Mass flow 16 kg (35.3 lb)/s. Pressure ratio 4.6.

Combustion Chamber

Annular.

Turbine

Single stage, TGT 877ºC.

Jetpipe

Variable-area nozzle.

Dimensions

Length	1,730 mm (68.1 in)
Diameter	550 mm (21.65 in)

Weight

Dry	225 kg (496 lb)

Performance Ratings

See model listing

Specific Fuel Consumption

T-O, S/L (fully rated)	30.54 mg/Ns (1.079 lb/h/lb st)

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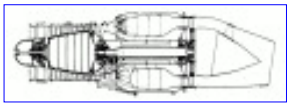
RU-19A-300 (Aerostar-built)



Longitudinal section through RU-19-300 (2001)

RU-19-300 installed in An-24RV

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

RYBINSK MOTORS JSC

RD-38

In one form also designated RD-60, this simple turbojet family was derived from the last survivor of the big RKBM effort on specialised lift jets for V/STOL fighters. Aerodynamically it is closely related to the RD-36-35BFR (Yak-36M/38) and RD-36-35PR (VVA-14). Major differences include installation in the horizontal attitude, long-life certification, reduced TGT, recirculatory oil system and a modified control system. It was developed under Novikov in 1983 and Kolesov in 1985. About 400 of the following versions have been delivered.

RD-38

T-O rating 32.5 kN (7,165 lb st), dry weight 231 kg (509 lb). Automatic starting and constant-speed full-throttle operation. Total of 190 delivered 1979-89 for vertical installation in Yak-38M. Also used horizontally in unmanned vehicles.

RD-38K, RD-60

T-O rating 27.50 kN (6,065 lb st), also given by Beriev as 24.21 kN (5,511 lb st). T-O booster for Beriev A-40 Albatross, mounted horizontally, shut down in cruise with inlet/nozzle faired off. Total of 24 delivered 1984-87. Described below.

RD-38A

T-O rating 27.50 kN (6,065 lb st). Produced from 1985 for An-71. Mounted horizontally in rear fuselage.

Long-life lubrication system.

Type

Single-shaft turbojet.

Compressor

Six-stage axial. Mass flow (38) 45.2 kg (99.65 lb)/s, (38K) 44.5 kg (98.1 lb)/s. Pressure ratio 5.0. OPR 5.2.

Combustion Chamber

Annular, with 30 fuel nozzles.

Turbine

Single stage with aircooled blades, originally with air-impingement starting. TGT (all versions) 1,097ºC.

Jetpipe

Fixed area nozzle.

Starting

Electric

Dimensions

Length	1,702 mm (67.0 in)
Inlet diameter	630 mm (24.8 in)

Weight, Dry

RD-38, RD-38K	223 kg (491.6 lb)

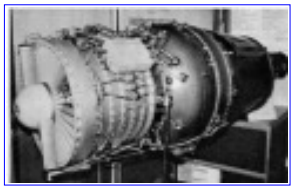
Performance Ratings

See model listing

Specific Fuel Consumption

RD-38K, RD-38A	39.63 mg/Ns (1.4 lb/h/lb st)
RD-38K	42.46 mg/Ns (1.5 lb/h/lb st)

UPDATED



RD-38K (RD-60)

RD-38
(2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

RYBINSK MOTORS JSC

RD-36-51

Despite the similarity of designation, this engine has no connection with the RD-36-35, which was a simple lift jet. Both were designed by P A Kolesov.

RD-36-51A

Designed for Tu-144D, cruising at M2.2. Production engines totalled 91, delivered 1968-78. In 1995, none of the Tu-144D aircraft was airworthy, and 08-2, in 1998 restored to flight status, has been re-engined with the NK-321. The RD-36-51 is included here for completeness. Set 13 load/speed/altitude records.

RD-36-51V

Simplified version with fixed nozzle, run at much lower rpm. Total of 15 produced 1973-78. Powers Myasishchev M-17 Stratosfera. Set 25 load/altitude records.

Type

Single-shaft turbojet.

Compressor

Fourteen-stage axial with variable inlet vanes and first five and last five stator stages. Mass flow (51A) 275 kg (606.3 lb)/s, (51A at 18 km, M2.2) 201 kg (443 lb)/s, (51V) 278.9 kg (615 lb)/s. Pressure ratio (51A) 15.8,

(51V) 7.6.

Combustion Chamber

Can-annular with 16 burners.

Turbine

Three-stage axial with cooled blades. Entry temperature (51A) 1,067ºC, (51V) 1,098ºC.

Nozzle

(51A) afterburner with multiflap. Laval type with adjustable spike, (51V) subsonic, fixed area.

Accessories

Airframe-mounted, driven via tower shaft and remote gearbox,at front on 51A, at rear on 51V.

Dimensions

Length	5,228 mm (205.8 in)
Diameter	1,415 mm (55.7 in)

Weight, Dry

RD-36-51A	4,125 kg (9,094 lb)
RD-36-51V	3,860 kg (8,510 lb)

Performance Ratings

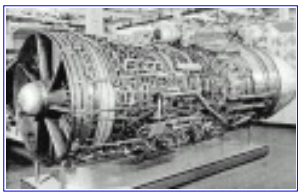
RD-36-51A:	
T-O	196.12 kN (44,090 lb st)
Cruise (11 km; 36,089 ft, 1,000 km/h)	29.4 kN (6,614 lb)
Cruise (18 km; 59,055 ft, 2,350 km/h)	49.03 kN (11,023 lb)
RD-36-51V:	
T-O	68.6 kN (15,430 lb st)
Cruise (25 km; 82,020 ft, M0.7)	5.88 kN (1,323 lb)

Specific Fuel Consumption

RD-36-51A, RD-36-51V, max dry	24.94 mg/Ns (0.883 lb/h/lb st)
RD-36-51A, subsonic cruise, as above	26.55 mg/Ns (0.94 lb/h/lb)
RD-36-51A, M 2.2 cruise, as above	34.8 mg/Ns (1.23 lb/h/lb)
RD-36-51V cruise, as above	34.23 mg/Ns (1.21 lb/h/lb)

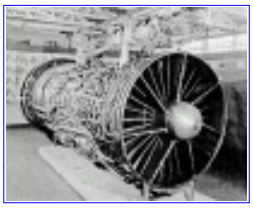
UPDATED

Two views of RD-36-51A



RD-36-51V
(2001)

Tu-144D showing the Laval-type spike nozzles of the RD-36-51A engines



RD-36-51A2
(2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

RYBINSK MOTORS JSC

RD-7, VD-7

Vladimir Alekseyevich Dobrynin was a distinguished designer of high-power piston engines. His M-250 was developed into the VD-4, rated at 3,207 kW (4,300 hp) at sea level and 2,088 kW (2,800 hp) at 11,000 m (36,090 ft), and was the most powerful piston engine ever to fly. Unlike many of his contemporaries, he easily moved into the era of gas turbines, and he designed the VD-7 to power the Myasishchev Type 103 (M-4, later 3M) strategic bomber. It was made in several versions, a total of 780 being delivered in 1957-65. Small numbers of most versions are still active. One version powered the largest Ekranoplans (eight for propulsion and lift, two for propulsion only).

VD-7

Initial preseries version, first run in late 1952 and qualified in 1956. T-O rating initially 107.9 kN (24,250 lb st).

VD-7B

Modified and derated to increase reliability. Selected to power six variants of 3M bombers and tankers. First delivery March 1957. Set 14 altitude/speed/payload records.

VD-7P

Rebladed compressor and other changes to increase power at high altitudes. T-O rating 110.8 kN (24,910 lb). Powered 3MYe.

RD-7M

Originally called VD-7M. Developed by P A Kolesov for supersonic flight. Higher rotational speed (7,400 rpm). Ratings: maximum dry 103.0 kN (23,149 lb st), maximum afterburner 156.9 kN (35,275 lb st). Initial engine for Tu-105 and Tu-22, also powers Myasishchev M-51. In 1960-65 a total of 510 were delivered.

RD-7M2

Fitted with improved afterburner and nozzle. Total of 1,865 produced 1965-77 for operational versions of Tu-22. In 2000 a few were still active in Ukraine.

RD-7MD

Engines removed from demilitarised Tu-22s and modified without afterburners. T-O rating 105.42 kN (23,700 lb st). Powers VM-T Atlant.

Type

Single-shaft turbojet, with or without afterburner.

Intake

Cast aluminium, with front bearing carried by six anti-iced struts.

Compressor

Nine-stage axial, with tapering outer diameter. No variable stators. All production engines have large bleed manifolds. Mass flow (VD-7B) 173 kg (381.4 lb)/s, (RD-7M) 177 kg (390.2 lb)/s, (RD-7M2) 181 kg (399.0 lb)/s. Pressure ratio (VD-7B, RD-7M) 11.2, (RD-7M2) 10.8.

Combustion Chamber

Can-annular, with multiple downstream vaporising burners.

Turbine

Two stages, with inserted solid blades. TET (VD-7B) 817ºC, (RD-7M) 862ºC, (RD-7M2) 860ºC.

Jetpipe

(VD-7B) simple fixed-area, (RD-7M2) large afterburner with three nozzle rings and multiflap variable nozzle.

Accessories

Gas turbine starter, alternator and other high-power accessories mounted on gearbox above compressor, driven by tower shaft in 12 o'clock inlet strut.

Dimensions

Length:	
VD-7B	4,247 mm (167.2 in)
RD-7M2	7,204 mm (283.6 in)
Inlet diameter:	
VD-7B	1,288 mm (50.7 in)
RD-7M2	1,216 mm (47.87 in)

Weight, Dry

VD-7B	2,765 kg (6,096 lb)
RD-7M	3,650 kg (8,047 lb)
RD-7M2	3,825 kg (8,433 lb)

Performance Ratings (T-O, S/L)

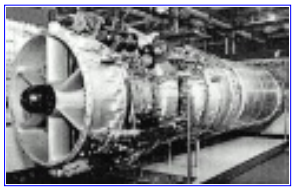
VD-7B	93.2 kN (20,950 lb st)
RD-7M2 max dry	107.9 kN (24,250 lb st)
RD-7M2 max afterburner	161.8 kN (36,376 lb st)

Specific Fuel Consumption (T-O, S/L)

VD-7B	22.66 mg/Ns (0.8 lb/h/lb st)
RD-7M	23.37 mg/Ns (0.825 lb/h/lb st)
RD-7M2 max dry	24.64 mg/Ns (0.87 lb/h/lb st)
RD-7M2 max afterburner	58.06 mg/Ns (2.05 lb/h/lb st)

UPDATED

Part cutaway VD-7B



RD-7M2 (2001)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

LYUL'KA SATURN INC

AL-31

This was Arkhip Lyul'ka's last and greatest engine. Intended for large supersonic fighters, design started in 1963, and the first AL-31 began testing in August 1974. AL-31 gas-generators are used in ship propulsion and gas pumping, with Rolls-Royce collaboration.

AL-31F

First production engine, fitted with afterburner and matched to the Sukhoi Su-27. Service designation **R-32**, reported to the FAI as the power plant of the P-42 (modified Su-27 prototype) which set 32 time-to-height records in 1986. The AL-31F entered production in late 1981, and received final qualification in 1985. By 1993, about 1,500 had been delivered from MMPP Salyut and UMPO Ufa. TBO 900 hours, with hot-section inspection at each 300 hours. The engine is entirely modular, with the ability to replace the nozzle, afterburner, mixer, LP turbine, LP compressor and gearbox without removing the remainder from the aircraft. With the engine installed, it is also possible to replace the 1st LP compressor blades or all rotor stages of the HP spool. A high proportion of the construction is steel or titanium. A particular design feature was to achieve LP and HP compressors which would not surge no matter what might be happening in the sharp-edged aircraft inlet at extreme or even negative AOA, and to eliminate acoustic connection between the afterburner and the LP spool.

AL-31A

In December 1999, the Russian arms-exporting organisation Rosvoorouzheniye said it was `establishing a facility in China for the production of spares and repair of AL-31A engines which power Chinese fighters'. The aircraft involved is doubtless the Chengdu J-10 (F-10). The first prototype of this single-engined aircraft underwent taxi trials in late 1997, but the first flight was reportedly `delayed by a major engine malfunction' until 24 March 1998. At that time 10 engines had been imported for this programme.

AL-31FP

Developed from 1988, with features noted in description below. Fitted with definitive thrust-vector control with AL-100 nozzle. The first AL-31 TVC nozzle was tested in 1986. This was then fitted to one engine of Su-27 07-02, flown on 21 March 1989. On 12 April 1996 aircraft 711 began testing with two TVC engines linked to the aircraft's FBW flight-control system. All these early nozzles had ±15º movement in the vertical plane, driven by the aircraft's hydraulic system. The AL-31FP has the refined AL-100 nozzle with an axis inclined at 32º (handed left/right in the aircraft) to give direct control in both transverse planes and facilitate single-engined flight. The nozzle is driven by two pairs of rams in a kerosene-operating system independent of the aircraft hydraulics. Nozzle TBO is 250 hours, that of the remainder of the engine being 1,000 hours. The AL-100 nozzle has now been integrated with the more powerful AL-37, described separately.

AL-31FN

Developed 1992-94. Increased thrust (maximum 125.5 kN, 28,218 lb st), FADEC control with hydraulic backup, and improved fuel economy. Fitted to Su-27IB (Su-34), Su-27M (Su-35) and Su-32FN, all of which have a range of 4,000 km (2,485 miles) on internal fuel.

AL-31F/VCN

Refined VCN (vector-controlled nozzle) version, cleared for flat, straight and inverted spins and to airspeed of minus 200 km/h (124 mph). Available in standard or tropical versions.

AL-31ST

Industrial (ST = shaft turbine) engine, see company introduction and UMPO entry.

Except where otherwise stated, the following description applies to the AL-31F:

Type

Two-shaft augmented turbofan.

LP Compressor

Four stages slotted into discs. First stage preceded by 23 guide vanes with 30º movement. Mass flow 110 kg (243 lb)/s.

HP Compressor

Variable inlet guide vanes followed by nine-stage spool with first three stators variable. Easy field

replacement of damaged blades. Overall pressure ratio 23. Bypass ratio, AL-31 0.6, AL-31F 0.571.

Combustion Chamber

Annular, with 28 downstream burners fed from inner manifold. Auto continuous ignition during missile launch. AL-31FP, machined chamber with welded fuel burners.

HP Turbine

Single-stage with cooled blades, using air/air heat exchanger in bypass duct. Entry gas temperature up to 1,427ºC. AL-31FP has new coated blades and upgraded rear bearings.

LP Turbine

Two-stage with cooled blades. Both turbines have active tip clearance control. AL-31FP, strengthened disk.

Jetpipe

Short mixer section to combine core and bypass flows upstream of afterburner.

Afterburner

Two flameholder rings downstream of multiple radial spray bars. Interlinked primary and secondary multiflap nozzles are angled about 5º downwards.

Accessories

Grouped above engine, with main banana gearbox ahead of inlet.

Control System

Hydromechanical full regime control giving smooth power from flight idle to maximum afterburner in all manoeuvre conditions. Auto elimination of surge `at Mach numbers 2 to 2.5 when normal, flat and inverted spins occur'. Linked via software to Su-27 fly-by-wire flight control system.

Dimensions

Length	
AL-31F	4,950 mm (195 in)
AL-31FP, F/VCN	4,990 mm (196.5 in)
AL-31FN	5,000 mm (196.9 in)
Max diameter	1,277 mm (50.27 in)
AL-31F	1,240 mm (48.82 in)
AL-31FP, F/VCN	1,277 mm (50.28 in)
AL-31FN	1,180 mm (46.46 in)
Inlet diameter	910 mm (35.8 in)

Weight, Dry

AL-31F	1,530 kg (3,373 lb)
AL-31FP, F/VCN	1,570 kg (3,461 lb)

AL-31FN	1,538 kg (3,391 lb)

Performance Ratings

Max augmented	122.6 kN (27,560 lb st)
Max dry	79.43 kN (17,857 lb st)

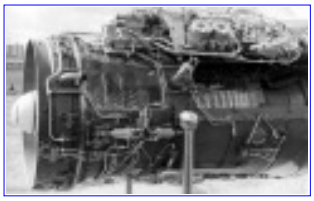
Specific Fuel Consumption

Max augmented (typical)	55.52 mg/Ns (1.96 lb/h/lb st)
Max dry:	
AL-31F	18.87 mg/Ns (0.666 lb/h/lb st)
AL-31FP	18.98 mg/Ns (0.67 lb/h/lb st)
AL-31FN	19.97 mg/Ns (0.705 lb/h/lb st)

UPDATED

AL-31F showing aircraft accessory gearbox (Nigel Eastaway) (1998)

AL-31F, without accessory gearbox



Cutaway AL-31F

Longitudinal section through AL-31F (fixed nozzle)

AL-31FP showing vectoring nozzle (Yefim Gordon) (1998)

Cutaway AL-31FP (Yefim Gordon) (1998)

The completely new AL-100 nozzle, ±15º pitch, ±12º yaw (Yefim Gordon) (1998)

AL-31F/VCN (2001)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

R127-300

This engine is intended for smaller business jets.

Type

Two-shaft turbofan.

Fan

Single stage. Mass flow 31 kg (68.3 lb)/s. Bypass ratio 4.8.

Compressor

Two centrifugal stages.

Combustion Chamber

Annular folded reverse-flow.

Turbines

Two-stage HP, two-stage LP.

Jetpipe

Mixer leading to combined nozzle. Provision for reverser.

Performance Ratings

(ISA, S/L)

T-O	8.83 kN (1,984 lb st)

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Date Posted: 12 April 2001

Jane's Aero-Engines 10

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

R126-300

This engine is intended for large business jets and regional transports. Tupolev has selected it for the projected Tu-324.

Type

Two-shaft turbofan.

Fan

Single stage.

Compressor

Five axial stages with variable stators, one centrifugal.

Combustion Chamber

Annular folded reverse-flow.

Turbines

Two-stage HP, three-stage LP.

Jetpipe

Mixer leading to combined nozzle. Provision for reverser.

Performance Ratings

(ISA, S/L)

T-O	39.2-53.9 kN (8,818-12,125 lb st)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

R123-300

Projected engine for low-cost propulsion of light high-subsonic aircraft.

Type

Two-shaft turbofan.

Fan

Single-stage. Bypass ratio about 6.

Compressor

Two axial stages, with variable inlet vanes, and single centrifugal.

Combustion Chamber

Annular folded reverse-flow.

Turbines

Single-stage HP, single-stage LP.

Jetpipe

Mixer leads to combined nozzle.

Performance Ratings

4.22 kN (948 lb st) class.

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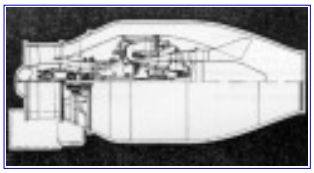

Longitudinal section through R123-300 (provisional)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

TMKB (TUSHINSKOYE [TUSHINO] ENGINE DESIGN BUREAU) `SOYUZ'

RD-1700

When what is today the VPK MAPO group designed the MiG-AT trainer it took the strategic decision to use foreign engines and avionics. The choice for both fell upon France (see Larzac 04-R20 under Turbomeca-SNECMA). Many Russians have emphasised to the Editor that this decision, taken to save time and improve export prospects, is now regarded as a mistake. The French engine is basically a 30-year-old design which has had to be put back into production to meet the initial Russian order for 10 engines. The Chief of Acquisition of the Russian Air Force, Yuri Klishin, stated in 1997 that the MiG-AT would be procured only if every part can be made in Russia. The aircraft's chief designer, Anatoliy Popov, jokingly said ``The amount of money required to purchase the rights to licence-produce the Larzac is enough to feed all of Russia for three years."

To provide a modern Russian alternative engine for the all-Russian MiG-UTS version, the Soyuz bureau, in collaboration with the Central Institute of Aviation Engines, has designed the RD-1700, based on a previously schemed gas generator (unrelated to the R123-300). The RD-1700 has a specification close to that of the Larzac, though in the initial version the bypass ratio is lower. Total service life is given as 6,000 hours, with 4,000 hours for the hot section. Later it is intended to develop advanced versions with a fan of greater diameter, as noted below. Production would be handled by MMP Chernyshov (which see).

Type

Two-shaft turbofan.

Fan

Two stages. Bypass ratio 0.78 (later versions, up to 3.3). Mass flow 30.0 kg (66.1 lb)/s.

Compressor

Four stages. Overall pressure ratio 14.3 (developed versions, 19.6).

Combustion Chamber

Annular.

HP Turbine

Single stage, with cooled blades. TET 1,420ºC.

LP Turbine

Single stage.

Jetpipe

Mixer and fixed-area nozzle; augmented version has variable nozzle.

Weight,:

Dry:	297.5 kg (656 lb)
Developed version	345.5 kg (761.7 lb)

Performance Rating

(ISA, S/L)

T-O	16.67 kN (3,748 lb st)
T-O (augmented)	19.61 kN (4,409 lb st)
T-O (developed version)	25.6 kN (6,658 lb st)

Specific Fuel Consumption

T-O, as above	19.165 mg/Ns (0.7 lb/h/lb st)
T-O (augmented)	35.59 mg/Ns (1.3 lb/h/lb st)
T-O (developed)	14.166 mg/Ns (0.5 lb/h/lb st)

UPDATED

Longitudinal section through RD-1700, lower half showing augmented version (2000)

RD-1700 display mockup (2001)

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

OMSK ENGINE DESIGN BUREAU

TVD-20

This turboprop was derived from the core of the TVD-10, with a zero stage, and a second stage on the power turbine. It has a complete reverse-flow layout, the inlet facing aft at the extreme rear. Development was carried out partly at Omsk Baranov.

TVD-20-01

Initial form, exhibited 1992 as engine of Antonov An-3. Also announced as engine of NIAT 2.5ST, believed discontinued project.

TVD-20

Initial production version, fitted to An-3 driving 2,650 mm (104.3 in) AV-36 or AV-106 six-blade propeller, max 1,700 rpm. Description below refers to this version, except where otherwise stated.

TVD-20M

Redesigned with two-stage centrifugal compressor and modified reduction gear, fitted to An-3 driving 3,600 mm (141.7 in) AV-17 three-blade propeller, max 1,581 rpm. Intended engine of ROKS-Aero T-101V. Was intended for Myasishchev M-102 and competing for M-202.

TVD-20-03

Derived from TVD-20 incorporating elements of TVD-10B and VSU-10 APU. Fitted to non-export versions of Antonov An-38, driving AV-36 or AV-106 propeller, rpm 1,100-1,700.

TVD-20V

Turboshaft version based on TVD-20M, with the same two-stage centrifugal compressor. Automatic starting cycle. Output 3,115 rpm.

Type

Free-turbine turboprop.

Intake

Light-alloy ring, facing aft and incorporating oil tank. No inlet guide vanes.

Compressor

Seven axial stages followed by one centrifugal, all rotating on same shaft. Mass flow 5.4 kg (11.9 lb)/s. Pressure ratio 9.0, (TVD-20M, TVD-20V) two-stage centrifugal.

Combustion Chamber

Drum containing short annular flame tube. Fuel flung off spinning disc on main shaft. Auxiliary starting burners at top and bottom, each with semiconductor igniter.

Turbines

Two-stage gas generator turbine. Blades held by fir-tree roots in discs pegged together. Two-stage power turbine with a shaft bearing on each side.

Output

Quill shaft splined to power-turbine shaft drives input pinion to two-stage planetary reduction gear giving propeller speed 1,700 rpm. Brake provided to stop propeller during fast loading of agricultural chemicals. Integral torquemeter.

Accessories

Main gearbox at front of engine on aft face of propeller gearbox, one drive going to 16 kW generator. A second gearbox at the rear, driven through tower shaft in 12 o'clock inlet strut, serves fuel and oil pumps, one or two 27 V starter motors and tachogenerator. Provision for drives to dusting/spraying gear.

Dimensions

Length:	
TVD-20, TVD-20-03	1,900 mm (74.8 in)
TVD-20M	1,770 mm (69.7 in)
TVD-20V	1,850 mm (72.83 in)
Width:	
TVD-20	850 mm (33.46 in)
TVD-20M, TVD-20-03	845 mm (33.27 in)
TVD-20V	855 mm (33.66 in)
Height:	

TVD-20	800 mm (31.50 in)
TVD-20M, TVD-20-03	850 mm (33.46 in)
TVD-20V	745 mm (29.33 in)

Weight, Dry

TVD-20	240 kg (529 lb)
TVD-20M	285 kg (628.3 lb)
TVD-20-03	250 kg (551 lb)
TVD-20V	210 kg (463.0 lb)

Performance Ratings

(S/L, ISA)

T-O:	
TVD-20	1,081 kW (1,450 shp)
TVD-20M, TVD-20-03	1,066 kW (1,430 shp)
TVD-20V	1,119 kW (1,500 shp)
Max continuous:	
TVD-20	1,044 kW (1,400 shp)
TVD-20M, TVD-20-03, TVD-20V	1,025 kW (1,375 shp)

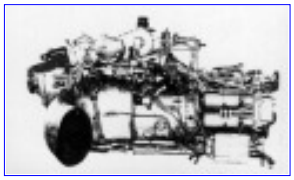
Specific Fuel Consumption

(T-O, as above)

TVD-20	81.86 Ág/J (0.485 lb/h/shp)
TVD-20M	91.14 Ág/J (0.540 lb/h/shp)
TVD-20-03	83.71 Ág/J (0.496 lb/h/shp)
TVD-20V	85.57 Ág/J (0.507 lb/h/shp)

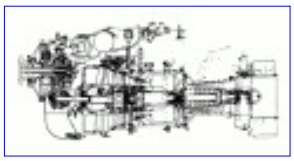
UPDATED

Cutaway section through TVD-20M (inlet at right)

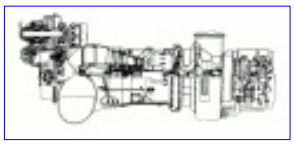


TVD-20M (inlet at right)

TVD-20-01



Longitudinal section through TVD-20-03 (2001)



Part-section through TVD-20V (2001)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-22

From 1957 the N D Kuznetsov design team led the world in the development of high-power augmented turbojets and turbofans for the propulsion of large supersonic aircraft. The biggest single advance was the NK-6, tested from May 1958, but this failed to go into production. This led to the NK-144, developed for the Tu-144 supersonic transport. A section drawing of the initial version of NK-144 is reproduced here. Unlike later versions, and the NK-22, this engine had only two fan stages.

NK-22

This engine was developed from the NK-144 to provide the power plant of the Tu-22M swing-wing supersonic bomber and missile platform. Most design parameters were similar, though the military application was aimed at a lower maximum high-altitude Mach number (1.88 instead of 2.35). The NK-22 differed from the NK-144 family mainly in being designed to military standards, and in having a completely different arrangement of accessories. The final order to build was received at Kuibyshyev on 28 November 1967. At this time design, again led by Ye M Semenov, had long since been completed, and much rig testing had been done, especially on the different afterburner system.

The NK-22 went on test in April 1968, and passed its State certification in October 1970. The first Tu-22M-0 flew in August 1969. One requirement was to run at full power for 15 hours with both (dual) fuel supply systems functioning. In July 1976 the improved NK-23 was tested, but this was discontinued in 1977, and the NK-22 remained the standard engine of the Tu-22M-1 and M-2, about 500 engines being delivered from the Samara factory. It was succeeded by the NK-25.

Type

Two-shaft augmented bypass turbojet (low-BPR turbofan).

Intake

Fixed geometry, fabricated in steel with 15 radial struts with interspersed peripheral guide vanes. Fully anti-iced by hot air.

Fan

Three stages, first rotor stage having straight (not kinked, swept forward, then back) blades with mid-span snubbers. Mass flow 303 kg (668 lb)/s. Bypass ratio 0.6.

IP Compressor

Three stages, rotating on the LP (fan) shaft.

HP Compressor

Six stages, with variable inlet guide vanes. OPR 14.75.

Combustion Chamber

Annular, derived from NK-8.

HP Turbine

Single stage, with air-cooled blades using vortex and convective film cooling. Stators covered with thin ceramic tiles. TGT 1,117ºC.

LP Turbine

Two stages.

Afterburner

Two rings of fuel burners in core flow with burning taking place inside open drum of refractory alloy, around which are a high-augmentation ring of burners injecting into bypass flow. Fully variable augmentation. Convergent/divergent 20-flap nozzle.

Control System

Separate duplicated hydromechanical for engine and afterburner.

Accessories

Grouped on underside of compressor case, driven by 6 o'clock tower shaft from front of LP shaft. Most items duplicated.

Dimensions

Length	5,200 mm (205 in)
Fan diameter	1,348 mm (53.07 in)
Envelope diameter	about 1,500 mm (59 in)

Weight

Dry	3,290 kg (7,253 lb)

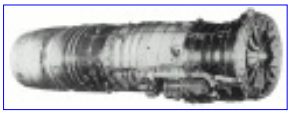
Performance Ratings (S/L, static)

Max T-O, with afterburner	196.15 kN (44,090 lb st)
Military (max dry)	158.9 kN (35,715 lb st)

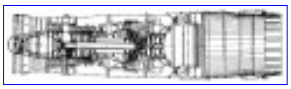
Specific Fuel Consumption

Max T-O, as above	55.26 mg/Ns (1.95 lb/h/lb st)

UPDATED



NK-144A
(2001)



A section drawing of the initial version of NK-144
(2001)

Longitudinal section through NK-22
(2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

TMKB (TUSHINSKOYE [TUSHINO] ENGINE DESIGN BUREAU) `SOYUZ'

R35-300

This turbojet was developed by a team led by Khachaturov on the basis of their preceding R29. The two engines are generally similar, but the R35 has an improved compressor and is cleared to higher temperatures.

R35-300

The principal production version, fitted to the MiG-23ML, MLA, MLD and P. As far as possible the engine was designed to withstand combat damage and near misses by large SAM warheads. Produced by AO MMP named for V V Chernyshov (which see) as Product 77.

Type

Two-shaft turbojet with afterburner.

LP Compressor

Five-stage. Mass flow 110 kg (242.5 lb)/s.

HP Compressor

Six stage. Pressure ratio 13.0.

Combustion Chamber

Annular, with downstream vaporising burners. Configured for minimal visible smoke.

HP Turbine

Single-stage with cooled blades. TET 1,250ºC.

LP Turbine

Single-stage.

Afterburner

Fully modulated, with soft light-up and single-lever control to maximum. Convergent/divergent hydraulically controlled variable nozzle.

Dimensions

Length	4,975 mm (195.9 in)
Diameter	912 mm (35.9 in)

Weight

Dry	1,794 kg (3,955 lb)

Performance Ratings

(S/L, ISA)

Max T-O	127.46 kN (28,660 lb st)
Max dry	83.85 kN (18,850 lb st)

Specific Fuel Consumption

Max T-O	55.32 mg/Ns (1.96 lb/h/lb st)
Max dry	27.09 mg/Ns (0.96 lb/h/lb st)

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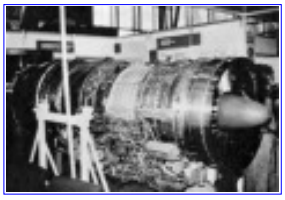

R35-300

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

MOTOR, GNPP (STATE SCIENTIFIC PRODUCTION ENTERPRISE)

R25

Whereas the R13 was based on the R11, Gavrilov's R25 was a fresh design, though installationally interchangeable with the R13 to enable it to be retrofitted into existing MiG-21 fighters. The new compressor handled an increased air flow with a higher pressure ratio, and was again improved to increase the surge margin, the combustion system was again refined, and a completely new afterburner fitted giving stepless augmentation. The proportion of titanium and alloys was again increased.

R25-300

Principal production version, used for the MiG-21bis and Su-15bis. Service life 400 hours. Manufactured as Product 25-11, about 3,200 being delivered in 1972-86. Also produced under licence by HAL (see under India) in 1977 for MiG-21bis.

R25-300-94

Remanufactured for longer life and enhanced reliability for MiG-21-93. Ratings unchanged.

Type

Two-shaft turbojet with afterburner.

Intake

No separate intake structure.

LP Compressor

Three-stage compressor with no inlet guide vanes or variable stators, with conical rotating spinner ahead of first stage with 21 large titanium blades.

HP Compressor

Five-stage HP spool. All rotor blades dovetailed into titanium discs carried on short tubular shafts. Mass flow 68.5 kg (151 lb)/s. Pressure ratio (1F) 9.55, (ChR) 9.8.

Combustion Chamber

Can-annular, with 10 flame tubes housed in casing welded from refractory sheet, with ceramic-coated air-film liners.

Turbines

Single-stage HP and LP turbines. Rotor blades of high-nickel alloy, with integral tip shrouds, inserted into discs. HP blades air-cooled. TET (1F) 1,037ºC, (ChR) 1,087ºC.

Afterburner

Three concentric spray rings and flameholder gutters spaced axially. Reprofiled multiflap variable nozzle driven by hydraulic rams.

Accessories

Main gearbox under centre of compressor casing with vertical tower shaft from HP spool. Pads for fuel and oil pumps, starter/generator, aircraft hydraulic pump and tachogenerator.

Control System

Hydromechanical, with separate afterburner/nozzle control giving infinitely variable reheat.

Dimensions

Length overall	4,615 mm (181.7 in)
Max diameter	907 mm (35.7 in)
Height overall	1,191 mm (46.9 in)

Weight

Dry	1,215 kg (2,679 lb)

Performance Ratings

(S/L, ISA)

T-O with afterburner:	
Regime 1F	66.05 kN (15,100 lb st) at 11,500 rpm
Regime ChR	68.47 kN (15,653 lb st) at 11,150 rpm
Regime Chr (Mach 1)	95.47 kN (21,825 lb)
Max dry	40.26 kN (9,050 lb st) at 11,150 rpm
Normal (cruise)	33.35 kN (7,496 lb st)

Specific Fuel Consumption

Regime 1F	63.67 mg/Ns (2.25 lb/h/lb st)
Regime ChR (static)	70.74 mg/Ns (2.5 lb/h/lb st)
Max dry	27.2 mg/Ns (0.96 lb/h/lb st)
Normal	25.78 mg/Ns (0.91 lb/h/lb st)

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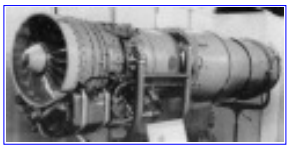

R25-300

HAL-built R25-300 with afterburner removed

Section through R25-300 (2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

MOTOR, GNPP (STATE SCIENTIFIC PRODUCTION ENTERPRISE)

R195

This engine was derived from the R95Sh by Gavrilov's team in 1986. The objective was to increase survivability in the face of enemy fire. The most severe requirement was that the engine should stand up to 23 mm gunfire and continue to operate after suffering damage in eight places. This is believed still to be unique to this engine.

The R195 entered production in 1987 at UMPO Ufa. Design TBO is 500 hours. The R195 powers the Su-25T, Su-25TK, Su-25UB, Su-28 and Su-39.

R195PF

Projected derivative with afterburner, with T-O rating of 60.8 kN (13,668 lb st). By 2001 this had not been tested, though it is the preferred engine of the proposed Sukhoi S-54.

Type

Two-shaft turbojet.

LP Compressor

Three stages. No inlet guide vanes or variable stators. Mass flow 66 kg (145.5 lb)/s.

HP Compressor

Five stages. No variable stators, but auto bleed valves. Pressure ratio (T-O) 9.0, (emergency) 9.35.

Combustion Chamber

Can-annular with multiple duplex burners. Cleared for kerosene, diesel oil and MT petrol.

Turbines

Single-stage HP, single-stage LP. Entry gas temperature (T-O) 915ºC, (emergency) 977ºC.

Jetpipe

No afterburner, simple fixed-area nozzle with central tube to suck out cooling air.

Control System

Duplicate hydromechanical giving single-lever control.

Dimensions

Length	2,880 mm (113.4 in)
Diameter	805 mm (31.7 in)

Weight

Dry	860 kg (1,896 lb)

Performance Ratings

(S/L, ISA)

T-O	42.17 kN (9,480 lb st)
Emergency	44.13 kN (9,921 lb st)

Specific Fuel Consumption (T-O):

	25.21 mg/Ns (0.89 lb/h/lb st)

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R195 display exhibit (plastic disc over inlet) (Piotr Butowski)

R195
(2001)

Section through R195
(2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

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MOTOR, GNPP (STATE SCIENTIFIC PRODUCTION ENTERPRISE)

R13

Gavrilov's most important work was to improve Tumanskii's R11 turbojet, and the initial result was the R13, which was first qualified in 1966.

Keeping the engine as nearly as possible interchangeable with the R11 in installation, the compressor was improved to increase the surge margin, the HP spool increased to five stages, the combustion chamber modified to improve high-altitude relight without the need for a separate starting tank of petrol and a completely new afterburner fitted. The proportion of titanium (previously very small) was increased; for example, titanium alloys were used for the compressor discs.

R13-300

Initial production version. Qualified in 1966, and produced in 1968-86 by UMPO (which see), and also by UMPK at Ulan-Ude as Product 95-1 (showing the close relationship with the R95, which see). Approximately 12,500 of all versions were delivered for the MiG-21SM, SMT and MF and for the Su-15M and TM. Several versions were later produced by LMC in China (which see) as the WP13.

R13F-300

Whereas the R13-300 had radial spray bars in the afterburner and a perforated liner, this version had the form described below, together with an improved variable nozzle. A special emergency afterburning regime was qualified (in flight only), and overhaul life was increased to 1,500 hours, with intermediate inspection/repair at 500-hour intervals.

R13F2-300

A special variant for the Su-15TM, with unchanged ratings.

Type

Two-shaft turbojet with afterburner.

Intake

No separate intake structure.

LP Compressor

Three-stages. No inlet guide vanes or variable stators. Hemispherical rotating spinner.

HP Compressor

Five-stages. Construction mainly titanium alloy, with rotor blades dovetailed into discs carried on short tubular shafts at each end. Mass flow (R13) 65.6 kg (144.6 lb)/s, (R13F) 66.0 kg (145.5 lb)/s. OPR 9.25 at 11,150 rpm.

Combustion Chamber

Can-annular, with 10 flame tubes housed in casing welded from refractory sheet, with ceramic-coated liners providing a cooling air film.

HP Turbine

Single-stage, with shrouded aircooled blades of high-nickel alloy. TET (max continuous) 920ºC, (first afterburning regime) 952ºC, (emergency) 1,005ºC.

LP Turbine

Single stage, with solid inserted blades.

Afterburner

Three concentric spray rings and flameholder gutters spaced axially. Multiflap variable nozzle driven by hydraulic rams.

Control System

Hydromechanical, with separate afterburner/nozzle control giving three stages of reheat.

Accessories

Main gearbox under centre of compressor casing with vertical tower shaft from HP spool. Pads for fuel and oil pumps, starter/generator, aircraft hydraulic pump and tachogenerator.

Dimensions

Length overall	4,605 mm (181.3 in)
Max diameter	907 mm (35.7 in)
Height overall	1,095 mm (43.1 in)

Weight, Dry

R13-300	1,205 kg (2,657 lb)
R13F-300	1,134.6 kg (2,501 lb)

Performance Ratings (S/L, ISA)

T-O with afterburner	63.65 kN (14,307 lb st) at 11,150 rpm
Emergency (3 min)	64.73 kN (14,550 lb st)
Max dry	39.92 kN (8,973 lb st) at 11,150 rpm

Specific Fuel Consumption

T-O, as above	59.23 mg/Ns (2.093 lb/h/lb st)
Max dry, as above	26.37 mg/Ns (0.931 lb/h/lb st)

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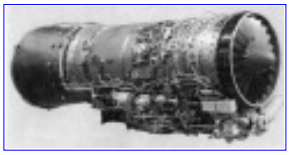

R13-300, afterburner removed

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 12 April 2001

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OMSK BARANOV - MOTOR-BUILDING ENTERPRISE NAMED FOR P I BARANOV

283 B Khmelnitsky St, 644021 Omsk
Tel: (+7 3812) 33 00 63
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Telex: 133112 MARS

General Director: Nikolai B Lvov

Founded in 1916, this claims to be "the biggest aircraft engine plant in Russia". Though it adjoins the OMKB, this is a distinct plant concerned solely with production, maintenance and repair. Its first important contract was to produce the OMKB's GTD-3. It also overhauled the AL-7. Engines handled today include the AL-21, NK-86, PS-90A, RD-33, TV7-117 and TVD-20. According to Klimov Corporation, this factory was one of two which in 2000 was in series production with the RD-33 family, the other being Chernishev. It also produces APVs, notably the VSU-10 for the Il-86 and Il-96.

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AERO-ENGINES - TURBOPROP/TURBOSHAFT, RUSSIA

Date Posted: 12 April 2001

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OMSK ENGINE DESIGN BUREAU

TV-O-100

This engine was designed jointly with AMNTK 'Soyuz' to provide a modern core in the 537 kW (720 shp) class, with modular construction, which could be developed in both turboshaft and turboprop forms.

TV-O-100

Basic turboshaft version. Ratings, contingency 537 kW (720 shp); T-O 522 kW (700 shp); max cruise 343 kW (460 shp). Initially certificated in 1989 and produced by Mars at Omsk (now OMKB) for Ka-126. It was originally the intention that this helicopter should be manufactured under licence in Romania, and that the further development of the engine should be carried out in collaboration with the Romanian industry. The TV-O-100 has potential for 857.5 kW (1,100 shp), with pressure ratio 10.2 and TET of 1,077ºC. A version flat rated at 529 kW (710 shp) is a candidate for the Ka-128.

TV-D-100

Turboprop version, with remote two-stage reduction gearbox driving tractor or pusher propeller. T-O rating 529 kW (710 shp). Candidate engine for Aeroprogress/ROKS Aero T-610 and other projected Russian aircraft.

CIAM, the Central Institute for Aviation Motors, has for five years been testing a heat exchanger inserted between the compressor delivery and combustion chamber (see drawing) with which sfc could be reduced by 15 to 25 per cent. The following description refers to the turboshaft version:

Type

Free-turbine turboshaft.

Intake

A scoop above the engine leads via a large particle extractor to a drum surrounding the core, from where the air passes to the compressor.

Compressor

Two axial stages, with variable IGVs and intermediate stator vanes, followed by one centrifugal. Mass flow 2.66 kg (5.86 lb)/s. Pressure ratio 9.2.

Combustion Chamber

Annular folded reverse-flow, with 12 forward-facing fuel nozzles around rear face and two igniters spaced 120º apart.

Compressor Turbine

Single stage with solid blades of advanced alloy. TGT 1,027ºC.

Power Turbine

Single stage overhung behind rear bearing. Output 6,000 rpm.

Jetpipe

Short pipe exhausting direct to rear, with extension to suit installation.

Output

The entire front of the engine is a large gearbox with two stages of spur gears, driven by a long quill shaft from the power turbine, providing a 6,000 rpm output at the top.

Accessories

The gearbox provides seven pads for engine and airframe accessories, normally including an electric starter/generator and a separate alternator.

Control System

Hydromechanical, with FADEC being developed.

Dimensions

Length	1,275 mm (50.2 in)
Width	780 mm (30.7 in)
Height	735 mm (28.9 in)

Weight

Dry	125 kg (275.5 lb)

Performance Ratings

See model listing

Specific Fuel Consumption

T-O	96.5 µg/J (0.571 lb/h/shp)
Cruise	108.44 µg/J (0.641 lb/h/shp)

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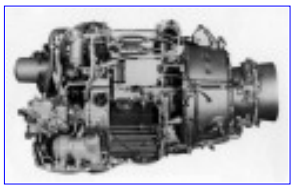
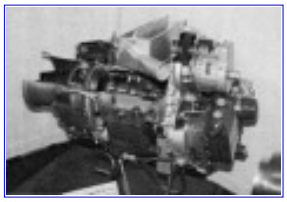
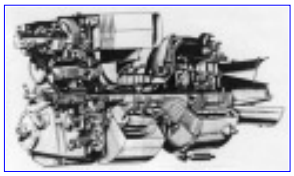


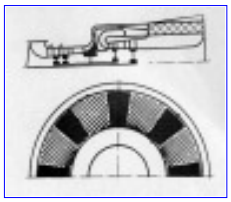
TV-O-100 left side



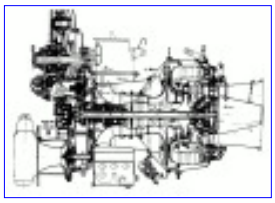
Cutaway TV-O-100



Cutaway and sectioned drawing of TV-O-100



Proposed heat exchanger for TV-O-100



Longitudinal section through TV-0-100 (2001)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

OMSK ENGINE DESIGN BUREAU

TRDD-50

Previously known as the TV-50, this small turbofan was designed to power the RKV-15B cruise missile (NATO AS-15 Kent). It has now been developed for a range of applications as a booster engine, and as the propulsion for small manned and unmanned vehicles. A design objective was minimal fuel consumption.

Type

Two-shaft turbofan.

Fan

Single-stage axial, without inlet guide vanes, with 18 titanium blades of low aspect ratio.

Compressor

Single-stage centrifugal HP compressor, with impeller of titanium.

Combustion Chamber

Exceptionally compact annular reverse-flow.

HP Turbine

Single-stage HP turbine.

LP Turbine

Single-stage, with integrally shrouded blades.

Dimensions

Length	850 mm (33.46 in)
Diameter	330 mm (13.0 in)

Weight

Dry	95.0 kg (209 lb)

Performance Ratings

S/L, static	3.92-4.90 kN (882-1,102 lb st)

Specific Fuel Consumption

As above	18.41 mg/Ns (0.65 lb/h/lb st)

TRDD-50M

This upgrade would add axial compressor stages behind the fan and ahead of the centrifugal. It would have the following characteristics:

Dimensions

Length	1,000 mm (39.37 in)
Diameter	470 mm (18.51 in)

Weight

Dry	130 kg (286.6 lb)

Performance Ratings

S/L, static	7.14 kN (1,323 lb st)

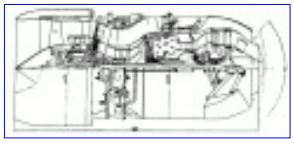
Specific Fuel Consumption

As above	13.87 mg/Ns (0.49 lb/h/lb st)

UPDATED

Two views of TVD-50 (inverted on missile pylon)

Two views of TVD-50 (inverted on missile pylon)



Longitudinal section through TRDD-50M with reverser (2001)

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 12 April 2001

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OMKB (OEDB) - OMSK ENGINE DESIGN BUREAU

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Teletype: 216274 MARS

Chief Designer: Vladimir G Kostogryz
Tel: (+7 095) 38 12 33 49 84
Deputy Chief Designer: Vladimir I Ustyugov
Tel: (+7 095) 38 12 33 70 86

This design bureau was formed in 1956, the original General Designer being V A Glushenkov. In partnership with the Central Institute for Aviation Motors, it began the development of small helicopter turboshaft engines in 1957. The first product was the 224 kW (300 hp) GTD-1, followed by the GTD-5 and -5M, which continue in production. The first major engine for aircraft propulsion was the GTD-3, produced from 1964 as a twin package plus common reduction gear for helicopters. In 1970 came the TVD-10 turboprop. Apart from APUs, the latest Omsk engines available from production are the TVD-10, TVD-20 and TVD-50. In 1996 the TV-O-100 was completing certification testing.

The OMKB, which for a time used the name Mars, also produces the VSU-10 APU fitted to the Il-86 and Il-96, and is developing the VGTD-43 APU intended for the Tu-204 and related aircraft. Other products include gas turbines for surface applications, turbostarters, ram-air turbines and self-contained fans.

Note: The English-language alternative to OMKB is OEDB.

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-321

With the experience of the NK-12, NK-22 and NK-25 behind it, the Kuznetsov team was the obvious choice in 1976 to develop the engine of Aircraft 70, which became the Tu-160 strategic bomber and missile carrier (NATO name `Blackjack'). The engine programme at Kuibyshyev was supervised by Ye M Semenov and the Chief Designer was Ye A Kuzmin. The chosen engine, the **NK-32**, first ran in 1980, but it was dropped in favour of an improved engine.

NK-321

This was the designation of the improved engine, differing from the NK-32 mainly in having an overhung first fan stage, ahead of the intake frame. This first ran in 1981, and four flight-cleared engines were supplied to Kazan in the same year, enabling the prototype Tu-160 to fly before the end of that year. Series production began in late 1983, and engines were shipped from 1985, about 250 being delivered of a planned total of over 400 for the intended production run of 100 aircraft (to match the USAF procurement of 100 B-1Bs). In the event, funding ran out at the 36th aircraft, and a number of engines remain in storage unflown.

NK-321-44

Modified to suit installation in Tu-144 08-2 (77114), for lease to NASA. Four unused engines leased from CIS air force, two (reconditioned) purchased from Dvigatel NK.

Type

Three-shaft augmented low-bypass ratio turbofan.

Compressors

Three-stage LP (fan); five-stage IP; seven-stage HP. Designed for maximum efficiency and highest overall pressure ratio. First stage also designed for minimal radar reflectivity of any radiation managing to reach it. Materials titanium, steel and (final stages) high-nickel alloy. Mass flow 365 kg (805 lb)/s. Pressure ratio (T-O) 28.2. Bypass ratio 1.36.

Combustion Chamber

Annular, paired vaporising burners, in inner and outer sections to give no visible smoke and near perfect uniformity of temperature at HP turbine face.

HP Turbine

Single-stage, diameter about 1,000 mm (39.34 in), cooled blades of single-crystal material. Entry gas temperature 1,630ºK (1,357ºC).

IP Turbine

Single-stage, DS blades.

LP Turbine

Two-stage, DS blades.

Afterburner

Designed for peak efficiency and maximum thrust for lowest gas temperature to minimise IR signature. No visible smoke. Downstream of multilobe flow mixer. Fully variable con/di nozzle.

Control System

Electrical, with hydromechanical back-up. Studies in progress for later switch to FADEC.

Dimensions

Length	7,453 mm (293.4 in)
Inlet diameter	1,700 mm (66.93 in)

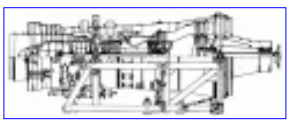
Weight

Dry	3,650 kg (8,047 lb)

Performance Ratings (ISA, S/L)

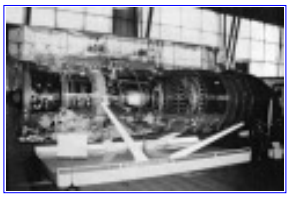
Max T-O	245 kN (55,077 lb st)
Max dry	137.2 kN (30,843 lb st)

UPDATED



Longitudinal section through NK-36, an industrial engine with the core of the NK-321 (2001)

NK-32 (Samara) (2001)



NK-321 (plastic-skinned cutaway exhibit)

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-12

This unique engine family has for over 45 years been, by a wide margin, the most powerful turboprop in the world, and no successor is in sight (though Kuznetsov has made bold attempts with the NK-62 and NK-110, described in other entries). The story began with Kuznetsov's TV-022 turboprop, designed in 1947-49 by a team made up mainly of German prisoners, most ex-Junkers, led by Austrian Ferdinand Brandner. The TV-2 first ran in June 1949 and was qualified in October 1950.

By coupling two of these engines into a common gearbox, Kuznetsov produced the 2TV-2F, rated at 9,200 kW (12,333 shp). This ran in September 1951 and was qualified in December 1952. Four powered the Tupolev 95-1 (or 95/I) strategic-bomber prototype, first flown on 12 November 1952. This crashed, and for various reasons Kuznetsov urgently decided to design a replacement engine with a single power section.

TV-12

Prototype engines. First run on dynamometer brake October 1952, and on Tu-4LL flight test bed with AV-60 propeller in 1953.

NK-12

Preproduction engines, rated at 9,200 kW (12,337 shp), cruise rating (11 km, M 0.68) 4,778 kW (6,407 shp). Fitted with 5.6 m (18 ft 4 in) AV-60 eight-blade coaxial propeller. Fitted to Tupolev 95-2 (95/II), first flown 16 February 1955, and to initial series Tu-95. TBO 150 hours. No longer in use.

NK-12M

Full production by UMPO and Motorostroitel. T-O rating 11,025 kW (14,785 shp, but loosely described as '15,000 hp'). State qualified 19 June 1956 and produced with 5.6 m AV-60K propeller giving T-O thrust at S/L of 88.3 kN (19,841 lb st). Fitted to Tu-95M. TBO 300 hours.

NK-12MV

Ratings unchanged. Matched with 5.6 m AV-60N propeller incorporating rapid autofeather system. State qualified September 1958. Fitted to all series versions of Tu-95, Tu-114, Tu-116 and Tu-142. TBO 300 hours, service life 5,000 hours.

NK-12MA

T-O rating unchanged, but cruise rating (10 km, M 0.56) 5,940 kW (7,966 shp). Minor differences from MV, and matched with AV-90 propeller of 6.2 m (20 ft 4 in) diamater, giving T-O thrust 137 kN (30,793 lb st). First tested June 1963, State qualified July 1965, in production 1966. Powers An-22. Service life 4,500 hours.

NK-12MP

Ratings unchanged from NK-12MV, but redesigned in detail for longer life and extended TBO. First tested 1978, State qualified September 1979. Powers all surviving Tu-95MS and Tu-142 versions.

NK-12MK

T-O rating unchanged, but cruise rating (1.5-15 m, M 0,345-0.43) 7,828 kW (10,498 shp). Based on NK-12MA, driving same 6.2 m propeller. State qualified October 1974. Fitted to *Orlyonok* family of Ekranoplans (wingships).

NK-12ST

Family of versions for ground power applications, most rated at 6,300 or 8,000 kW. Three-stage compressor turbine and independent power turbine with rear drive. Exported to Argentina, Bulgaria and Poland.

Type

Single-shaft turboprop, with split drive to two coaxial propellers.

Intake

Aluminium casting. Annular air inlet around central propeller gearbox. Front bearing and propeller loads taken through six radial struts.

Compressor

14 stage axial, with large controllable blow-off valves in casing. Stators all of fixed incidence, slotted round casing made from four sectors precision-welded from steel sheet. Rigidity ensured by four tie rods linking front frame and engine mount to compressor delivery casing. Mass flow at 8,300 rpm (MV) 55.8 kg (123 lb)/s. Pressure ratio varied by blow-off valves and altitude between 9.5 and 13.0.

Combustion Chamber

Can-annular type. Outer casing houses a ring of 12 quasi-conical flame tubes which at the rear merge into a continuous annular ring. Downstream airspray injectors (24) from two fuel manifolds. See drawing overleaf.

Turbine

Five stages, with blades held by fir-tree roots in air-cooled discs. TET (NK-12) 977~C, (12MV) 1,060~C.

Jetpipe

Diffusing outlet pipe with six aerofoil struts carrying rear bearing. All aircraft installations have a bifurcation which leads to exits on each side of the nacelle.

Output

Single quill shaft from the front of the compressor carries a pinion providing the input to a double planetary system with a ratio of 0.09 (11.067). This turns concentric output shafts driving the two independent contrarotating propellers. Integral torquemeter.

Starting

Automatic starting cycle, with main shaft run up to self-sustaining rpm by 48.5 kW (65 hp) TS-12 gas-turbine starter (itself started electrically).

Control System

Hydromechanical, maintaining a constant speed of 8,300 rpm, output power being varied by electronic control of fuel flow in conjunction with propeller pitch.

Dimensions

Length:	
NK-12M, MV	4,785 mm (188.4 in)
NK-12MA	5,005 mm (197.0 in)
Diameter (inlet and jetpipe flange)	1,190 mm (46.85 in)

Weight, Dry

NK-12M	2,900 kg (6,393 lb)
NK-12MV	3,065 kg (6,757 lb)

Performance Ratings

(See individual model listings)

Specific Fuel Consumption

T-O, S/L:	
NK-12	83.80 Ág/J (0.496 lb/h/shp)
NK-12M, MV, MP	78.21 Ág/J (0.463 lb/h/shp)
Cruise (11,000 m, 36,090 ft, M0.68):	
NK-12	61.45 Ág/J (0.364 lb/h/shp)
NK-12M, MV, MP	58.846 Ág/J (0.348 lb/h/shp)

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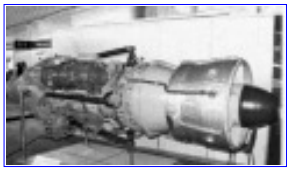
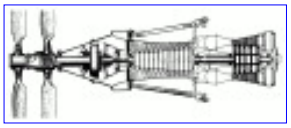
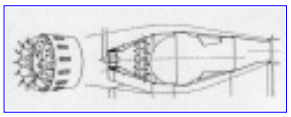


NK-12MV (Piotr Butowski)

NK-12MV
(2001)



Simplified longitudinal section through NK-12
(2001)



NK-12 combustion chamber and flame tube

NK-12MV left side, showing TS-12 starter

NK-12MV from rear

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 12 April 2001

Jane's Aero-Engines 10

ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-88, NK-89

Samara/Trud has been working since about 1968 on the use of LH_2/LNG (liquid hydrogen and liquefied natural gas) for various gas-turbine engines. A particularly important development programme was the conversion of an NK-8-2 engine to burn either of the two new liquids, and of the tankage, piping and control system of a Tu-154 to accept such an engine.

NK-88

Work began using LH_2, which because of its intensely cold temperature (near absolute zero) required extremely careful design and choice of materials. The cryogenic programme has been supervised by V N Orlov, chief designer NK-88 is N Ye Priyampolskii and chief engineer G A Yakovlev. NK-8-2U test engines were modified to burn LH_2 converted to gas in a heat exchanger. The cryogenic control system was contained in an insulated box above the compressor case. Bench testing began in February 1980.

The test aircraft was redesignated Tu-155. The test engine was installed in the No 3 (starboard) position (the cutaway drawing shows the centre engine, which was a normal NK-8-2U). Flight testing began on 15 April 1988, 12 flights being made on this fuel.

NK-89

In March 1989 bench testing began on the NK-89. This engine was designed to run on LNG, with which low-temperature problems are much less severe. Supervised by Orlov, the chief engineer is V V Kharlamov. The

cryogenic control system was again grouped above the front of the engine, supplying liquid fuel to the heat exchanger in the jetpipe behind the core. Later in 1989 the test engine was installed in the Tu-155, replacing the NK-88. A demonstration flight was made Moscow-Bratislava-Nice-Moscow, in May 1990 the Tu-155 attended the Hanover Air Fair, and in 1991 the ILA in Berlin.

Tupolev and partners have for a decade been seeking funds to build and fly derived aircraft designated Tu-156S and Tu-156M to use cryogenic fuels. These would have a large insulated tank for the cryogenic fuel in the rear fuselage, while retaining the normal kerosene tankage. These aircraft are intended to lead to later aircraft with the projected NK-94 or NK-112 engines.

The following are specification parameters which differ from the basic NK-8-2U:

Fan

Bypass ratio (NK-88) 1.06, (NK-89) 1.05.

HP Compressor

Core pressure ratio (NK-88) 11.0, (NK-89) 10.73.

HP Turbine

TGT (NK-88) 867ºC, (NK-89) 882ºC.

Thrust Ratings

Unchanged.

Weight, Dry

With cryopump and heat exchanger but no reverser	2,300 kg (5,071 lb)

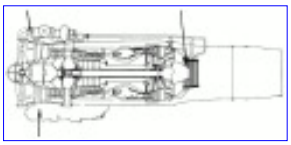
Specific Fuel Consumption

Max T-O:	
NK-88	6.23 mg/Ns (0.22 lb/h/lb st)
NK-89	15.89 mg/Ns (0.561 lb/h/lb st)
Cruise (11,000 m, 36,090 ft; M0.8)	
NK-88	8.38 mg/Ns (0.296 lb/h/lb)
NK-89 (LNG)	20.97 mg/Ns (0.74 lb/h/lb)
NK-89 (propane)	20.54 mg/Ns (0.725 lb/h/lb)
NK-89 (kerosene)	22.95 mg/Ns (0.81 lb/h/lb)

UPDATED

NK-88

Internal arrangement of Tu-155; 1, technicians; 2, control engineers; 3, hydrogen and helium tanks; 4, guest cabin; 5, buffer zone; 6, hermetically sealed fuel cabin; 7, auxiliary drain/vent; 8, main drain and vent; 9, main control complex; 10, nitrogen bottles



Section through NK-89. Arrows show the cryogenic control system, the heat exchanger in the core jetpipe and, underneath, the original accessory group (2001)

NK-88 with control box open (2001)

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 12 April 2001

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ST PETERSBURG NPO IM KLIMOV

TV7-117S

Described as a third-generation engine, the TV7-117 has a modular core incorporating advanced features and materials, and envisaged as the basis for various jet and shaft engines.

TV7-117S

Turboprop version, selected (as **TV7-117-3**) to power the Il-114, produced in collaboration with Polish industry. Certificated for this application January 1997. Also selected for Il-112, four-engined Aeroprogress (ROKS-Aero) T-274 Titan and Tu-130. Flight tested on two Il-76s and the prototype Il-114. Drives Stupino SV-34 propeller with six composite blades (diameter 3,600 mm, 141.7 in). Certificated in 1993, with series production at Chernyshov and Mars (Omsk).

TV7-117S Series 2

Described separately.

TVD-117E

Turboprop tailored to Ekranoplan propulsion, rated at 1,840 kW (2,467 shp). Three power the Raketa 2.2.

TV7-117M

Same ratings as TV7-117S. Selected for Aeroprogress (ROKS) T-710.

TV7-117V

Also known as VK-3000. Described separately.

A growth core with two centrifugal stages is under development. Output will be in the 2,237 kW (3,000 shp) class, and will be unaffected by sand or dust. Other developments include liquefied gas fuels and a turbofan with T-O rating of 15.68 kN (3,527 lb st).

The following description refers to the basic TV7-117S:

Type

Free turbine turboprop.

Compressor

Annular ram inlet around reduction gear tapers to entry to five-stage axial compressor, with variable inlet guide vanes and next two stators, followed by centrifugal stage on same shaft. Mass flow 7.95 kg (17.53 lb/s). Pressure ratio 16.

Combustion Chamber

Annular folded reverse flow. Minimum pollution with wide range of fuels, including LNG and LPG.

HP Turbine

Two-stage gas generator turbine with cooled blades. Entry temperature 1,227ºC.

Power Turbine

Two-stage.

Output

Planetary type, with new tooth profiles and anti-vibration mountings.

Starting

Pneumatic air turbine, mass flow 0.2 kg (0.44 lb)/s.

Control System

FADEC, with separate automatic control for ground and flight operation.

Dimensions

Length	2,143 mm (84.37 in)
Height	940 mm (37.0 in)
Width	886 mm (34.88 in)

Weight

Dry	530 kg (1,168 lb)

Performance Ratings

Max T-O	1,839 kW (2,466 shp) to 35ºC and 250 m (820 ft)
Cruise (6,000 m; 19,685 ft at 270 kt; 500 km/h; 311 mph)	1,342 kW (1,800 shp)

Specific Fuel Consumption

Cruise, as above	67.09 µg/J (0.397 lb/h/shp)

UPDATED

Simplified longitudinal cross-section through TV7-117 (2001)

Two views of the TV7-117S with SV-34 propeller

Two views of the TV7-117S with SV-34 propeller

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 12 April 2001

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GRANIT - MACHINE-BUILDING DESIGN BUREAU 'GRANIT'

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Tel: (+7 095) 369 80 13
Fax: (+7 095) 366 10 10
Telex: 205390 NEPTUN

General Manager: Semyon R Sarkisov

This design bureau was founded in 1945 as OKB-45 by V Ya Klimov and A A Mikulin. It played a central role in development of the former's VK-1. In 1963 it became OKB-45-165, being named MKB 'Granit' in 1966. In 1982 it was reorganised as part of 'Saturn' (see Lyul'ka Saturn). Today it is also a production plant, which delivered large numbers of the most powerful supersonic turbojets of the AL-7, AL-21 and R-15B families. It is also a leading design and production organisation for small gas turbines for the propulsion of manned aircraft, unmanned vehicles and cruise missiles, and for APUs.

Today it continues on a reduced scale to develop aero-engines and gas-turbines rated at 20 to 500 kW (26.8 to 670 shp) and 20 MW (26,820 shp) for surface applications. It is eager to find foreign partners. Current small engines include the MD-120 turbojet for unmanned vehicles, the new TVD-150 family and the TVD-400 turboshaft.

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 09 January 2001

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ST PETERSBURG NPO IM KLIMOV

TV7-117S Series 2

This engine was designed specifically to provide improved propulsion for the Il-114. It is also aimed at the Il-112, MiG-110 and other aircraft.

The air intake and all the main modules have been refined in design, to give a significant increase in power and fuel efficiency, combined with a reduction in weight. The integrated engine/propeller FADEC control and monitoring system is new, and Aerosila has produced a propeller with improved aerodynamics. The Series 2 engine is designed for a 20,000 hours service life, with a TBO of 6,000 hours.

The following are principal specifications:

Compressor

Pressure ratio 17.

HP Turbine

Max entry temperature 1,257ºC.

Dimensions

Length	2,136 mm (84.09 in)
Max diameter	940 mm (37.0 in)

Weight, Dry

	430 kg (948 lb)

Performance Ratings

T-O, S/L	2,088 kW (2,800 shp) to ISA + 35ºC
OEI	2,610 kW (3,500 shp) to ISA + 15ºC
Max continuous	1,715 kW (2,300 shp) to ISA + 35ºC
Cruise (S/L, static)	1,491 kW (2,000 shp) to ISA + 35ºC

Specific Fuel Consumption

T-O	70.64 µg/J (0.418 lb/h/shp)
Cruise, as above	63.20 µg/J (0.374 lb/h/shp)

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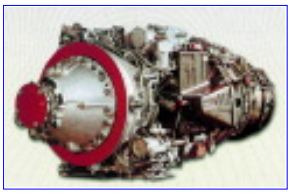

TV7-117S Series 2
(2001)

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Date Posted: 09 January 2001

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ROSVOOROUZHENIYE - STATE CORPORATION FOR IMPORT/EXPORT OF ARMS

18/1 Ovchinnikovskaya Quay, 113324 Moscow
Tel: (+7 095) 231 00 49
Fax: (+7 095) 233 02 72
Telex: 411428

DIRECTOR-GENERAL: Alexey Ogarev

This organisation manages the export of all Russian military equipment. In 1998, it managed sales totalling US$2 billion, and in 1999 this increased to almost US$2.6 billion. Contracts with many countries extending up to 2005 are expected to result in sales of about US$9 billion. About half of this is accounted for by military aeroplanes and helicopters.

In December 1999 Director-General Ogarev was interviewed by *Jane's Defence Weekly*. Among other things he commented ``We are establishing aircraft maintenance and repair enterprises outside Russia. The volume of work in this field in 1999 will total about US$429 million. For example we plan to carry out maintenance work on Indian AF aircraft and helicopters and their engines. We will establish a facility in China specialising in the production of spares and repair of AL-31A engines. In Ethiopia a centre is being organised to support the MiG-21bis, MiG-23 and their engines.

``Military technical co-operation with Ukraine is increasing. The best example of this is the Medium Transport Aircraft Consortium, the contractor for the An-7X derived from the An-70. This will be a strong rival to the C-17, C-130J and the non-existent A400M".

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 09 January 2001

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CIAM - CENTRAL INSTITUTE OF AVIATION MOTORS

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Fax: (+7 095) 267 13 54
Telex: 411686 CIAM RU
e-mail: CIAM@AVIM.msk.ru

DIRECTOR: Prof Donat A Ogorodnikov

Established in 1930, and named for P I Baranov, this was one of the gigantic research and development centres of the Soviet Union. Despite drastic cuts in funding, it remains an immensely capable and impressive establishment, with massive facilities and a highly talented and experienced staff of scientists and engineers. Though heavily committed to tasks of an immediate nature, such as testing prototype engines and parts and development of test facilities and methods. CIAM's underlying work is fundamental research and providing scientific underpinning of the various RFAS engine design and production teams. It also performs market forecasts, programme co-ordination and the drawing up of specifications for future engines. At the *Engines `96* show in Moscow in June 1996 a major part of the CIAM exhibit was devoted to a test programme on an engine designed to propel vehicles at 6,500 km/h (4,039 mph) burning hydrogen. In 2000 it had large teams working on environmental and fuel-usage problems.

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 09 January 2001

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ASSAD - ASSOCIATION OF AERO-ENGINE MANUFACTURERS

16 Ulansky Avenue, 101849 Moscow
Tel: (+7 095) 207 01 71, 09 97, 04 70 or 925 15 64
Fax: (+7 095) 207 54 91, 56 67, 09 62

PRESIDENT: Viktor M. Chuiko

This organisation was established by Viktor M Chuiko, a former Deputy Minister of Aviation Industry. Its objectives are to assist national strategic planning throughout the Russian Federation and Associated States (CIS) (RFAS) and foster international economic links. It is open to membership by foreign companies, many of whom have joined. In the summers of 1992, 1994, 1996 and 1998 ASSAD, with many other Russian organisations, has staged an International Engines Trade Fair in Moscow. Despite intense efforts by ASSAD and others, overall production in the Russian aero-engine industry was in 1999 running at only 8 to 10 per cent of the 1990 level, though the labour force had remained at almost half the 1990 level. By 1999 it was at last being recognised that there have to be firmer moves towards consolidation and rationalisation.

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 09 January 2001

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ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

NK-8

Development of this two-shaft bypass turbojet (low-BPR turbofan) began in May 1961, to power the Il-62 long-range passenger transport. Experience with the huge NK-6 supersonic-bomber engine proved valuable, and also facilitated development of the NK-8 into the NK-144 engine for the Tu-144 SST, and the NK-22 and NK-25 for the Tu-22M. The NK-144 is no longer in use. Several versions of the NK-8 were developed:

NK-8 Series I

Initial version for Il-62. Inlet fitted with splinter ring dividing core flow from bypass flow, with 15 radial guide vanes in core flow and 30 around outer periphery. No longer operational.

NK-8 Series III

Modified with new inlet with 12 radial struts, with intermediate part-span guide vanes, modified reverser and increased TET. First run September 1965, passed State test April 1967, in production 1966-68.

NK-8-4

Refined and upgraded engine for long-range passenger version of Il-62. Increased OPR and TGT, improved reverser increasing length but reducing weight. Tested at OKB-276 in October 1963, flown on Il-62 July 1965, in production at Factories 16 (Kazan) and 24 (Kuibyshyev) 1968-79, Overhaul life 2,000 hours, total life increased from 7,000 to 9,000 hours, with objective of 18,000 hours.

NK-8-2

Version developed for Tu-154 by team supervised by A A Ovcharov, with leading designer M T Vasilishin. Different reverser with inclined cascade exits for side engines, integral APU rated at 110 kW (150 hp). Tested at Factory 276 in January 1967, certificated January 1970, initial version of Tu-154 certificated 25 August 1971, in production 1970-72.

NK-8-2U

Improved engine for all Tu-154 versions prior to Tu-154M. Weight reduced, and cleared to much greater cruise rating. Target TBO 8,000 hours, total life 18,000 hours. First tested 13 December 1972, State test May 1973.

NK-8-4K

Developed by team led by V S Gagai for *Orlyonok* (Eaglet) ekranoplans. Many changes to suit sea-level marine environment, revised fuel, lubrication and diagnostic systems, used at starting to blow under wing. First engine test 1972, certification October 1979.

NK-8-5I

Improved engine for Il-62M. Increased HP airflow and increased TGT. First tested September 1973, certificated November 1974. Overhaul life 5,000 hours.

NK-86, 86, 88, 89

Derived engines, described separately.

The following details refer to all versions later than the NK-8 Series I and Series III.

Type
Two-shaft bypass turbojet (low-BPR turbofan).

Intake
Fabricated light alloy, fixed geometry, 12 radial struts with interspersed short guide vanes around outer periphery. Bleed-air anti-icing.

Fan
Two stages, first stage rotor having 27 blades of kinked (swept forward, then back) shape, with part-span clappers. Pressure ratio (typical) 2.15 at 5,350 rpm. Mass flow (8 Series I) 214.5 kg (473 lb)/s, (8-4) 222 kg (489.4 lb)/s, (8-2U) 228 kg (503 lb)/s, (8-4K) 227 kg (500.4 lb)/s, (8-5I) 235 kg (518 lb)/s. Bypass ratio (8, Series I and III) 0.984, (8-4) 1.04, (8-2, 8-2U) 1.05, (8-5I) 1.034. Diameter (all) 1,355 mm (53.35 in).

HP Compressor
Six stages, mainly of titanium alloy. Core pressure ratio, (8 Srs I and III) 10.25 at 6,835 rpm, (8-4) 10.8 at 6,950 rpm, (8-2) 9.6, (8-2U) 10.7 at 6,835 rpm, 11.14 in cruise, (8-5I) 11.1.

Combustion Chamber
Annular, with 139 burners. Claimed to produce no visible smoke.

HP Turbine
Single-stage turbine, all with shrouded rotor blades, air-cooled discs and hollow nozzle blades (stators). All shafting carried between shock-absorbing bearings at each end, with labyrinth and contact (rubbing) graphite seals to prevent gas leakage. Maximum TGT (8 Srs I) 927ºC, (8 Srs III) 947ºC, (8-4) 982ºC, (8-2) 927ºC, (8-2U) 957ºC, (8-4K) 987ºC, (8-5I) 1,002ºC.

LP Turbine

Two stages.

Jetpipe
Mixer leads bypass flow into common jetpipe which may be fitted with blocker/cascade type reverser giving up to 48 per cent (NK-8-2, 45 per cent) reverse thrust, and noise suppressor.

Accessories
These include automatic flight deck warning of vibration exceeding permissible limit, ice and fire. All accessories group beneath fan duct casing. Engine claimed to need no attention for long periods, other than inspection of fuel and oil filters. RTA-26-9-1 turbine gas temperature controller by Smiths Industries.

Starting
HP spool driven by constant-speed drive type PPO-62M, or started pneumatically by air from APU type TA-6, from ground hose or by air from another engine (NK-8-2, pneumatic starter only). Time to idling speed not over 80 s. Engine can be windmill-started in the air under all conditions, up to altitudes of 11,000 m (36,090 ft).

Fuel Specification
T-1 and TS-1 to GOST 10227-62 or T-7 to GOST 12308-66 (equivalent to Avtur 50 to DERD.2494 or MIL-F-5616.

Oil System
Continuous pressure feed and recirculation. Oil consumption not over 1.3 kg (2.87 lb)/h. Pressure not less than 2.28 kg/cm^2 (33 lb/sq in).

Oil Specification
Mineral oil MK-8 or MK-8P to GOST 6457-66 (DERD.2490 or MIL-O-6081B). External tank on left side of front casing.

Dimensions

Diameter:	
NK-8, 8-4, 8-2U	1,442 mm (56.8 in)
NK-86	1,600mm (63.0 in)
Length:	
NK-8, with reverser	4,766 mm (187.6 in)
NK-8-4, with reverser	5,101 mm (200.8 in)
NK-8-2U, with reverser	4,762 mm (187.5 in)

Weight, Dry

NK-8 (both), with reverser	2,500 kg (5,511 lb)
NK-8-4, without reverser	2,200 kg (4,850 lb)
NK-8-4, with reverser	2,440 kg (5,379 lb)
NK-8-2, without reverser	2,150 kg (4,740 lb)
NK-8-2U, without reverser	2,170 kg (4,784 lb)
NK-8-2U, with reverser	2,350 kg (5,181 lb)
NK-8-4K	2,200 kg (4,850 lb)
NK-8-5I, with reverser	2,400 kg (5,291 lb)

Performance Rating

(T-O, S/L):	

NK-8 (both), NK-8-2	93.2 kN (20,950 lb st)
NK-8-4, NK-8-2U, NK-8-4K	103 kN (23,150 lb st)
NK-8-5I	107.9 kN (24, 250 lb st)
(cruise, 11,000 m, 36,090 ft; 850 km/h; 530 mph):	
Nk-8 (both)	22.1 kN (4,960 lb)
NK-8-4, NK-8-5I	27.0 kN (6,064 lb)
NK-8-2	17.66 kN (3,968 lb)
NK-8-2U	21.58 kN (4,846 lb)

Specific Fuel Consumption

S/L, static:	
NK-8 Series I	17.55 mg/Ns (0.62 lb/h/lb st)
NK-8 Series III	17.85 mg/Ns (0.63 lb/h/lb st)
NK-8-2	16.93 mg/Ns (0.598 lb/h/lb st)
NK-8-2U	16.42 mg/Ns (0.58 lb/h/lb st)
NK-8-4K	17.27 mg/Ns (0.61 lb/h/lb st)
NK-8-5I	16.99 mg/Ns (0.60 lb/h/lb st)
Cruise, as above:	
NK-8 (both)	23.50 mg/Ns (0.83 lb/h/lb)
NK-8-4	24.65 mg/Ns (0.81 lb/h/lb)
NK-8-2	22.38 mg/Ns (0.79 ln/h/lb)
NK-8-2U	21.70 mg/Ns (0.766 lb/h/lb)
NK-8-5I	22.10 mg/Ns (0.78 lb/h/lb)

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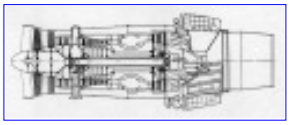
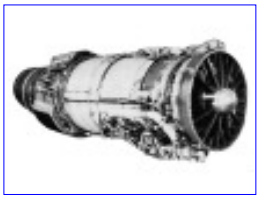
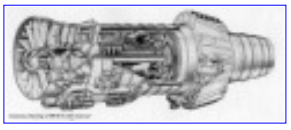
NK-8-1 (2001)



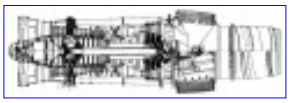
Simplified longitudinal section through NK-8-4 with reverser



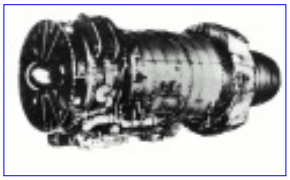
NK-8-4



Cutaway drawing of NK-8-4 with reverser

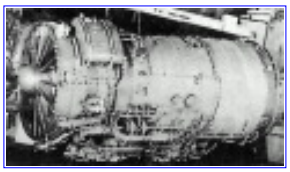


NK-8-4
(2001)



NK-8-5I
(2001)

NK-8-2U
(2001)



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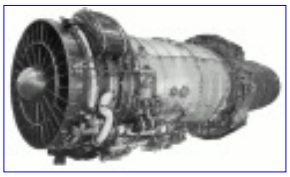

NK-8 Series III
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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 23 November 2000

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RYBINSK MOTORS JSC

Rybinsk RD-38

This simple engine was the first designed in the Soviet Union purely to provide lift for V/STOL aircraft. It was produced in 1980-89, entered service in 1984 on the Yak-38, and was subsequently used on the Yak-38M.

RD-38A

Booster engine for An-71, with long-life lubrication system. Installed horizontally in rear fuselage. In limited production 1985. T-O thrust 27.5 kN (6,063 lb st). Sfc, as RD-38.

Type
Single-shaft turbojet for near-vertical installation.

Compressor
Six-stage axial. Mass flow 45.2 kg (99.65 lb)/s. Pressure ratio 5.2.

Combustion Chamber
Annular.

Turbine
Single-stage with aircooled blades.

Weight, Dry
226 kg (498.2 lb)

Performance Rating (S/L)
32.50 kN (7,165 lb st)

Specific Fuel Consumption
39.63 mg/Ns (1.4 lb/h/lb st)

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RD-38
(2001)

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 23 November 2000

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RYBINSK MOTORS JSC

Rybinsk D-30KU-154 Series III

A full description of the D-30KU appears under the original design bureau, now called Aviadvigatel. Rybinsk Motors has produced large numbers of these engines, and in 1999 began marketing the Series III. This incorporates numerous generally minor modifications which enable the same ratings to be achieved at reduced TET. These include reduction in radial clearances, introduction of noise-absorbing structures, replacement of the propulsive nozzle by an improved 12-lobe mixer and replacement of the reverser cascades by a type with slanting ribs to eliminate FOD. These changes can be made during routine shop visit. They give reduced noise and emissions to meet ICAO 2000 requirements, a 3 per cent reduction in fuel burn, and a service life extension of 800 hours. Fan diameter is unchanged at 1,455 mm (57.28 in), but engine dry weight (without reverser) has been reduced from 2,318 kg (5,110 lb) to 2,307 kg (5,086 lb).

Rybinsk Motors has published the following comparison:

	D-30KU-154 II	**D-30KU-154 III**
At T-O rating of 103 kN (10,500 kg, 23,148 lb st):		
Mass flow	263 kg (580 lb)/s	265 kg (584 lb)/s
Turbine entry temperature	1,063ºC (1336K)	1,043ºC (1316K)
Specific fuel consumption	14.055 mg/Ns (0.498 lb/h/lb st)	13.604 mg/Ns (0.482 lb/h/lb st)
At cruise rating of 26.97 kN (6,063 lb) at 11,000 m (36,090 ft) and M0.8:		
Specific fuel consumption	20.11 mg/Ns (0.71 lb/h/lb)	19.546 mg/Ns (0.69 lb/h/lb)

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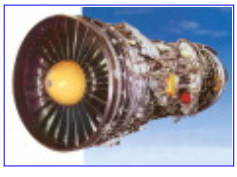

D-30KU-154 Series III (2000)

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RYBINSK MOTORS JSC

Rybinsk D-277

Lack of money has seriously delayed progress with this potentially important high-BPR turbofan. It was to have been the principal engine based on the Type 77 core, which was designed in the late 1980s and first ran on the bench in 1992. Back in 1990, RKBM publicised its plan to use the 77 gas generator in a family of turboshaft and turboprop engines of 4,480 to 5,970 kW (6,000 to 8,000 shp), a turbofan of 117.5 kN (26,455 lb st) with two core booster stages, and an advanced propfan with a third core booster stage with a thrust up to 147.1 kN (33,070 lb st).

D-277

By 1992, the decision had been taken to concentrate on this turbofan, which matched the requirements of several existing or forthcoming aircraft (stretched Tu-334, An-180 and Yak-46 and re-engined Tu-154, Il-76 and Il-86) and could provide a quickly maturing Russian competitor to the Ukrainian D-27 propfan. It adds additional core booster and LP turbine stages. The complete engine has yet to run.

Type
Two-shaft turbofan.

Fan
Single stage, with aerodynamically advanced titanium blades. Mass flow 416 kg (917 lb)/s. Bypass ratio 7.4.

LP Compressor
Four core booster stages, rotating with the fan.

HP Compressor
Nine stages, with six stages of variable stators. Engine OPR 32.

Combustion Chamber
Annular, designed for minimum emissions.

HP Turbine
Single stage, with air-cooled monocrystal blades. TGT 1,488ºC.

LP Turbine
Five stages. Active clearance control on both turbine casings.

Jetpipe
Multilobe forced mixer leading to single jetpipe.

Reverser
Translating cowl blocks core and opens peripheral cascade ring.

Control System
FADEC, with dual redundancy, incorporating diagnostic and maintenance functions.

Dimensions

Length	4,400 mm (173.2 in)
Overall diameter	1,852 mm (72.9 in)

Weight, Dry
2,350 kg (5,181 lb)

Performance Ratings

T-O, S/L	122.8 kN (27,649 lb st)
Cruise (11,000 m, 36,090 ft at M0.8)	31.1 kN (7,000 lb)

Specific Fuel Consumption

Cruise, as above	15.84 mg/Ns (0.559 lb/h/lb)

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RKBM exhibit showing future engines based on a common core

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 23 November 2000

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NK - ND KUZNETSOV SCIENTIFIC AND TECHNICAL COMPLEX

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Telex: 214172 TSEL

General Director: Evgenii A Gritsenko
Deputy Chief Designer: L M Shirkin

Previously with a name meaning labour or toil, this is the large engine design and prototype-construction bureau, which for more than 40 years from 1949 was headed by Nikolai D Kuznetsov, and in his honour has recently reverted to using his initials. The bureau was formed by A A Mikulin in Moscow, and in 1941 it was hastily evacuated to Kuibyshyev, 900 km (559 miles) to the east. Gas-turbine work was launched with the aid of large numbers of German engineers, the first fully successful engine to go into service being the TV-12 turboprop which was redesignated as the NK-12.

Today, more than 45 years later, this is still by an enormous margin the most powerful turboprop in service in the world. In the 1980s, the NK design team developed and tested an even more powerful turboprop. This engine, the NK-62, is described in the pages which follow, and its designers wish funding could be made available to develop a large aircraft to use it. They are convinced there is a need for such an engine, and are astonished that West European nations should be trying to produce an engine (for the proposed A400M aircraft) only half as powerful. Also described is a later high-power turboprop, the NK-110 which has been tested on the bench without its pusher propellers.

The bureau has always concentrated on large high-power engines. A landmark in 1956 was the first run of the very large NK-6 two-shaft afterburning turbojet, which was later developed into a family of NK-144 engines, which are no longer operating. As well as the engines listed, this bureau has been a pioneer of large rocket engines, especially high-energy types burning liquid hydrogen. It claims to have produced 34 per cent of the gas turbines for Russian gas-pumping stations. As *Konversiya* products it is also developing small piston engines. Today Kuibyshyev has reverted to its pre-Communist name of Samara.

The NK complex claims that it had, by 1999, become the largest single aero-engine group in Russia. Apart from the original design bureaux and associated factories, it now includes major production plants, among them KMPO and SMPO, which see. In November 1995 the Council of Experts in Economy awarded the title ``Russian economy leader'' to the Kuznetsov STC selected from 5,000 entrants.

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 23 November 2000

Jane's Aero-Engines 09

CHERNYSHEV - V V CHERNYSHEV STATE MOSCOW MECHANICAL ENGINEERING PRODUCTION ENTERPRISE

7 Vishnevaya St, 123362 Moscow
Tel: (+7 095) 491 58 74
Fax: (+7 095) 490 56 00
Tx: ELIKA

General Director: Dr J. A. Leikovsky

Founded in 1932 as a Soviet State Aviation Factory (GAZ), this enterprise produces two important aero-engines, the RD-33 (chiefly for the MiG-29) and TV7-117S (for the Ilyushin 114). It also produces a range of *konversiya* consumer products. An alternative name is the V V Chernyshev (Moscow) Machine-Building Enterprise. In a move to strengthen Russian industry this production plant has become part of the MAPO-MiG enterprise.

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AERO-ENGINES - TURBOSHAFT, RUSSIA

Date Posted: 30 August 2000

Jane's Aero-Engines 08

MACHINE-BUILDING DESIGN BUREAU 'GRANIT'

GRANIT TVD-400

This family of shaft engines is derived from the MD-120 turbojet. TVD-400 versions are being developed as turbofans, turboprops and turboshaft engines, which are expected to find applications in civil and military aeroplanes and helicopters. Granit emphasize the core's basic simplicity, which `should lead to low cost of ownership'. The following refers to the TVD-400A turboshaft engine.

Type
Free-turbine turboshaft.

Compressor
Single-stage axial followed by single-stage centrifugal. Mass flow 2.15 kg (4.74 lb)/s. Pressure ratio 7.0.

Combustion chamber
Annular, with eight burners.

Compressor turbine
Single-stage axial, 52,000 rpm.

Power turbine
Single-stage axial.

Output
Modular gearbox driven by shaft at rear of power turbine. Output speed selectable 2,500-6,000 rpm.

Starter
Pneumatic or electrical.

Control system
Electronic.

Accessories
Core drives fuel pump and alternator.

Fuel grade
Any kerosene.

Dimensions

Length	820 mm (32.28 in)
Width (without reduction gear)	380 mm (14.96 in)
Height	400 mm (15.75 in)

Weight, Dry

	96.0 kg (211.64 lb)

Performance Ratings (S/L)

	298.3 kW (400 shp)

Specific Fuel Consumption

	104.3 µg/J (0.617 lb/h/shp)

NEW ENTRY

TVD-400A (2000)

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 30 August 2000

Jane's Aero-Engines 08

MACHINE-BUILDING DESIGN BUREAU 'GRANIT'

GRANIT MD-120

Though initially intended for unmanned applications, this simple turbojet could be developed into a long-life derivative for ultralight aircraft. Granit is also testing the smaller MD-45, of similar configuration, with a thrust at sea level of 0.61 kN (136.7 lb st).

Type

Single-shaft turbojet.

Compressor

Single-stage axial followed by single-stage centrifugal. Mass flow 2.1 kg (4.63 lb)/s. Pressure ratio 7.0.

Combustion chamber

Annular, with eight burners.

Turbine

Single-stage axial, 52,000 rpm.

Fuel grade

JP-4, JP-5, Avtur NATO F-35.

Dimensions

Length	1,290 mm (50.79 in)
Diameter overall	265 mm (10.43 in)

Weight, Dry

	35.0 kg (77.2 lb)

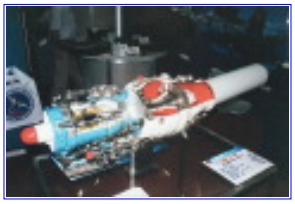
Performance Ratings

S/L, ISA, static	1.175 kN (264.6 lb st)

Specific Fuel Consumption

	29.45 mg/Ns (1.04 lb/h/lb st)

NEW ENTRY



MD-120 (2000)

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 30 August 2000

Jane's Aero-Engines 08

ST PETERSBURG NPO IM KLIMOV

VKS-800

This multipurpose engine was disclosed at the Moscow `Engines 98' exhibition, at which time testbed running was about to begin. It is derived from the TVa-3000 turboshaft engine.

VKS-800S

Turboprop for fixed-wing aircraft of about 1 tonne (2,205 lb) payload.

VKS-800V

Turboshaft for helicopters of about 1.5 tonnes (3,307 lb) payload, a particular target application being the Ka-126.

GTP-0.5

Ground mobile power plant.

Intake
Inwards radial.

Compressor
Two-stage centrifugal. Pressure ratio 12.

Combustion Chamber
Annular, folded, 20 airspray burners on rear face.

Compressor Turbine
Single stage, uncooled. Inlet 1,050~C (1,922~F).

Power Turbine
Single stage.

Dimensions

Length:	
VKS-800S	1,276 mm (50.24 in)
VKS-800V	1,270 mm (50.0 in)
Height:	
VKS-800S	589 mm (23.19 in)
VKS-800V	590 mm (23.23 in)
Width (both)	555 mm (21.85 in)

Weight, Dry

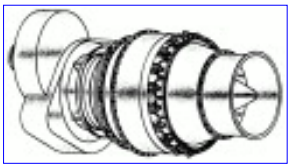
VKS-800S	125 kg (275.6 lb)
VKS-800V	110 kg (242.5 lb)

Performance Rating (ISA, S/L)

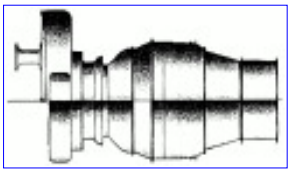
T-O (both)	597 kW (800 shp)
Cruise:	
VKS-800S	462 kW (620 shp)
VKS-800V	425-462 kW (570-620 shp)

Specific Fuel Consumption

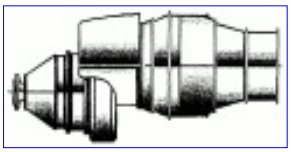
T-O:	
VKS-800S	87.39 Ág/J (0.517 lb/h/shp)
VKS-800V	88.59 Ág/J (0.524 lb/h/shp)
Cruise:	
VKS-800S	73.70 Ág/J (0.436 lb/h/shp)
VKS-800V	92.96-96.85 Ág/J (0.550-0.573 lb/h/shp)



VKS-800V (computer generated) (1998)



VKS-800V (1998)



VKS-800S (1998)

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AERO-ENGINES - TURBOSHAFT, RUSSIA

Date Posted: 30 August 2000

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ST PETERSBURG NPO IM KLIMOV

TV3-117

This second-generation turboshaft engine was designed in the late 1960s. The first versions went into production in 1972. Since then more than 10,000 have been supplied to over 40 countries. Since 1990 two completely dissimilar turboprop versions have also been developed.

TV3-117MT

T-O rating 1,434 kW (1,923 shp). Initial production turboshaft version, for later versions of Mi-8, and most versions of Mi-14, Mi-17 and Mi-24.

TV3-117V

T-O rating 1,633 kW (2,190 shp). Powers Ka-27, Ka-29, Ka-31 and Ka-32.

TV3-117VK

T-O rating 1,838 kW (2,465 shp). Electronic control suitable for wide range of fuels including LNG, propane and butane. Powers Ka-52 and Mi-28N.

TV3-117VM

T-O rating (Series 01) 1,545 kW (2,070 shp). (Series 02) 1,618 kW (2,170 shp). Electronic control. Powers later versions of Mi-17 including Kazan-built versions, as well as Mi-8AMT and Mi-28.

TV3-117VMA

New compressor with increased efficiency. T-O rating 1,633 kW (2,190 shp). Powers Ka-32A, Ka-50 and most versions of Mi-35.

TV3-117VMA-SB3

Growth version with same gas generator as SB2 turboprop. Performance: contingency (to 25ºC) 2,088 kW (2,800 shp); T-O (to 30ºC) 1,864 kW (2,500 shp); max cruise (S/L, to 30ºC) 1,417 kW (1,900 shp). Powers Ka-32M, Ka-50H and intended for Mi-14P retrofit, Mi-18 and Mi-24M.

Type
Free-turbine turboshaft.

Compressor
12-stage axial with inlet guide vanes and first three stators variable. Mass flow 8.7-9.0 kg (19.18-19.84 lb)/s. Pressure ratio (early versions) 9.0, (VMA versions) 9.4.

Combustion Chamber
Annular, improved from TV2-117.

Compressor Turbine
Two-stages. Max TET (early versions) 917ºC (VMA versions) 990ºC.

Power Turbine
Two stages.

Output
Rear drive behind compact side-curved exhaust unit.

Starting
MT pneumatic, other versions electric.

Dimensions

MT, V, VK:	
Length	2,085 mm (82.1 in)
Width	640 mm (25.5 in)
Height	725 mm (28.5 in)
VM, VMA:	
Length	2,055 mm (80.9 in)
Width	650 mm (25.6 in)
Height	728 mm (28.7 in)

Weight, Dry

MT, V, VK	285 kg (628 lb)

VM	294 kg (648 lb)
VMA-SB3	310 kg (683 lb)

Performance Ratings
See variants list.

Specific Fuel Consumption (S/L, T-O)

MT, V, VK	96.3 µg/J (0.57 lb/h/shp)
VM, VMA	85.6 µg/J (0.507lb/h/shp)

Turboprop Versions:

TV3-117VMA-SB2

This version was designed by Klimov in partnership with Motor Sich and SMNPP *Yupiter* (Jupiter), both of Ukraine. It is in production by Motor Sich. In order to drive the propeller the basic TV3-117VMA turboshaft is fitted with a primary gearbox at the rear to raise the output to a drive shaft passing across the top of the engine to a propeller gearbox at the front. This provides an output at 900 to 1,200 rpm to a Stupino AV-140 six-blade propeller of 3,720 mm (146.5 in) diameter. This engine was to be produced by Motor Sich for An-140, and also intended for Il-112, Il-114, MiG-110, MiG-111 and Tu-130. In May 1997 it was reported that 60 engines for the An-140 will be made in Iran under a US$145 million intergovernment deal. However, in 1998 it appeared that, following pressure to make the An-140 an almost wholly Ukraine-made product, the engine in production aircraft would be the AI-30 Series 1, which see under Ivchenko Progress. This engine is derived from the TV3-117MA-SB2.

Dry Weight

	560 kg (1,235 lb)

Performance Ratings

Contingency (S/L, ISA+25ºC)	2,088 ekW (2,800 ehp)
T-O (S/L, ISA+28ºC)	1,864 ekW (2,500 ehp)
Cruise (6,000 m; 19,685 ft, M0.52)	1,268 ekW (1,700 ehp)

Specific Fuel Consumption

T-O as above	81.26 µg/J (0.481 lb/h/ehp)
Cruise, as above	72.63 µg/J (0.430 lb/h/ehp)

TV3-117VMA-SBM

This improved turboprop is the result of a combined effort by Klimov with two design bureaux in Zaporozhye, Ukraine: Ivchenko Progress and SMNPP Jupiter. The manufacturer is again Motor Sich. It uses essentially the same gas generator as the previous turboprop, but is neater in having the output shaft taken through the centre of the engine to a reduction gear centred in the inlet. The latter has accordingly been redesigned as an annular ring of increased diameter. This engine has FADEC control, and is expected to power later versions of the An-140.

Weight, Dry

490 kg (1,080 lb)

Performance Ratings

Contingency (S/L, flat-rated to 42ºC)	1,864 ekW (2,500 ehp)
T-O (S/L, flat-rated to 36ºC)	1,790 ekW (2,400 ehp)
Maximum cruise (6,000 m; 19,685 ft, MO.52)	1,342 ekW (1,800 ehp)

Specific Fuel Consumption

as TV3-117VMA-SB2

UPDATED

TV3-117

Cutaway drawing of TV3-117

Two views of TV3-117 VMA-SB2 (1998)

Two views of TV3-117 VMA-SB2 (1998)

TV3-117VMA-SBM

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 30 August 2000

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ST PETERSBURG NPO IM KLIMOV

RD-33

The contract for this important fighter engine was won in competition with two other engine bureaux, and it was designed before Isotov's death. Detail design began in 1968, meeting the requirements of the MiG-29, and the first bench run took place in 1972. Development of new versions continues in 1999 under the direction of Chief Designer Valentin Stavroitenko.

RD-33

Initial production version. Ratings (max dry) 49.4 kN (11,110 lb st), (a/b) 81.4 kN (18,300 lb st). Powers MiG-29 and MiG-29UB. Deliveries were initiated in 1976, and mass production at Chernyshov and Omsk followed in 1981. TBO for the RD-33 Series 1 was 300 hours, increased in Series 2 engines to 1,600 hours and to 2,000 hours in the Series 3. Basis for SMR-95 (see Aerosud-Marvol in International section).

RD-33K

Uprated version, with greater air flow and increased turbine entry temperature; 86.0 kN (19,335 lb st). Powers MiG-29K and -29M. Special anti-corrosive coatings on all blading.

RD-33N

Klimov designation for SMR-95 (see below). Accessories on underside and modified lubrication system. Thrust ratings as RD-33. Powers Super Mirage F1 and Super Cheetah D2. Available from Klimov for MiG-21 upgrades.

RD-93

Accessories on underside and other minor changes. Thrust ratings as RD-33. Engine of Chinese (Chengdu/Mikoyan) FC-1 (Super-7). Possibly to be produced under licence by LMC (see under China).

SMR-95

See under Aerosud-Marvol in International section.

RD-133

This is basically an RD-33 to the latest production standard, fitted with KLIVT (Klimov's Vecloring Thrust) axisymmetric nozzle with vectoring to any position within a cone of 15º semi-angle. Control of nozzle angle is effected by the conventional hydromechanical flight controls of the MiG-29 by a simple interlink. The RD-133 is installationally interchangeable with the RD-33, and the nozzle can be retrofitted to existing RD-33 engines. T-O rating (a/b) 88.25 kN (19,841 lb st), max dry 54.89 kN (12,346 lb st).

RD-333

Though derived from the RD-33 this is regarded by Klimov as a new `fifth-generation' engine. Features include a new fan handling 85 kg (187.4 lb)/s, a rebladed compressor, an HP turbine with entry gas temperature up to 1,527ºC and a rectangular 2-D nozzle vectoring in the vertical plane. Maximum thrust is to be 98.1 kN (22,057 lb st). Testing complete engines is to start in 1999. The RD-333 is intended for the LMFI (Light Multirole Fighter), also known as the MiG-35.

VKS-5

Fourth-generation engine, unaugmented. HP turbine gas temperature up to 1,527ºC. T-O thrust 51.46 kN (11,574 lb st) with sfc 20.95 mg/Ns (0.74 lb/h/lb st). Dry weight 885 kg (1,951 lb). Length 2,900 mm (114 in).

VKS-10

Augmented version of VKS-5.

The following refers to the original production RD-33:

Type
Two-shaft afterburning bypass turbojet (low-ratio turbofan).

LP Compressor
Four stages. Front bearing carried in four-strut nose, but no inlet guide vanes. Mass air flow 76.0 kg

(167.5 lb)/s. Bypass ratio 0.49.

HP Compressor
Nine stages. Overall pressure ratio 21.

Combustion Chamber
Annular with air-blast fuel nozzles giving generally smokeless combustion of a range of fuels.

HP Turbine
Single-stage with single-crystal cooled blades. Maximum entry gas temperature, 1,257ºC at T-O, 1,407ºC in flight.

LP Turbine
Single-stage turbine.

Afterburner
Combustion in both core and bypass flows. Nozzle with fully variable area and profile in primary and secondary flows. Outer nozzle has 24 flaps. Vectoring nozzles under preliminary development by CIAM and Soyuz.

Accessories
Tank for IPM-10 oil, hydromechanical fuel control and auxiliaries grouped above engine to reduce cross-section. Closed lubrication system functions under all positive or negative *g*-loads. Multipurpose self-diagnostic system.

Dimensions

RD-33, RD-133:	
Length	4,229 mm (166.50 in)
Inlet diameter	730 mm (26.74 in)
Max diameter	1,000 mm (39.37 in)
RD-33N:	
Length	5,440 mm (214.17 in)
Max diameter	1,040 mm (40.94 in)
RD-93:	
Length	4,230 mm (166.54 in)
Max diameter	1,040 mm (40.94 in)

Weight, Dry

RD-33, RD-93	1,055 kg (2,326 lb)
complete power plant	1,217 kg (2,683 lb)
RD-33N	1,295 kg (2,855 lb)
RD-133	1,145 kg (2,524 lb)

Performance Ratings
See model listing; time from idle to max afterburner 4.0 s. Maximum Mach number 2.35.

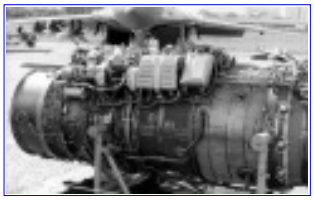
Specific Fuel Consumption

RD-33, RD-33N, RD-93:	
Max augmented	52.40 mg/Ns (1.85 lb/h/lb st)
Max dry, S/L	21.8 mg/Ns (0.77 lb/h/lb st)
RD-133:	
Max dry, S/L	22.08 mg/Ns (0.78 lb/h/lb st)

UPDATED

RD-33

RD-33 afterburner and nozzle



RD-33 accessories

RD-33K

Cutaway drawing of RD-33
(1998)

Vectoring nozzle for RD-133 and VKS-10. The logo is that of the Klimov Corporation (Yefim Gordon)
(1998)

RD-33N
(2000)

RD-93
(2000)

RD-133 with KLIVT Thrust Vectoring Nozzle
(2000)

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 17 August 2000

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SKBM - SAMARA MACHINE-BUILDING DESIGN BUREAU

29 Zavodskoye St, 443009, Samara
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GENERAL DIRECTOR: Gennady A Burmistrov

JSC `SMBDB' (SKBM in Russian) is currently designing and testing various aircraft gas-turbine engines, generating plants to 10 MW, piston engines up to 75 kW (100 hp) and other products. A major part of its current activity is increasing engine reliability and service life.

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Date Posted: 17 August 2000

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MOTOROSTROITEL - MOTOROSTROITEL JSC

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GENERAL DIRECTOR: Dr Igor L. Shitarev
DEPUTY GENERAL MARKETING DIRECTOR: Igor Kuprijanov

Founded in October 1912 this is one of the largest and longest-established engine production plants. It has also been known as the Samara Motor-Production Organisation named for M.V. Frunze, and even as SEBPA (Samara Engine-Building Production Organisation). It manufactures a range of aircraft gas turbines (mainly of NK design) and derived engines for pumping and electricity generation, large rocket engines for space launchers and other products.

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 17 December 1999

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JSC `AVIADVIGATEL' - PERM SCIENTIFIC AND PRODUCTION ENTERPRISE `AIRCRAFT ENGINES'

RD-35

This is the Russian designation of the DV-2 turbofan, produced by PSLM (see under Slovakia). If the Yak-130 goes into production it would probably be powered by the RD-35, which would then be produced by Klimov under licence. A competitor is the AI-22 (see under Ivchenko Progress, Ukraine).

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 17 December 1999

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RYBINSK-GE AVIATION MOTORS

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GENERAL DIRECTOR: Vladimir Raschupkin
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This joint venture between Rybinsk Motors and GE Aircraft Engines was agreed on 30 May 1996. Essentially it will co-produce GE aero and aero-derived gas-turbine engines, assembled, tested and subsequently supported at Rybinsk, for use in Russian aircraft and surface applications. The first jointly produced engines are the CT7-9 turboprop for the Sukhoi S-80, the CT7-2D1 turboshaft, which is an option for the Kamov Ka-64 and the LM2500 industrial/marine engine.

In January 1998, it was reported that the Rybinsk management was discussing the possibility of assembling or co-producing the CFM56 family of engines (see CFMI introduction in International section).

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 17 December 1999

Jane's Aero-Engines 07

JSC `AVIADVIGATEL' - PERM SCIENTIFIC AND PRODUCTION ENTERPRISE `AIRCRAFT ENGINES'

MANUFACTURER DETAILS

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Fax: (+7 83422) 45 97 77
Tx: 134802 LAVA SU
Teletype: 134135 LAVA

GENERAL DESIGNER: Yuri Yevgen'evich Reshetnikov
HEAD OF FOREIGN ECONOMIC RELATIONS: Edward G. Tsyrlin
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This large design bureau, until 1990 named MKB and now Joint Stock Company `Aviadvigatel', was founded before the Second World War by Arkadiya D Shvetsov, who was followed in 1953-89 by P A Solovyov. The engines they created powered 26 types of production aeroplanes and helicopters.The first Perm gas turbine, the 4,176 kW (5,680 shp) TV-2M, performed excellently but never went into production. In 1962 the D-20P became the first turbofan to enter scheduled service.Aggregate flight

time of Solovyov's engines exceeds 60 million hours. They are fitted to the Tu-124, Tu-134, Tu-154M, Il-76, Il-62M, MiG-31, A-40 and M-55, and Mi-6 and Mi-10 helicopters. These are flown by 39 countries.

The PS-90A, Solovyov's last design, is fitted to the Tu-204, Il-76MF and Il-96-300. The engines designed by `Aviadvigatel' are manufactured by Perm Motors, except for the D-30KU family, which are made at Rybinsk.Many of the new designs are based on the PS-90A core: the PS-90A2, PS-90A10, PS-90A12, PS-90A76, D-30KU-90 and D-100.Attention is also being paid to propfans with contrarotating fans, with a bypass ratio of about 15. Another area of design is industrial gas-turbine plant for gas pumping and power generation. Aviadvigatel has been able to preserve its personnel, and is developing international links.

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AERO-ENGINES - TURBOJET, RUSSIA

Date Posted: 26 October 1999

Jane's Aero-Engines 07

ST PETERSBURG NPO IM KLIMOV - KLIMOV CORPORATION

VK-1

When in 1946 the British government foolishly shipped Rolls-Royce Nene turbojets to Moscow, this engine was immediately dissected and put into production as the RD-45, the designation reflecting the GAZ number of the production factory. General Designer Klimov also at once set about improving this engine and increasing its power. This had to be done without increasing the engine's overall size, because the uprated engine, designated VK-1, had to be installationally interchangeable with the RD-45.

VK-1

The main task was to increase the mass flow, and this was done by increasing the size of the compressor, which in turn resulted in a reduction in rpm, which in turn required increasing the size of the turbine. The mass flow was increased from 40.5 to 48.0 kg (105.8 lb)/s, while keeping total engine dry weight down to 870 kg (1,918 lb), an increase of 56 kg. T-O rating was increased to 26.52 kN (5,952 lb st). Large numbers are still flying, mainly in the MiG-15UTI.

VK-1A

Refined version with improved materials and greater potential for development. Ratings unchanged. Standard on later MiG-17 and Ilyushin 28 versions.

VK-1F

Fitted with afterburner. Dry weight 978 kg (2,156 lb). T-O rating 33.18 kN (7,452 lb st). Still flying, mainly in versions of MiG-17.

WP5

Produced under licence in China, initially from June 1956 by SEF (later called LM) and later by XAE.

WP5A

Version with slight modifications tailored to H-5 (licence-built Ilyushin 28). Increased demand from 1963 led to the production line being transferred to XAE.

WP5D

Modified variant produced in large numbers by XAE for JJ-5 trainer.

WP5B

Produced from 1966 for retrofit to Soviet-supplied MiG-15bis fighters.

WP5C

Modified long-life version produced from 1976 for retrofit to Soviet-supplied MiG-17PFU interceptors.

DIMENSIONS (typical):

Length, without afterburner	2,640 mm (103.9 in)
Diameter	1,273 mm (50.1 in)

WEIGHT, DRY:

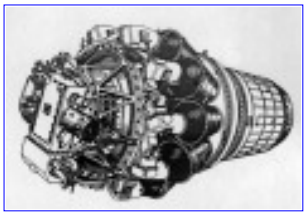
Typical, without afterburner	872 kg (1,922 lb)

PERFFORMANCE RATINGS: See under individual model listing

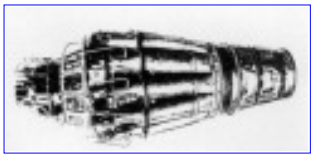
SPECIFIC FUEL CONSUMPTION (T-O):

VK-1	30.3 mg/Ns (1.07 lb/h/lb st)
VK-1F (dry)	32.57 mg/Ns (1.15 lb/h/lb st)
VK-1F (afterburner)	56.65 mg/Ns (2.0 lb/h/lb st)

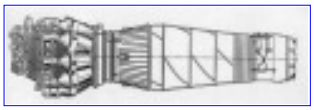
RD-45 (Soviet copy of Nene) (2000)



Russian drawing of VK-1 (1996)



VK-1A (1996)



VK-1F (1996)

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AERO-ENGINES - TURBOPROP/TURBOSHAFT, RUSSIA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

TV-116-300

Now under development, on the basis of a proven experimental gas generator, this engine is intended for light aeroplanes and helicopters.
TYPE: Turboprop or turboshaft with single-shaft gas generator.
COMPRESSOR: Two-stage centrifugal.
COMBUSTION CHAMBER: Annular folded reverse-flow.
TURBINES: Two-stage gas generator, with cooled blades. Two-stage power turbine with drive shaft to front of engine and suitable reduction gearbox for application. Soyuz says ``gearbox like that of Garrett (AlliedSignal) TPE 351".
ACCESSORIES: Bevel drive to tower shaft at 12 o'clock position above inlet.
DIMENSIONS:

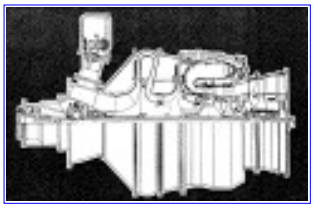
Length (without gearbox)	800 mm (31.5 in)
Diameter	420 mm (16.54 in)
WEIGHT, DRY (without gearbox):	150 kg (331 lb)

PERFORMANCE RATING (ISA, S/L):

Max T-O	805 kW (1,080 shp)

SPECIFIC FUEL CONSUMPTION:

Max T-O	78.2µg/J (0.463 lb/h/shp)



Provisional longitudinal section through TV-116-300 (1997)

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AERO-ENGINES - TURBOSHAFT, RUSSIA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

TV-O-100-300

This engine was designed at AMNTK Soyuz in 1983 to provide a modern core in the 537 kW (720 shp) class, with modular construction, which could be developed in both turboshaft and turboprop forms. It was initially developed in partnership with Omsk MKB to power the Ka-126. The engine is described under OEDB.

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

SMPO - SAMARA MOTOR PRODUCTION ORGANISATION

MANUFACTURER DETAILS

Named after M V Frunze
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Fax: (+7 8462) 27 16 00, 27 16 70

GENERAL DIRECTOR: I Shitarev

This organisation, which also styles itself M V Frunze Engine-Manufacturing Co and SEBPA (Samara Engine-Building Production Association) is now a joint stock company fighting for survival after having made engines in large numbers to the designs of others since 1912. It had close links with the Mikulin and Kuznetsov design bureaux, and it also produced the R-7A rocket engines used in all flights by Cosmonauts. It still makes small numbers of 'NK' engines, as well as the Vikhr-30M, P-020 and P-065 small piston engines. Its biggest programmes are the aero-derived NK-36ST and NK-37ST industrial gas-turbine engines. In 1997, in a move towards rationalisation, this production plant became part of the NK conglomerate.

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AERO-ENGINES - TURBOFAN, RUSSIA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

KMPO - KAZAN MOTOR-BUILDING PRODUCTION

AL-35

The **AL-35F** is essentially the same engine as the AL-31FM, but fitted with a variable vectoring nozzle. This nozzle has the same variable area and profile as before, but can be pivoted hydraulically over an angular range of ±15º up or down (that is, about a horizontal axis). The AL-35F was bench-tested in 1987, and has been on flight test in the Su-27LL-PS since 1989. The first flight, on 21 March, took place with the nozzles locked. From 31 March 1989 flight testing explored vectoring, beginning with movement to +5º in a shallow dive. These engines power the Su-35 (LMK-2405), with a main control column and an additional sidestick controller.

The engine specification is similar to that for the AL-31FM. Since 1995 this work has been superseded by the AL-37FU, which in turn has led to the production AL-31FP.

This experimental two-dimensional nozzle was flown on the Su-27LL-PS in 1989 (1996)

The Su-35 was originally fitted with twin AL-35F engines, here at -15º nozzle angle (1996)

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

KLIMOV CORPORATION - ST PETERSBURG NPO IM KLIMOV

PK206

Licensed variants of the PW200 family are specified for: the Kamov Ka-115, one PK206D; and Kazan/Mil Ansat, two PK206C.

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

KLIMOV CORPORATION - ST PETERSBURG NPO IM KLIMOV

PK100

The first of this family is the PK127, for export versions of An-140 and Il-114. T-O rating is 1,864 kW (2,500 shp). A more powerful (turboshaft) version will be an option on the Mi-38 helicopter and its proposed Euromil export version.

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AERO-ENGINES - TURBOPROP, RUSSIA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

KLIMOV CORPORATION - ST PETERSBURG NPO IM KLIMOV

PK6A

Licensed variants of the P&WC PT6A turboprop cover power ranges from 373 kW (500 shp) to 1,235 kW (1,657 shp). Among announced applications are: Beriev Be-32, two PK6A-65B; AeroRIC Dingo, one PK6A-65B; ROKS-Aero T-101E Grach, one PK6A-65AR (or imported PT6A-65AR); ROKS-Aero T-201 Aist, one PK6A-67; ROKS-Aero T-203 Pchela, one PK6A-67; ROKS-Aero T-204 Griffon, two PK6A-34C; ROKS-Aero T-205 Karavan, one PK6A-67; ROKS-Aero T-610, one PK6A-114; Myasishchev M-101T, one PK6A-114; Myasishchev M-112, two PK6A-65B; Myasishchev/NAL M-102, two pusher PK6A-66; Khrunichev T-201, one PK6A-65AR (or imported PT6A-65AR); Tupolev C-Prop, two pusher PK6A-67; Khrunichev T-440 Merkury, two PK6A-135 (or imported PT6A-135); and PZL-130TC, one PK6A-62 (or imported PT6A-25C).

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AERO-ENGINES - MANUFACTURER, RUSSIA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

V/O AVIAEXPORT plc

MANUFACTURER DETAILS

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Fax: +7 095 207 0780
Tx: 411257 AVEX RU
Email: aviaex.msk.ru

GENERAL DIRECTOR: Feliks N. Myasnikov

This vast organisation handles every aspect of international trade in all forms of aerospace products. It operates through 15 subsidiary companies, two of which are:

DVIGATEL

(pronounced `dvigger-tel', means `engine')

Tel: +7 095 417 0435
Fax: +7 095 417 0144

DIRECTOR: Sergei Alyaev

Export/import of all forms of aerospace engines.

PR AND ADVERTISING

Tel: +7 095 207 5500
Fax: +7 095 207 2977

CHIEF: Alexei Didenko

Establishment of business contacts, media relations, conferences and seminars.

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AERO-ENGINES, RUSSIA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

Introduction

Since 1989 the entire industry of the former Soviet Union has been undergoing great changes. Of course, the most profound changes have reflected the break-up of the Soviet Union into separate republics. In this binder, engine centres are now listed under the name of the republic in which they are situated. Most are in this section, but Progress will be found under Ukraine. In addition, the entire management of the industry has been revolutionised. The changes are generally intended to increase commercial viability, and they are superimposed on changes of name caused by the replacement of deceased or retired chief constructors by their successors. For nearly 70 years the procedure was for new engines to be designed at a KB (construction bureau) always identified by the name of the bureau head, or chief constructor. Their work had the benefit of support from the laboratories of the Central Institute of Aviation Motors. Once cleared for production, an engine would be assigned to a factory for series production. The factory (or factories) remained anonymous, though its GAZ (state aviation factory) number might become known (for example see introduction to Klimov). Today the production plants are being named, many even have a chief designer, and they are taking display stands at exhibitions alongside the KBs. Meanwhile, confusion is increased further by the fact that some of the KBs have received names in honour of former chief designers or based on the city where they are located. As in the case of aircraft, each bureau no longer has guaranteed large home orders but has to fend for itself. This is reflected in the pages that follow. One immediate result is the formation of **ASSAD** in March 1992.

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AERO-ENGINES - TURBOPROP, RUSSIA

Jane's Aero-Engines 04

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

TVD-450

This turboprop may use the same core as the GTE-400. Rated at 336 kW (450 shp) it has been selected for the Myasishchev M-500 agricultural aircraft, driving an MTV-9 three-blade propeller. A twinned version, driving through a coupling gearbox to a single propeller, powers the Sukhoi S-86, as an alternative to the Lyul'ka Saturn AL-34. Each power section is rated at 336 kW (450 shp).

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AERO-ENGINES - TURBOSHAFT, RUSSIA

Jane's Aero-Engines 04

AMNTK (AIRCRAFT ENGINE SCIENTIFIC/TECHNICAL COMPLEX) `SOYUZ'

GTE-400

This new turboshaft, rated at 298 kW (400 shp), has been selected for the twin-engined version of the Ka-118.

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AERO-ENGINES - TURBOSHAFT/TURBOPROP, RUSSIA

Jane's Aero-Engines 03

OMSK ENGINE DESIGN BUREAU - OEDB

TVD-10B

Military designation: GTD-3
This engine was originally designed by V A Glushenkov in 1961-63 to power the Ka-25 helicopter. It has since been made in turboshaft and turboprop versions.

GTD-3F

T-O (5 min) rating 671 kW (900 shp). Turboshaft engine for most versions of Ka-25.

GTD-3BM

Uprated version retrofitted to many Ka-25 helicopters. T-O rating 738 kW (990 shp).

PZL-10W

Polish designation of turboshaft version, now made only in that country by PZL-Rzeszów (which see for description).

TVD-10B

Turboprop version, designed for An-28. T-O rating 754 kW (1,011 shp). The engine is dominated by the propeller drive passing across the top of the gas generator from the rear output gearbox to the propeller gearbox at the front.

TWD-10B

Polish designation for TVD-10B, which is made only by PZL-Rzeszów (which see for full description).

TVD-10BA

Uprated version being developed by OEDB at Omsk for An-28A Polar service. T-O rating 790 kW (1,060 shp).

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AERO-ENGINES - TURBOFAN, RUSSIA

Jane's Aero-Engines 03

LYUL'KA SATURN INC - LYUL'KA SATURN

AL-32

Marking a departure from supersonic fighter engines into *konversiya* products, this much smaller engine is being developed in turboshaft and turboprop forms for general aviation. The only announced application is as a turboshaft for the Mil Mi-54 twin-engined helicopter.

The AL-32 forms the power sections of the AL-34, described next. No attempt has been announced to match the AL-32 with a regenerator, and the engine has been made as simple as possible.

PERFORMANCE RATING:

T-O, S/L	574 kW (770 shp)

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AERO-ENGINES - TURBOJET, RUSSIA

Jane's Aero-Engines 03

LYUL'KA SATURN INC - LYUL'KA SATURN

AL-21

Lyul'ka began the development of this improved turbojet in late 1966. Objectives included the achievement of substantially improved specific fuel consumption, compared with the AL-7, and of similar dry and afterburning thrusts in a smaller engine. Prototype engines ran in 1967, leading to production of several versions.

AL-21F

Initial production version, fitted to early versions of Su-17.

AL-21F-1S

Fitted with large bleed manifold for blown flaps. Fitted to T6-1.

AL-21F-3

New compressor giving greatly improved performance. Produced in large numbers by Saturn and

Granit. Powers Su-17M, Su-24 (two engines) and MiG-23B.
TYPE: Single-shaft turbojet with afterburner.
INTAKE: Light alloy fabrication, with eight radial struts, 6 o'clock strut housing drive to accessory gearbox. Complete bleed air anti-icing.
COMPRESSOR: Fourteen-stage axial. Inlet frame carries front bearing, with hot air de-icing of fixed inlet vanes. Variable stators downstream pivoted to casing and central bullet. Variable stators ahead of first five stages. Parallel HP section with independently scheduled variable stators ahead of stages 10, 11, 12, 13 and 14. Mass flow at max 8,316 rpm (F) 85.5 kg (195.1 lb)/s, (F-3) 104 kg (229 lb)/s. Pressure ratio (F) 12.7, (F-3) 14.6-14.75.
COMBUSTION CHAMBER: Can-annular type, with 12 flame tubes each with duplex downstream burner.
TURBINE: Three stages with air-cooled first-stage blades. TET (F) 990ºC, (F-3) 1,112ºC. Rotor assembly supported by rear bearing held in eight-strut rear frame and drives via cone coupled to rear of compressor shaft.
AFTERBURNER: Rapid combustion downstream of three injector/flameholder rings. Corrugated insulated liner with ceramic coating. Fully variable nozzle with 24 flaps driven by six rams. Full regime control giving smooth light-up and modulated power over whole range from flight idle to maximum afterburner.
DIMENSIONS:

Length:	
AL-21F	5,340 mm (210.2 in)
AL-21F-3	5,160 mm (203.9 in)
Diameter:	
Inside inlet	885 mm (34.8 in)
Overall	1,030 mm (40.55 in)

WEIGHT, DRY:

AL-21F	1,581 kg (3,485 lb)
AL-21F-3 (MiG-23B)	1,845 kg (4,067 lb)
AL-21F-3 (Su-24)	2,005 kg (4,420 lb)

PERFORMANCE RATINGS (S/L):

Max dry:	
AL-21F	62.5 kN (14,050 lb st)
AL-21F-3	76.5 kN (17,200 lb st)
Max afterburner:	
AL-21F	87.3 kN (19,630 lb st)
AL-21F-3	110.5 kN (24,800 lb st)

SPECIFIC FUEL CONSUMPTION (S/L):

Max afterburner:	
AL-21F	53.6 mg/Ns (1.90 lb/h/lb st)
AL-21F-3	52.5 mg/Ns (1.86 lb/h/lb st)

Cruise regime:	
AL-21F	20.37 mg/Ns (0.72 lb/h/lb st)
AL-21F-3	21.47 mg/Ns (0.76 lb/h/lb st)

Cutaway drawing of AL-21F-3

Longitudinal section through AL-21F-3 (1996)

AL-21F-3 (1996)

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AERO-ENGINES - TURBOJET, RUSSIA

Jane's Aero-Engines 03

LYUL'KA SATURN INC - LYUL'KA SATURN

AL-7

This turbojet was the first of A M Lyul'ka's engines to go into mass production. The compressor, based on the Central Institute of Aviation Motors TR-11, was innovative in that at maximum rpm the air flow through the outer part of the first two rotor stages was transonic. Preproduction engines passed 100-hour tests in 1955, and the first series engines came off the production line in 1959. Small numbers are still active. There were four production versions:

AL-7F-1

In production 1959-70. Powers Su-7B, BM, BMK, BKL, UM and UMK, and formerly Tu-128A and UT.

AL-7F-2

Numerous detail improvements. In production 1960-70. Powers Su-9 and Su-11.

AL-7PB

Without afterburner; formerly powered Be-10 and Tu-110.

AL-7F-4

Uprated version with improved afterburner. Formerly powered Tu-128M.
TYPE: Single-shaft turbojet, with or without afterburner.
INTAKE: Fabricated aluminium assembly, with front bearing carried by 12 slim radial struts. All versions fully anti-iced by bleed air.
COMPRESSOR: Nine stages, with one row of variable stators. Drum-type construction with blades held in dovetail slots. Mass flow (F-1, PB) 114 kg (251 lb)/s, (F-2) 115 kg (253.5 lb)/s (F-4) 117.5 kg (259 lb)/s. Pressure ratio (F-1, PB) 9.1, (F-2) 9.3, (F-4) 9.38.
COMBUSTION CHAMBER: Annular type, with film-cooled inner flame tube of refractory alloy with front apertures generating rapid spiral vortices around the 18 fuel burners.
TURBINE: Two stage. Both rotors overhung behind rear bearing, with front disc bolted to tubular drive shaft. Unshrouded blades, air-cooled first stage in F-4. TET (F-1, F-2, PB) 927ºC, (F-4) 990ºC.
AFTERBURNER: (F versions): Upstream diffuser section leads to main combustion section with upstream pilot section with single spray ring and gutter flameholder, followed by larger main spray ring and flameholder. Ceramic-coated refractory liner. Variable con-di nozzle with 24 hinged flaps driven by four hydraulic rams.
ACCESSORIES (typical F versions): Main gearbox underneath engine driven via radial shaft from rear of compressor. Quickly replaced packages include fuel and oil pumps, hydromechanical control unit, aircraft hydraulic pump and alternator(s), pneumatic or electric starter and tachometer. Fuel/oil heat exchanger.
DIMENSIONS:

Length:	
F-1	6,630 mm (261 in)
F-2	6,650 mm (262 in)
F-4	6,810 mm (268 in)
PB	3,310 mm (130 in)
Diameter:	
F-1	1,250 mm (49.2 in)
F-2, F-4	1,300 mm (51.2 in)
PB	1,062 mm (41.8 in)

WEIGHT, DRY:

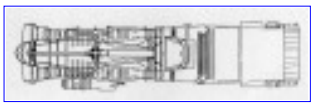
F-1	2,010 kg (4,431 lb)
F-2	2,103 kg (4,636 lb)
F-4	2,119 kg (4,672 lb)
PB	1,746 kg (3,849 lb)

PERFORMANCE RATINGS (Max T-O, S/L):

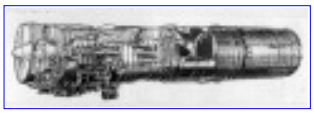
F-1	90.2 kN (20,280 lb st)
F-2	99.12 kN (22,282 lb st)
F-4	104.94 kN (23,590 lb st)
PB	71.2 kN (16,000 lb st)

SPECIFIC FUEL CONSUMPTION (S/L):

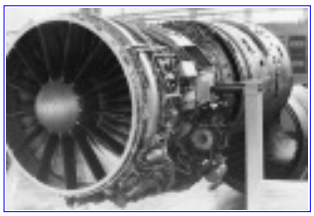
F-1, F-2, max T-O	56.4 mg/Ns (1.99 lb/h/lb st)
F-1, F-2, max continuous	25.49 mg/Ns (0.90 lb/h/lb st)
PB, max T-O	24.49 mg/Ns (0.865 lb/h/lb st)



Longitudinal section through AL-7F (1996)



Cutaway drawing of AL-7F-1 (1996)



AL-7F1 (Yefim Gordon) (1997)

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AERO-ENGINES - TURBOPROP, RUSSIA

Jane's Aero-Engines 03

ST PETERSBURG NPO IM KLIMOV - KLIMOV CORPORATION

TVA-3000

Described as a `fifth-generation engine', this is the most powerful shaft-drive gas turbine yet developed by the Klimov corporation. It will be made available with front or rear drive.

TVA-3000/120

Turboshaft version selected for twin-engined Ka-40. Design ratings: T-O 1.838 kW (2,465 shp); OEI contingency (30 s) 2,800 kW (3,755 shp).

TVA-3000/127

Turboshaft version, under development to power twin-engined Mi-38. Design ratings: T-O 1.838 kW (2,465 shp); OEI contingency 2,610 kW (3,500 shp).

L-3000

Designation of projected Chinese version intended to succeed WJ5.

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AERO-ENGINES - TURBOSHAFT, RUSSIA

Jane's Aero-Engines 03

ST PETERSBURG NPO IM KLIMOV - KLIMOV CORPORATION

TV2-117

This turboshaft engine was designed in 1960-62 to power the Mi-8 helicopter. The design was deliberately conservative, no attempt being made to rival the small size and light weight of Western engines in the same power class. Perhaps as a result, over 16,000 TV2-117 engines have been delivered.

TV2-117A

Basic version, in production since 1965. The power installation of the Mi-8 comprises two TV2-117A engines coupled through a VR-8A gearbox. As is common with modern Soviet helicopters, the engines and gearbox are delivered and thereafter treated as a single unit. The complete package incorporates a control system (separate from the control system of each gas generator) which maintains desired rotor speed, synchronises the power of both engines, and increases the power of the remaining engine if the other should fail.

TV2-117AG

Updated version to prolong life and TBO, ratings unchanged. Fitted to current production Mi-8 versions from Ulan-Ude.

TV2-117TG

Qualified to operate on all normal gas-turbine fuels, and on gasoline (petrol), benzine, diesel oil, liquefied natural gas, propane or butane gas. Flown on Mi-8TG, ratings unchanged, and selected as interim engine for Mi-38. A foreign production facility is sought.

The following description applies generally to all versions:

TYPE: Free turbine helicopter turboshaft.

INTAKE: Direct pitot, with main front casing providing vertical upper and lower driveshafts to accessory packages. Main accessory group above the engine projects ahead of intake face. Casing incorporates variable incidence inlet vanes.

COMPRESSOR: Ten-stage axial. Construction principally in titanium to reduce weight in comparison with the steel that would otherwise be used. Inlet guide vanes and stators of stages 1, 2 and 3 are of variable incidence to facilitate starting and increase compressor efficiency over a wide speed range; for the same reasons the casing incorporates automatic blow-off valves. Mass flow 8.4 kg (18.5 lb)/s. Pressure ratio 6.6 at 21,200 rpm.

COMBUSTION CHAMBER: Annular, with eight burner cones. Fabricated from inner and outer diffuser casings, flame tube, casing, burners, and anti-icing bleed air pipe.

COMPRESSOR TURBINE: Two-stage bolted to rear of splined shaft with front extension to drive accessories. Solid rotor blades, held by fir-tree roots in discs cooled by bleed air (first disc 10th stage air, all other discs 8th stage). First- and second-stage stators have 51 and 47 inserted blades respectively.

POWER TURBINE: Free power turbine of similar two-stage design; its rotors have 43 and 37 blades respectively.

JETPIPE: Large fixed-area duct which deflects the gas out at 60º. It comprises a pipe, pipe shroud and tie band, shroud connector links and exhaust pipe attachments. The exhaust pipe and shroud together form a double-wall assembly which minimises heat transfer into the power plant nacelle, the pipe being cooled by air circulating in the double wall.

OUTPUT: The main driveshaft is an extension of the power turbine rotor shaft. It conveys torque from the free turbine to the overrunning clutch of the helicopter main gearbox (VR-8A) and is also coupled to the speed governor of the free turbine rotor. Max output speed 12,000 rpm; main rotor speed 192 rpm.

ACCESSORIES: Mounted on the main drive box, above the intake casing, in which a train of bevel and spur gears provides drives for airframe and engine accessories. The engine automatic control system includes a fuel system, hydraulic system, anti-icing system, gas temperature restriction system, engine electric supply and starting system, and monitoring instruments. The hydraulic system positions the variable stators according to a preset programme, depending on compressor speed and air temperature at the inlet; it also sends electrical signals to control the starter/generator system, close the starting bleed air valves and restrict peak gas temperature to 600ºC. Air up to 1.8 per cent of the total mass flow can be used to heat the intake and other parts liable to icing. Fire extinguishant can be released manually by the pilot, upon receipt of a fire warning, through a series of spray rings and pipes.

STARTING: Electrical, fuel, and ignition systems are integrated. The SP3-15 system comprises DC starter/generator, six storage batteries, control panel, ground supply receptacle, and control switches and relays; of these all are airframe mounted except for the GS-18TP starter/generator which cranks the compressor during the starting cycle. The ignition unit comprises a control box, two semiconductor plugs, solenoid valve, and switch. The starting fuel system comprises an automatic starting unit on the NR-40V fuel regulating pump, constant-pressure valve, and two igniters.

FUEL SPECIFICATION: T-1 or TS-1 to GOST 10227-62 specification (Western equivalents, DERD.2494, MIL-F-5616).
OIL SYSTEM: Pressure circulation type. Oil is supplied by the upper pump and scavenged from the five main bearings by the lower pump, returned through the helicopter-mounted air/oil heat exchanger and thence to the helicopter tank. The oil seals and air/oil labyrinth seals are connected to a centrifugal breathing system.
OIL SPECIFICATION: Synthetic, permitting operation at oil temperatures above 200ºC, combined with easy starting at -40ºC without heating the oil. Grade B-3V to MRTU 38-1-157-65 (nearest foreign substitute Castrol 98 to DERD.2487). Consumption, not over 0.5 litre per hour per engine.
DIMENSIONS:

TV2-117A:	
Length	2,835 mm (111.5 in)
Width	547 mm (21.5 in)
Height	745 mm (29.25 in)
TV2-117AG:	
Length	2,842 mm (111.9 in)
Width	550 mm (21.65 in)
Height	748 mm (29.4 in)

WEIGHT, DRY:

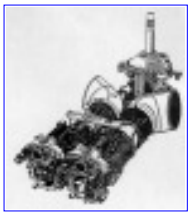
TV2-117A	330 kg (728 lb)
TV2-117AG	338 kg (745 lb)
VR-8A gearbox, less entrapped oil	745 kg (1,642 lb)

PERFORMANCE RATINGS:

Max	1,250 kW (1,677 shp)
T-O (S/L, static)	1,118 kW (1,500 shp)
Max continuous	895 kW (1,200 shp)
Cruise (122 kts; 225 km/h; 140 mph at 500 m; 1,640 ft)	746 kW (1,000 shp)

SPECIFIC FUEL CONSUMPTION:

T-O, as above	102.4 µg/J (0.606 lb/h/shp)
Cruise, as above	115.4 µg/J (0.683 lb/h/shp)



Russian drawing of Mi-8 installation of two TV2-117A engines and VR-8A gearbox (1996)

TV2-117A (1996)

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AERO-ENGINES - TURBOFAN, SLOVAKIA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

POVAZSKÉ STROJÁRNE LETECKÉ MOTORY, a.s.

DV-2

This engine was required to power a new trainer, the L-39MS/L-59, designed by the Czechoslovak airframer Aero Vodochody to replace the L-39. Under an October 1980 agreement between the Czechoslovak and Soviet governments, the ZVL (predecessor of PSLM) and Lotarev bureau (predecessor of ZMKB Progress) were assigned responsibility for the development of the baseline engine, and the ZVL for prototype, pre-production and serial engine production. All the DV-2 development costs were paid to the former Soviet Union under this agreement, and the ZVL acquired exclusive design rights for the engine. After state certification of the engine in 1990, the ZVL took over responsibility for independent development of the baseline engine (to improve TBO and engine life) and for a range of derivative engines sharing the same core.

DV-2

Baseline engine, originally designed to power the Aero L-59, and sized to match the AI-25TL as a `bolt-on' replacement, so that L-39s can be re-engined. Developed in 1982-87 with 25 prototype engines tested at ZMKB in Ukraine and at Povazska Bystrica. Total of 124 delivered for Czech and Slovak L-39MS and Egyptian and Tunisian L-59. In 1999 a new standard was certificated upgrading TBO to 500 h and service life to more than 2,000 h. Modification of the control system (see under that heading) was covered by final certification in October 2000. The improved FADEC can be retrofitted.

DV-2S/OX

Three engines modified for Yak-130D demonstrator, first flown 25 April 1996.

DV-2A

Baseline improved engine, with FADEC control, fluent control of HPC IGV and of slide valve controlling air bleed from behind the LPC, offering greater power at TIT decreased by 55ºC and rotors speed by 2 per cent, available for L-59, optionally for other trainer/light attack. Being developed according to MIL-STD with considerably extended engine life.

DV-2A.2

Uprated version with unchanged rotor speeds and TIT range, available for L-59 and L-159.

DV-2S, DV-2S.2

Versions of DV-2A, DV-2A.2, size and weight reduced, installation features and interfaces intended for Yak/AEM-130 (later Aermacchi AY-130) aircraft.

DV-2C, DV-2C.2

New titanium fan and fully variable LPC bleed into bypass.

DV-2F

Afterburning engine for supersonic applications.

DV-2.40

Projected turbofan for local-service and executive aircraft.

Except where otherwise indicated, the following refers to the DV-2C:

Type

Two-shaft turbofan.

Fan

Single stage, overhung, supersonic, blisk type, made of aluminium alloy with wide-chord blades without snubbers, spinner heated by bleed air. Hub/tip ratio 0.37, bypass ratio 1.46; DV-2, 1.34; DV-2A.2/S.2/C.2, 1.4; DV-2.40, 5.31. Diameter (except DV-2.40) 645 mm (25.4 in). Mass flow 49.7 kg (109.57 lb)/s; DV-2, 49.2 kg (108.47 lb)/s; DV-2A.2/S.2/C.2, 52.5 kg (115.75 lb)/s; DV-2.40, 134 kg (295.4 lb)/s. Pressure ratio 2.01; DV-2, 2.01; DV-2A.2/S.2/C.2, 2.15.

LP Compressor

Two core booster stages, rotating with fan. Fluent slide valve to control air bleed from LPC outlet.

HP Compressor

Seven stages. Variable IGV, cantilevered stator stages 3 to 6. Three fully variable bypass valves. Overall pressure ratio 14.0; DV-2, 13.5; DV-2A.2/S.2/C.2, 15.5.

Combustion Chamber

Annular, with 16 airspray nozzles, giving smokeless combustion with low emissions. Two torch igniters.

HP Turbine

Single stage, with fully aircooled NGVs and rotor blades. TET, 1,127ºC; DV-2, 1,170ºC; DV-2A.2/S.2/C.2, 1,115ºC.

LP Turbine

Two-stage, first-stage NGVs aircooled.

Starting

Air turbine. Manufactured by PBS Velká Bites (Czech Republic).

Control System

Dual-channel FADEC, with hydromechanical backup with redundant MIL-1553B bus for communication with aircraft. Developed by ELECMA (France) with Jihostroj-Velesin (Czech Republic) and EHA (Russia). DV-2, main electronic-hydromechanical, with digital block, plus reserve and emergency hydromechanical systems.

Lubrication System

Autonomous, recirculating, fully aerobatic.

Oil Specification

IPM-10, VNII-50 or Aero Shell 555.

Dimensions

DV-2	
Length (with exhaust system)	3,773 mm (148.54 in)
Width	994 mm (39.13 in)
Height	1,037 mm (40.82 in)
DV-2A, -2A.2	
Length (with exhaust system)	3,821 mm (150.43 in)
Width	994 mm (39.13 in)
Height	1,044 mm (41.16 in)
DV-2S, -2S.2	
Length (with exhaust nozzle)	2,245 mm (88.39 in)
Width	823 mm (32.40 in)
Height	1,046 mm (41.18 in)

DV-2C, -2C.2	
Length (with exhaust nozzle)	2,245 mm (88.39 in)
Width	823 mm (32.40 in)
Height	1,042 mm (41.02 in)
DV-2F	
Length	2,900 mm (114.17 in)
Width	823 mm (32.40 mm)
Height	1,118 mm (44.01 in)
DV-2.40	
Length (with exhaust nozzle)	2,429 mm (95.63 in)
Width	1,106 mm (43.54 in)
Height	1,606 mm (63.23 in)

Weight, Dry

DV-2	474 kg (1,045 lb)
DV-2A, -2A.2	435 kg (959 lb)
DV-2S, -2S.2, -2C, -2C.2	430 kg (948 lb)
DV-2F	About 630 kg (1,389 lb)
DV-2.40	613.5 kg (1,352.5 lb)

Performance Rating

(S/L, T-O)

Flat-rated to 24ºC:	
DV-2	21.58 kN (4,852 lb st)
Flat-rated to 26ºC	
DV-2A, -2S, -2C	21.58 kN (4,852 lb st)
DV-2A.2, -2S.2, -2C.2	24.5 kN (5,507 lb st)
DV-2F	42.0 kN (9,442 lb st)
Flat-rated to 25ºC	
DV2.40	35.1 kN (7,890 lb st)

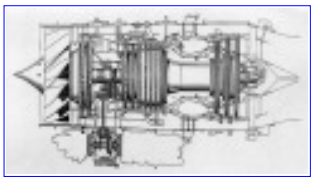
Specific Fuel Consumption

(S/L, T-O)

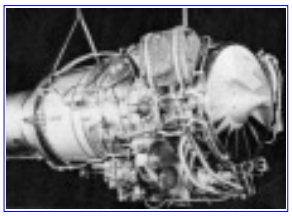
DV-2	17.015 mg/Ns (0.6012 lb/h/lb st)
DV-2A, DV-2S, DV-2C	16.931 mg/Ns (0.5982 lb/h/lb st)
DV-2A.2, DV-2S.2, DV-2C.2	17.485 mg/Ns (0.6178 lb/h/lb st)
DV-2F (augmented)	51.341 mg/Ns (1.814 lb/h/lb st)
DV-2.40	10.964 mg/Ns (0.3874 lb/h/lb st)

UPDATED

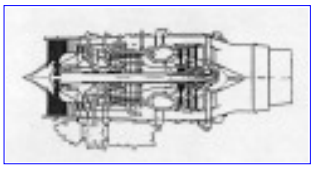
A DV-2 exhibited by Slovakia's PS



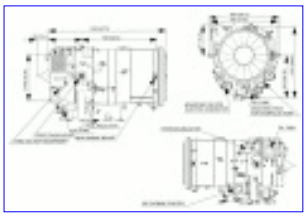
Section/cutaway of DV-2



Production DV-2A



Longitudinal section through DV-2 (DV-2A, DV-2A.2 similar)



DV-2
(2000)

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AERO-ENGINES - MANUFACTURER, SLOVAKIA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

PSLM - POVAZSKÉ STROJÁRNE LETECKÉ MOTORY, a.s.

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PSLM a.s. succeeded PS Aero Engine Division, a daughter company of Povazské Strojárne a.s. Povazská Bystrica (former ZVL). Under a ten-year agreement signed in October 1980 between the former Soviet and Czech governments, it partnered the Progress/Lotarev (ZMKB) bureau at Zaporozhye in the former USSR (Ukraine) in designing the DV-2 turbofan. This was required to power the Aero Vodochody L-39MS and L-59 trainer and light attack aircraft.

The DV-2 first flew in the X-22, a re-engined L-39, on 30 September 1986. In 1990 Yakovlev began design of an advanced trainer to meet a Russian requirement for 200 to replace the L-39, and at an early stage decided to use the DV-2 rather than the AI-25TL. This opened up a potentially large market for a version designated DV-2S to power the twin-engined Yak-130, which first flew on 25 April 1996. In collaboration with Progress ZMKB and Klimov, plans were made for possible production of the DV-2S in Russia for Yak-130 aircraft under the designation Klimov RD-35 (which see).

Following partition of Czechoslovakia PSLM became a Slovak company. Progress ZMKB went ahead on its own using the DV-2S gas generator as the basis of the AI-22 (which see). All development costs and intellectual properties reimbursment were paid to the former USSR according to the 1980 agreement and its attachments. PSLM sought partners to underpin the proposed DV-2S for the Italian derivative of the Yak-130, the Aermacchi AY-130. Rolls-Royce said ``We will be exclusive partners with PSLM, a regular contract should be signed in early 2000", but the agreement expired on 31 December 1999, the British company concluding ``Following a detailed technical evaluation, it has become apparent that the costs involved... do not present a viable case." P&W Canada came to the same conclusion.

Failure to reach a bilateral agreement was a severe blow to heavily indebted PSLM, which has not delivered an engine since the shipment of the last DV-2 for a Tunisian L-59T in 1996. It was also a blow to Aermacchi, which had just relaunched the Yak-130 as the AY-130. According to *Jane's* Prague correspondent, the Italian company is insisting that the Slovak Government ``guarantees more than Kcs 1 billion (US$24 million) to complete DV-2S development before allowing PSLM's continued participation in the AY-130 programme". In February 2000, Vojtech Panik, Slovak Director of Manufacturing Industry, said ``We will do everything possible to ensure that PSLM completes work on the DV-2S". Moreover, according to Fyodor Muravchenko of ZMKB Progress, if the Russian Air Force does select the AY-130 as its preferred advanced trainer, the engines would almost certainly be the AI-22, though he also insists that PSLM will participate.

By 2002 PSLM was determined to launch improved versions, described separately. One, the afterburning DV-2F, is the subject of discussions with the aircraft industry of China.

UPDATED

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AERO-ENGINES - MANUFACTURER, SOUTH AFRICA

Jane's Aero-Engines 03

ATLAS AVIATION - ATLAS

MANUFACTURER DETAILS

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Tx: 724403

CEO: J J Eksteen

Atlas is manufacturing the Rolls-Royce Viper 540 turbojet under sub-licence from Piaggio of Italy, for use in Atlas Impala attack trainers.

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AERO-ENGINES - MANUFACTURER, SPAIN

Date Posted: 30 August 2000

Jane's Aero-Engines 08

ITP - INDUSTRIA DE TURBO PROPULSORES SA

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TECHNICAL DIRECTOR: Manuel Rodriguez
COMMERCIAL OFFICE: Travesifa Costa Brava 6 (Carretera Colmenar Viejo), E-28034 Madrid
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DIRECTOR, CIVIL AND MAINTENANCE: Jorge Lluch

ITP was formed in 1989 as Spain's aero-engine company to participate in the design, development, manufacture, sale and support of gas-turbine engines. Shareholders are IBV Corporation (4 per cent), Rolls-Royce plc (45 per cent) and Turbo 2000 SA (51 per cent), the latter being in turn owned by Bazan (Spanish government) and Sener (private sector). The Ajalvir plant near Madrid, acquired from CASA, overhauls engines and accessories (GE, PWC, AlliedSignal, Turbomeca, SNECMA and Rolls-Royce/Allison). The newly constructed Zamudio plant near Bilbao manufactures components for civil and military engines. Deliveries began December 1991. ITP is the Spanish participant (13 per cent) in the EJ200 engine programme to power the Eurofighter 2000, and is a partner in the projected M138 turboprop (see Airbus in International section). It also participates in the Trent, BR710 and 715 and Honeywell AS907, and produces components for the RB211-524, RR535 and IAE V2500.

Over 40 per cent of its workforce is highly qualified. Sales in US$ in 1998 comprised: civil, 80

million; military 57.16 million; maintenance 41.67 million. Over 80 per cent of the turnover was for non-Spanish customers. ITP has major non-aero business. The following are current programmes in the aero sector:

Eurojet EJ200

ITP has a 14 per cent share, and is responsible for the bypass duct, front jetpipe, turbine exit case, externals and complete nozzle, and is developing a thrust vectoring nozzle (see Eurojet in International section).

General Electric

ITP repairs and overhauls the CF700, CT7, F404 and J79. ITP also shares in the F414, T700 and GE participation in the LHTEC T800.

M138

ITP is a member of the consortium developing this turboprop (which see, under Airbus Military). It is responsible for the inlet casing and dressing, intermediate casing and turbine exhaust casing.

Honeywell AS907

ITP is responsible for design and manufacture of LP turbine static components. ITP repairs and overhauls the T53, T55, TFE731 and TPE331.

Rolls-Royce RR BR 715: Share 4 per cent, external dressings and exhaust system.

Rolls-Royce Trent

ITP has a 2.5 per cent share in the Trent 700 and 800, with responsibility for the IPC case Stages 1-2 and 3-8, and the 05 and 08 IP turbine modules. ITP had a 3 per cent share in the Trent 500, with responsibility for the IPC case Stages 1-2 and Stages 3-8 and the tail bearing housing. By 2000, ITP had added responsibility for IPC disks 1 to 4 and for stages 1 to 4 of the LP turbine. This increases the company's share in the Trent 500 to 10.1 per cent.

SNECMA Atar Plus

ITP has responsibility for OGVs (outlet guide vanes).

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AERO-ENGINES - TURBOFAN, SWEDEN

Date Posted: 22 March 2002

Jane's Aero-Engines 02

VOLVO AERO CORPORATION

RM8

The RM8 is a Swedish military version of the Pratt & Whitney JT8D turbofan which Flygmotor developed to power the Saab 37 Viggen supersonic combat aircraft. It was produced in two versions:

RM8A

Initial production version to power the AJ37, SF37, SH37 and SK37. The first production engine was shipped on 28 October 1970.

RM8B

This version was developed to power the JA37 fighter Viggen. Research showed that a changed design could improve the stability of operation at high altitudes and in severe manoeuvres, as well as increase thrust in all regimes. In collaboration with Pratt & Whitney the design of the RM8B was completed in late 1971. The major change to improve functional stability at high altitude involved replacing the first stage of the LP compressor by a third stage on the fan. To increase thrust the RM8B has a four-nozzle burner combustion system and a new HP turbine. Delivery of production engines began in 1978 and was completed on 19 September 1988.

Type

Two-spool bypass jet (turbofan of low bypass ratio) with modulated afterburner.

Intake

Annular, with 19 fixed inlet guide vanes.

Fan

Front fan with solid titanium blades. RM8A two stages, RM8B three stages.

LP Compressor

RM8A four stages, RM8B three stages, rotating with fan. Blades of titanium. Steel casing.

HP Compressor

Seven stages. Blades of high temperature alloys. Overall pressure ratio (RM8A) 16.5. Bypass ratio approximately 1. Mass flow 145 kg (320 lb)/s.

Combustion Chamber

Can-annular type with nine cylindrical flame tubes, each downstream of a quadruple Duplex fuel nozzle. Two high-energy spark plugs, each with its own igniter box.

HP Turbine

Single stage with cast air-cooled blades.

LP Turbine

Three-stage axial flow, with cast blades. Exit guide vanes after turbine.

Afterburner

Outer skin titanium. Inner skin special alloys. One hot streak igniter. Hydraulically actuated fully variable nozzle, using fuel as the operating fluid.

Reverser

The Viggen rear fuselage incorporates three large inlets admitting air to form an ejector surrounding the engine afterburner nozzle. At subsonic speeds these admit air to reduce fuselage base drag, but at supersonic speeds they are closed to convert the nozzle into an efficient con-di profile. On landing, compression of the main landing gears automatically activates reverser blocker doors in the afterburner nozzle, all the engine efflux thereafter being ejected diagonally forward through the ejector apertures.

Mounting

Three point. Main mountings on each side of compressor casing; one under turbine casing.

Accessories

Via gearbox, under engine, driven from HP turbine shaft.

Control System

The main system for the gas generator comprises a Bendix hydromechanical control. A further Bendix unit controls the afterburner and nozzle. A single power lever controls thrust from maximum afterburner

down to idle; movement below idle actuates the fuel cut-off valve.

Dimensions

Length overall: RM8A	6,197 mm (244 in)
RM8B	6,279 mm (247.2 in)
Max diameter (both versions)	1,397 mm (55.0 in)
Inlet diameter (both)	1,030 mm (40.55 in)

Weight, Dry

RM8A	2,100 kg (4,630 lb)
RM8B	2,250 kg (4,960 lb)

Performance Ratings

(ISA, S/L)

Max T-O, augmented	
RM8A	115.6 kN (25,990 lb st)
RM8B	125 kN (28,110 lb st)
Max T-O, dry:	
RM8A	65.6 kN (14,750 lb st)
RM8B	71.1 kN (16,200 lb st)

Specific Fuel Consumption

Max augmented:	
RM8A	70.0 mg/Ns (2.47 lb/h/lb st)
RM8B	71.4 mg/Ns (2.52 lb/h/lb st)
Max dry:	
RM8A	17.8 mg/Ns (0.63 lb/h/lb st)
RM8B	18.1 mg/Ns (0.64 lb/h/lb st)
Max continuous (both)	17.3 mg/Ns (0.61 lb/h/lb st)

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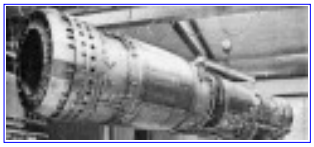
Cutaway drawing of RM8A



RM8B

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AERO-ENGINES - MANUFACTURER, SWEDEN

Date Posted: 22 March 2002

Jane's Aero-Engines 02

VOLVO - VOLVO AERO CORPORATION

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Volvo Aero produces aircraft engines and space propulsion components. It is the successor to SFA (Svenska Flygmotor AB) which had a proud history of piston and turbojet engines, including the RM6 and RM8.

Volvo Aero is a major partner in the production of commercial jet engines, including the following:

BR715

Design and manufacture of complete intermediate case.

CF6-80

LP spool 2-5, HP spool 3-9, compressor rear frame and diffuser case, LP turbine case.

CFM56

All versions, LP turbine case and vanes and LP turbine shaft.

GE F414

Fan disk, compressor case and mid-frame.

JT8D-200

Compressor hub and discs, drive shaft and intermediate case.

PW2000

Bearing/housing, compressor hub, intermediate case, HP rear frame, LP turbine case, turbine exhaust case.

PW4000

Compressor hub and discs, turbine exhaust case.

Tay

Flame Tubes.

Trent 500 and Trent 900

Intermediate case.

V2500

Bearing/housing/bearing support, LP turbine case, turbine exhaust case.

In addition to aircraft engines, Volvo Aero has many spaceflight contracts, notably on engine components for various Ariane versions, and is the world's largest supplier of nozzles and combustion chambers for launch-vehicle rocket engines. In Malmö, Volvo Aero Turbines develops and markets small gas turbines. Volvo Aero Engine Services is a commercial maintenance company.

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AERO-ENGINES - TURBOFAN, SWEDEN

Date Posted: 11 January 2002

Jane's Aero-Engines 11

VOLVO AERO CORPORATION

RM12

The RM12 is a version of the F404, developed jointly by General Electric and Volvo to power the JAS 39 Gripen. General Electric retains rights to the design, and its Lynn, Massachusetts, plant (home of all GE smaller engines) supplies approximately 60 per cent by value of parts, in the form of a self-contained production kit.

In return, Volvo Aero is a partner in all F404 applications. It supplies parts to General Electric similar to those parts that it manufactures for the RM12.

The RM12 thrust improvement has been achieved by increasing the turbine inlet temperature by up to 105ºC and by increasing fan air flow. The fan meets more stringent bird strike requirements, and this has required changes to the control system, with built-in redundancy to ensure get-home power. Increased temperature and pressure have required changes to hot section materials. RM12 testing started at GE in June 1984, and the Gripen first flew on 9 December 1988. Eight flight-rated prototypes and 120 production engines had been delivered by February 2000 from over 220 then on order.

By late 2001 the RM12 was well on the way to establishing a significant presence in the global marketplace. Contracts had been discussed with a number of air forces, among them Austria and South Africa. In December 1999 South Africa placed an order for nine two-seat Gripens, with an option on 19 single-seaters, for delivery from 2002. The Gripen has been studied by all former Warsaw Pact countries, and on 10 December 2001 the Czech Republic announced that it had selected 24 Gripens to replace its fleet of MiG-21 fighters. This contract is worth approximately US$70 million to GE.

RM12+

Studies are continuing into this improved engine, which would incorporate `third-generation' monocrystal turbine blades, a new radial flameholder in the afterburner and a new FADEC. This engine could be introduced late in Lot II aircraft, and is also being considered for a fleet retrofit.

The following are features of the RM12 which differ from the GE engine:

Fan

Variable first-stage stator. Air flow 68 kg (150 lb)/s. Bypass ratio 0.28.

Weight, Dry

	1,050 kg (2,315 lb)

Performance Ratings

Max T-O: dry	54 kN (12,140 lb st)
augmented	80.5 kN (18,100 lb st)

Specific Fuel Consumption

Max T-O, augmented	50.45 mg/Ns (1.78lb/h/lb st)

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Cutaway drawing of RM12

Volvo RM12

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AERO-ENGINES - TURBOFAN, SWEDEN

Date Posted: 17 August 2000

Jane's Aero-Engines 08

VOLVO AERO CORPORATION

RM5, RM6

The Rolls-Royce Avon turbojet was manufactured under licence in Sweden by Svenska Flygmotor, now Volvo Aero. It was produced in several versions, for Swedish military aircraft, small numbers of which are still in use.

RM5A2

This was essentially the Avon RA.7R, of the initial 100 series, fitted with an afterburner and nozzle designed and built by Flygmotor. The afterburner had three fuel spray rings with downstream injection, fed by an air-turbine centrifugal pump. Control was exercised by the Lucas full-range flow control system governing the main engine, together with the variable-area nozzle. The latter had two clamshell shutters positioned by four Dunlop pneumatic actuators and electronic valves. Starting was by a Plessey gas-turbine unit. Thrust ratings were 33.35 kN (7,500 lb st) dry and 46.07 kN (10,362 lb st) with afterburner. This engine powers surviving J32D Lansens.

RM6B

This engine was based on the Avon 200 series, and was also fitted with an improved Flygmotor Type 65 afterburner. This was fitted with multiple spray rings, each mounted at the front of a V-gutter flameholder. A small proportion of the fuel was injected into the flameholder, the major part being injected upstream. Ignition was by a hot-shot (hot streak) system, briefly fed from a liquid-injection

upstream of the turbine. Afterburner fuel was fed by a centrifugal pump driven by an axial turbine running on compressor bleed air controlled by main-engine fuel pressure. Afterburner fuel was regulated at the pump outlet by a sensor controlled to maintain a constant pressure ratio across the main-engine turbine. Again, the nozzle had two-position pneumatically driven clamshell shutters. This engine had ratings of 51.58 kN (11,600 lb st) dry and 67.64 kN (15,212 lb) with afterburner. It powers surviving J32E Lansens and several early J35 and SK35 Drakens.

RM6C

This engine was based on the Avon 300, or RB.146, and was also known as the Avon Mk 60. It was fitted with a further improved Flygmotor afterburner, with a greater jetpipe diameter to handle the increased mass flow and fuel burn and an improved ejector nozzle with hydraulic control. Thrust was infinitely variable throughout the afterburning range by controlling main-engine fuel flow. Thrust ratings are 56.87 kN (12,790 lb st) dry and 76.75 kN (17,262 lb st) with maximum afterburner. This engine powers surviving Drakens of J35D type and later, including a few still flying in Austria and Finland.

For further description, see Rolls-Royce Avon in UK section.

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AERO-ENGINES - TURBOFAN, TAIWAN

Date Posted: 18 September 2001

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AIDC - AEROSPACE INDUSTRIAL DEVELOPMENT CORPORATION

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Tel: (+886 4) 256 22 26
Fax: (+886 4) 256 23 70
Web: http://www.aidc.com.tw

Director-General: General WenLi Lin

Originally called Aero Industry Development Centre, this organisation develops aircraft and engines. It was formed from a previous organisation, BAI, in March 1969, and in 1996 became a state-owned corporation, run by the Ministry of Economic Affairs. From the outset its work on engines has been closely linked with the US company Garrett (which became AlliedSignal and is now Honeywell) in several programmes. These included licence-production of 154 T53-13 turboshaft engines for AIDC-built Bell UH-1H helicopters, and almost as many TFE731-2-2L turbofans to power the 60 AIDC AT-3 twin-jet trainers. A considerably larger programme was to play a major role in both the design and production of the ITEC TFE 1042 for the AIDC Ching-Kuo supersonic IDF (Indigenous Defence Fighter). This engine is described under Honeywell, the partner in ITEC, in the US section.

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AERO-ENGINES - TURBOFAN, UKRAINE

Date Posted: 18 April 2002

Jane's Aero-Engines 12

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

AI-222

Having derived the AI-22 (which see) from the DV-2 (see under Slovakia), the Ivchenko Progress designers used this engine as the basis for a substantially different engine with the above designation. Major changes are found in the fan modules (dramatically reducing BPR), LP turbine (single stage), accessory gearbox and nozzle, and the compressor has an additional stage. Despite these changes, the Progress designers describe the AI-222 as ``using the debugged AI-22 gas generator''.

Complete engine testing of the AI-222 was scheduled to begin in 2000, for certification in 2003, but the plan in 2001 is for the first engine to run in that year, and for flight testing to take place ``a year later as part of the powerplant of the Yak-130'', suggesting that a single AI-222 will be installed in this twin-engined aircraft. Other target applications include the Karakorum K-8 and Aero L 159.

The following versions are planned:

AI-222-25

The baseline engine, for advanced trainers and light attack aircraft.

AI-222-28

An upgraded engine with higher thermodynamic-cycle characteristics, to be rated at 27.46 kN (6,173 lb st), and eventually at 29.42 kN (6,614 lb st).

AI-222-25F

The AI-222-25 fitted with an afterburner, increasing T-O thrust to 41.19 kN (9,259 lb st). This engine would be cleared to M1.5.

AI-222-25FK

A derivative with a short afterburner, not expected to increase overall engine length by more than 500 mm (19.69 in). This would increase T-O thrust to 29.42 kN (6,614 lb st) and is targeted at ``aircraft that need only one reheat mode in flight".

AI-222-28F

A future upgraded afterburning version with a T-O rating of up to 49.03 kN (11,023 lb st).

Except for the data, the following applies to the baseline AI-222-25:

Type

Two-shaft turbofan of low BPR (bypass turbojet).

Fan

Two stages, could be regarded as the LP compressor. Diameter 630 mm (24.80 in). Mass flow 50.3 kg (110.9 lb)/s. BPR 1.19.

Compressor

Eight stages, with variable guide vanes in first three stages. Disks and rotor blades fabricated as blisks in titanium alloy, assembled with end-to-end bolts. OPR 15.6.

Combustion Chamber

Annular, designed for low emissions. Film-coated flame tube and 16 airblast fuel nozzles.

HP Turbine

Single stage with convective cooling. TET 1,172ºC. DS cast nozzle guide vanes. Two HP shaft bearings.

LP Turbine

Single stage. Three LP shaft bearings.

Jetpipe

Combined convergent mixer. Future plans envisage a rotating nozzle vectoring ±20º driven by aircraft hydraulics.

Control System

Fully diagnostic FADEC with ``new type hydromechanical fuel system".

Fuel Grade

T-1 or Ts-1.

Mounting

On intermediate case with rear support on turbine bearing outer ring.

Accessory Gearbox

Variable type and position, depending on application.

Dimensions

Length: AI-222-25	1,960 mm (77.17 in)
AI-222-25F	3,010 mm (122.05 in)
AI-222-25KFK	2,460 mm (96.85 in)
Height	1,050 mm (41.34 in)

Weight, Dry

AI-222-25	440 kg (970 lb)
AI-222-25F	540 kg (1,190.5 lb)
AI-222-25KFK	520 kg (1,146.4 lb)

Performance Ratings

T-O, S/L: AI-222-25	24.51 kN (5,511 lb st)
AI-222-25F	41.18 kN (9,259 lb st)
AI-222-25KFK	30.10 kN (6,768 lb st)
Cruise: AI-222-25 (10,000 m, 32,808 ft, M0.6)	2.94 kN (661 lb)
Max: AI-222-25F (11,000 m, 36,089 ft, M1,4)	27.26 kN (6,129 lb)
AI-222-25KFK, S/L static, max dry	25.00 kN (5,622 lb st)

Specific Fuel Consumption

AI-222-25, T-O	18.14 mg/Ns (0.64 lb/h/lb st)
AI-222-25, cruise, as above	24.8 mg/Ns (0.875 lb/h/lb)
AI-222-25F, max T-O	51.87 mg/Ns (1.83 lb/h/lb st)
AI-222-25KFK, max T-O	26.36 mg/Ns (0.93 lb/h/lb)

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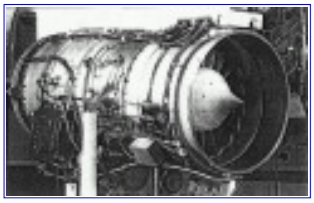
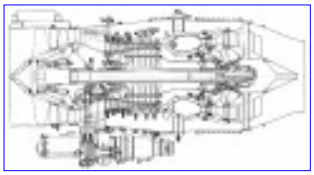
Mockup of AI-X22, also designated AI-222 (2000)

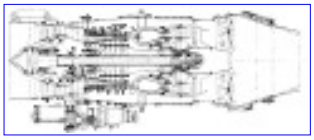


AI-222-25 (2001)



AI-222-25 longitudinal section (2001)

AI-222-25F longitudinal section (2002)



AI-222-25KFK longitudinal section (2002)

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AERO-ENGINES - TURBOSHAFT, UKRAINE

Date Posted: 22 March 2002

Jane's Aero-Engines 02

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

AI-450

This turboshaft engine is a completely new design, planned in the late 1990s to power helicopters of up to 3.5 tonnes (7,716 lb) MTO weight. The intended applications are initially the Kamov Ka-115, and a version of the Ka-226 designated Ka-228. According to Progress, plans for the Ka-115 to be powered by the Canadian PW206D ``have not been realised", and that in any case the AI-450 is ``20 to 30 per cent cheaper". An unofficial source stated that US$10 million would be needed to fly the prototype Ka-115, but that at least 1,000 of these helicopters would be needed, priced at ``about US$800,000", to replace the Ka-26 and Mi-2.

The engine comprises three modules: reduction gear and accessory gearbox, the gas generator, and the power turbine and driveshaft. It makes use of the most modern materials and coatings for all-weather operation in all environments. Emphasis has been placed on reliability and low cost. In 2001 components were being tested, the first complete-engine run being due in November that year.

Type

Free-turbine turboshaft.

Compressor

Single-stage centrifugal, delivering into radial diffuser. Mass flow 1.8 kg (3.97 lb)/s.

Combustion Chamber

Annular, with two 180º folds to reduce length.

Compressor Turbine

Single-stage axial with aircooled nozzle ring and inserted rotor blades.

Power Turbine

Single-stage axial, uncooled.

Output

Front or rear drive, primary gearbox ratio 6.5, integral torquemeter.

Dimensions

Length	950.1 mm (37.406 in)
Width	554 mm (21.81 in)
Height	592.3 mm (23.32 in)

Weight, Dry

	103 kg (227.07 lb)

Performance Ratings

(S/L)

Emergency	410 kW (550 shp)
T/O	346.75 kW (465 shp)
Max continuous	298 kW (400 shp)
Max cruise	223.7 kW (300 shp)

T/O	101.48 µg/J (0.573 lb/h/shp)
Max cruise	110.63 µg/J (0.655 lb/h/shp)

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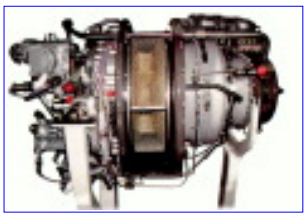

AI-450
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AERO-ENGINES - TURBOFAN, UKRAINE

Date Posted: 04 July 2001

Jane's Aero-Engines 10

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

AI-V

The above heading is merely shorthand for ``ZMKB Progress studies into tip-drive helicopter (*Vintrokryl*)". For many years this design bureau has been studying how best to drive the rotors of future large helicopters. Unlike Western countries, Russia and Ukraine can see that eventually helicopters much larger than the Mi-26 will be needed, and even that helicopter caused problems with the need to transfer power at the rate of 17,160 kW (23,000 shp) through gearteeth. A team at ZMKB Progress, led by S I Slobodkin, has studied tip-drive systems. This work has been partnered by studies at M I Mil helicopter company, led by leading designer Gennadi Lazarev.

The two design teams began working together on tip-drive in the mid-1950s. This led to the small V-7 helicopter of 1959, with a two-blade rotor carrying an AI-7 ramjet on each tip (with tip drive there is almost no torque imparted to the helicopter, so no tail rotor is needed). This work was little publicised at the time, and nothing at all was said about later work with MD-3 tip turbofans.

Current work is centred mainly on tip-drive turbofan or turbojet engines, started by compressed air piped along the blade spar (a byproduct of this is said to be immediate warning of air leak through any fatigue crack). Research has concentrated on shaft bearings, lubrication (including the design of oil radiators and use of hot lubricant for anti-icing) and the achievement of uniform fuel spray or vapour distribution in the combustion chamber.

Extensive research has been carried out into rotor dynamics and aerodynamics. With the helicopter parked, the tip-engine axis is preferably about -6º; at maximum lift this changes to about +7º. A

difference of 1 kg between the weight on each tip, which apparently could result from a difference in oil flow, results in an out-of-balance force on the hub of over 300 kg. The engine's presence has a useful endplate effect on rotor lifting power, and in autorotation the engine's drag is more than cancelled out by the greater kinetic energy of the rotating mass.

This work is directed towards future helicopters in the 100-ton class. In 1951 the Editor happened to be on an undergraduate course at Westland Aircraft. He was asked his opinion on a range of studies for large helicopters with turbojet engines mounted on the tips of the single free-running rotor. The largest of these would have had a 62.79-m (206-ft) rotor with three blades, each carrying an Armstrong Siddeley Sapphire turbojet, then of 36.9 kN (8,300 lb st). My comment was: ``Apart from the shaft bearings, I can't see a problem, perhaps you'll build these one day''. Westland's attitude was ``These schemes are ridiculous, we wish we hadn't published them''. Their attitude was coloured by the flat statement of the Sapphire designers at Parkside, Coventry: ``No turbojet will ever run successfully on a rotor-blade tip''. Half a century later the Russians and Ukrainians are doing just that.

NEW ENTRY

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AERO-ENGINES - TURBOFAN, UKRAINE

Date Posted: 04 July 2001

Jane's Aero-Engines 10

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

AI-22

This turbofan is based on an upgraded version of the gas generator used in the DV-2 (see under PSLM, Slovakia), which the forerunner company of Ivchenko Progress developed in the USSR. General designer Muravchenko says ``We have radically revamped the design of the gas generator for the AI-22, and in early 1999 were bench-testing major modules and parts. The complete engine is to go on test late this year''. He commented that the business link with what is now PSLM is being restored, and ``We are ready to co-operate in the design of the AI-22''.

AI-22

The basic engine, described below, is aimed at long-range business jets and small regional aircraft, notably the projected Tu-324 and Yak-48 (the Il-108, originally projected with the AI-22, is at present shelved, though it appeared in a list of possible applications in June 2000). The gas generator was first tested on 28 April 1999, followed by the first complete engine test on 26 September 2000.

In early 2001 it was announced that production of the AI-22 was to be undertaken in partnership with Motor Sich (see under Ukraine) and KMPO (see under Russia). From this engine has been derived the AI-222, described separately.

Type

Two-shaft turbofan.

Fan

Single stage. Diameter 830 mm (32.2 in). Mass flow 140 kg (309 lb)/s. BPR, approximately 5.

LP Compressor

Two-stage core booster, rotating with the fan.

HP Compressor

Seven stages. OPR 19.

Combustion Chamber

Annular, with 16 airblast nozzles.

HP Turbine

Single stage with convective cooling. TET 1,147ºC.

LP Turbine

Two stages.

Jetpipe

Combined mixer nozzle, cascade-type reverser.

Control System

(All) FADEC.

Fuel Grade

T-1 Or Ts-1.

Dimensions

Overall diameter	not stated
Length	2,400 mm (94.5 in)

Weight

Dry	765 kg (1,687 lb)

Performance Ratings

T-O (S/L)	37.95 kN (8,532 lb st)
Max OEI	41.19 kN (9,260 lb st)
Cruise (11,000 m, 36,090 ft, M0.8)	7.60 kN (1,708 lb)

Specific Fuel Consumption

T-O (S/L)	10.5 mg/Ns (0.37 lb/h/lb st)
Cruise	18.42 mg/Ns (0.650 lb/h/lb st)

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Computer image of AI-22 with reverser (1999)

Computer image of AI-22 with reverser (1999)

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AERO-ENGINES - MANUFACTURER, UKRAINE

Date Posted: 04 July 2001

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IVCHENKO PROGRESS ZMKB - IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

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This renowned establishment can trace its origin back to 1916, when the Decca (Deka) stockholding company established a design office at its aero-engine factory at Aleksandrovsk. After the Bolshevik revolution the city became Zaporozhye, and the factory became GAZ No 29. After a decade of near-chaos, Arkadii Sergeyevich Nazarov was appointed Chief Designer and told to get things in order. He was later replaced by Sergei K Tumanskiy (later famous at Soyuz) and Ye V Umin, but the office did not design its own engines until Aleksandr Georgiyevich Ivchenko arrived in May 1945. Then, very quickly, the AI series of piston engines appeared, some (AI-14 and AI-26) still being in production in modified forms today.

In 1953 Ivchenko was handed Kuznetsov's TV-2 turboshaft engine and told to do something with it (so was rival Aviadvigatel). Ivchenko developed it into the TV-2T turboprop and TV-2VK turboshaft, but the rival D-25V was chosen for the Mi-6 and Ivchenko chose to develop the TV-2T into the TV-20. First tested in 1956, this had a rival in Kuznetsov's NK-4. Both designers were called to the Kremlin, where Kuznetsov irritated the Soviet leaders by lecturing them, whereas Ivchenko just said ``We have the TV-20 and it works'', and sat down. It went into production as the AI-20.

Ivchenko died in 1968, and was succeeded by Vladimir Alekseyevich Lotarev, who concentrated on three-shaft turbofans. In turn, in 1988 he was succeeded by Muravchenko. Until 1991, Progress was a member of the group of enterprises of the USSR Aviation Industry Ministry. Now it is a state enterprise of Ukraine. Even though this is now a separate country from Russia, the latter's ASSAD states that it is providing half the development funds for Zaporozhye's most important new engine, the D-436. When fully developed this engine is to be produced in Russia, as explained in the D-436 entry.

Progress carries out the complete cycle of creation of gas-turbine engines, the cycle comprising the design, development, manufacture of the development batch and certification, as well as supervision of series production and operation. Series production of engines designed and developed by ZMKB Progress is carried out by the Motor Sich enterprise (formerly, Zaporozhye Industrial Association Motorostroitel) in Ukraine and by Salyut and UMPO in Russia. The company collaborates especially closely with Stupino-based Aerosila for propellers and propfans, OMKB for digital and hydromechanical control systems, Molniya of Ufa for FADEC and ignition systems, FED of Kharkov for hydraulic and fuel control systems, and SKTB of Odessa for electronic and pressure test systems.

Not including Ivchenko's AI-14 and AI-26 piston engines, over 20 types of Progress engines are in service on 42 types of aircraft. These engines are in service in 60 countries. The total of more than 30,000 Progress aircraft gas-turbine engines has accumulated over 300 million hours in service.

As the Lotarev bureau, the Zaparozhye design team led the design and development of the DV-2 turbofan, under a 1980 agreement with what became the Slovak company PSLM. From 1986, all development was concentrated in the Slovak company, through Mr Muravchenko has kept in close touch and has continued to search for Russian applications.

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AERO-ENGINES - MANUFACTURER, UKRAINE

Date Posted: 12 April 2001

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MOTOR SICH - jsc MOTOR SICH

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This enterprise, which celebrated its 80th anniversary in 1996, claims to be the world's largest producer of gas-turbine engines. In the field of aviation its engines power `53 types of aircraft in 96 countries', and embodies eight production factories located in Ukraine. These work closely with Ivchenko Progress, UMPO, ZMKB Progress and Klimov Corporation in order to manufacture a wide range of products, including the following complete aero-engines: Al-20, Al-25 and -25TL, D-18T, D-36, D-136, D-436T and TV3-117. In 2000 it was heavily engaged on the VK-2500 turboshaft (see under Klimov). It is tooling up to produce the D-436T1 and T2, and the D-18TM and TR turbofans, the TV3-117VMA-SB2 turboprop and also the newly developed AI-30 and VK-1500 turboprops and the AI-22 and 222 turbofans. It also produces mobile turbogenerator stations and consumer goods.

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AERO-ENGINES - TURBOFAN, UKRAINE

Date Posted: 12 April 2001

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IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

D-27

Since 1985 the engineering team at Zaporozhye has been striving to create a gas generator which can serve as the core in a range of turboshaft, turboprop and turbofan engines. This core became identified by the designation 27, and most of the effort has been applied to the D-27 propfan, using propeller blades of advanced design as noted below. It is designed for transport aircraft flying at speeds in the region of 800 km/h (500 mph). In such aircraft the D-27 could improve take-off and landing performance and offer reduced fuel burn and noise. Overall thermal efficiency is 37 per cent. In 1998 the D-27 Project Engineer was Vladimir Pechev.

The D-27 was derived from the D-36 via the D-136 and D-236, but with important differences. The propeller is the contrarotating Stupino SV-27, developed by the joint-stock company Aerosila, with a diameter of 4.5 m (14 ft 9 in). In comparison with the previous-generation SV-36 propeller, the SV-27 has blades made of advanced composites with significantly reduced thickness/chord ratio and pronounced scimitar-like curvature on the leading-edge, which is fitted with an electric anti-icing strip. It offers increased efficiency under high-speed cruise, and improved acoustics. There are six blades in the front propeller and eight in the rear, the latter absorbing most of the power and providing most of the thrust. Rotational speed: T-O 1,000 rpm, cruise 850 rpm. The D-27 meets ICAO (FAR) and AP-36 noise legislation.

The gas generator has been tested extensively over several years, and in 1992 the complete D-27 began bench testing. By this time the matchless combination of power, low fuel burn and low noise had been appreciated by designers of the following aircraft:

An-70

The D-27 was the basis for the design of the An-70 heavy airlifter in all its versions. The prototype An-70 began

flight testing on 16 December 1994, but unfortunately was destroyed after a mid-air collision on 10 February 1995. A second An-70 prototype was undergoing systems testing in September 1996, preparatory to joining the flight-test programme in December. The governments of Russia and Ukraine have options to purchase 400 An-70 aircraft, though at present not all can be funded. These will be produced by a 13-company joint venture called STS, with final assembly in both countries (Samara and Ukraine). All engines would be produced by Ivchenko Progress, possibly assisted by Motor Sich. At the 1996 Farnborough airshow Antonov General Designer Pyotr V. Balabuyev tried to interest European manufacturers in collaborating on the An-70 as an FLA (A400M) already in the air. So far a total of 15 D-27 development engines have been run. Four were lost in the first An-70, leaving 11 in the test programme, which by early 1999 had completed 15,000 hours. Of these, some 400 were in flight, almost all in the No 2 An-70 but including initial testing in the No 2 position on a Zhukovski Il-76LL testbed and brief flying in the first An-70. In 1999 the An-70 test programme was based at the Antonov complex at Kiev.

An-180

Antonov design bureau has planned a totally different and unconventional installation of two tractor D-27 engines for this 150/200-passenger aircraft, each being mounted on the tips of the tailplane. Money has not yet been committed to build a prototype.

Be-42

Also known as the A-45 for turbofan propulsion, this multirole amphibian may now fly powered by two D-27s mounted on the tips of an auxiliary wing carried on top of the existing engine pylons. In this application the T-O power is given as 10,291 kW (13,800 shp). Ivchenko Progress General Designer Muravchenko regards the combination as potentially important.

Yak-42-2: Whereas the engine and both the preceding aircraft are Ukrainian, the Yak Corporation is Moscow based. In the Yak-46-2 two D-27 engines would be installed in a back-to-front arrangement with front ram inlets and pusher propellers, each mounted on a short pylon strut on the rear fuselage. Compared with tractor installations this is expected to improve propulsive efficiency by at least eight per cent.

Yak-46

This 150/156-seat twin-jet was designed to be powered by D-627 engines (see later), which after 2001 would be replaced by two engines derived from the D-27 but with 3.8 m (12 ft 5.6 in) open contrarotating fans (T-O rating 109.8 kN (24,700 lb st).

A400M

The D-27 would be ideally suited to the Airbus Military Co A400M (previously Future Large Aircraft). No suitable engine exists in the West, and even the gearbox would present a major challenge. Instead of adopting the D-27, Western companies have been trying to find a way to use the Progress technology. Rolls-Royce Deutschland was one of the companies eager for an A400M contract and, since 1994, had been talking with Progress about possible collaboration. However, by 2001 any use of D-27 technology appears to have been rejected.

By July 2000 a total of 16 D-27 engines had run on test and in flight, logging 6,000 hours.

Type

Three-shaft propfan.

Intake

Direct annular intake surrounding the propeller reduction gear.

LP Compressor

Five-stage axial, with variable IGVs and first two stator rows. Mass flow (T-O) 27.4 kg (60.41 lb)/s. Note: the D-136 has a six-stage LP compressor.

HP Compressor

Two-stage axial followed by one centrifugal. OPR, 29.25.

Combustion Chamber

Annular, of part-conical form with air delivered from the HP diffuser round the outer periphery. Fuel nozzles are diagonal with a 90º bend on entering the chamber. Two radial high-energy igniters.

HP Turbine

Single stage with air-cooled single-crystal blades, driving HP compressor at high rpm. Max TGT 1,367ºC.

IP Turbine

Single stage, driving LP compressor.

Power Turbine

Four stages, driving via a long quill shaft to the reduction gear. The two-stage epicyclic reduction gear (see section drawing) provides contrarotating outputs for the Aerosila-developed open propfan described previously. Incorporates thrust meter.

Jetpipe

Single pipe at rear with fixed nozzle.

Control System

Dual FADEC, plus auto checkout.

Accessories

Mounted on three remote gearboxes, one below the compressor case driven from the HP spool, and one above and one below the intake casing driven from the main reduction gear.

Dimensions

Length	4,198 mm (165.28 in)
Width	1,260 mm (49.61 in)
Height	1,370 mm (53.94 in)
Fan diameter	4,500 mm (177 in)

Weight, Dry

Without propfan	1,650 kg (3,638 lb)
Total	2,300 kg (5,071 lb)

Performance Ratings

T-O, S/L, ISA:	10,440 ekW (14,000 shp)
Max cruise (11,000 m; 36,089 ft at M0.75)	5,033 ekW (6,750 ehp)

T-O thrust, S/L:	109.8 kN (24,690 lb st)

Specific Fuel Consumption

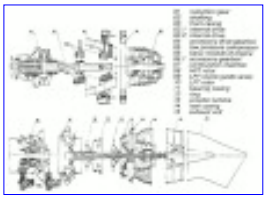
T-O, as above	63.34µg/J (0.375 lb/h/shp)
Cruise, as above	3.68 mg/Ns (0.13 lb/h/lb st) (48.44 µg/J)/(0.287 lb/h/lehp) 3.825 mg/Ns (0.135 lb/h/lb st)

UPDATED

Three views of D-27 with SV-27 propeller (1998)

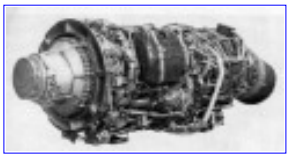
Three views of D-27 with SV-27 propeller

Three views of D-27 with SV-27 propeller (1998)



D-27 modules (2001)

Longitudinal section through D-27



D-27 prototype power section

D-27 on outdoor testbed with SV-27 propeller

D-27 test installation in Il-76LL

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AERO-ENGINES - TURBOFAN, UKRAINE

Date Posted: 12 April 2001

Jane's Aero-Engines 10

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

D-436

This is a growth version of the D-36, which has been developed through several distinct stages:

D-436K

Initial version, designed from 1980 to meet preliminary requirements of later versions of the Yak-42 and An-72 and -74. Improved aerodynamics and higher rotational speeds. BPR 6.2, OPR 21.0. The D-436K first ran in 1985, when the D-36 had flown 1 million hours. Certificated in 1987.

D-436T

Fan with improved aerodynamics (BPR 6.0); compressors unchanged, but run faster to give increased pressure ratio (21.9); combustion chamber modified for higher TGT (1,197ºC) and better high-altitude starting with 18 single-jet nozzles, (some with pneumatic atomisers), and an HP turbine and peripheral cascade-type reverser scaled from those of the D-18T. Tested from 1990. Selected for EKIP L3-1 flying vehicle.

D-436T1

Considerably redesigned to meet requirements of future regional aircraft. Designed for rear-fuselage or underwing mounting. Service life 24,000 hours. Selected for Tu-334-100. Originally scheduled for certification 1996 but delayed three years by lack of finance and other factors. The first Tu-334-100 began its flight-test programme on 8 February 1999. According to Vladimir Suralev, deputy director of Salyut (which see), Iran has been discussing

the manufacture under licence of both the Tu-334-100 aircraft and its engine. Salyut will share domestic D-436 production with UMPO.

D-436T1-134

A version of the D-436T1 tailored to re-engining surviving Tu-134 family aircraft, replacing the Aviadvigatel D-30 III.

D-436T2

Almost identical to D-436T1 but uprated. Service life 24,000 hours. Selected for Tu-334-100D.

D-436TP

Marinised version derived from D-436T1 with improved economy and reduced emissions. Powers Be-200.

D-436T3

Growth version with larger fan. Intended for Tu-230 and Tu-334-200.

The D-436, designed in Ukraine, is one of the most important engines in the Russian Federation because it powers the basic versions of the Tu-334. In Ukraine itself it powers the Be-200, which is also seen as an important product with export possibilities. Though export versions of both aircraft might have foreign engines, such as the BR715, a major production plan for D-436 versions has been arranged. In 1999 tooling-up was almost complete at Motor Sich in Ukraine, at Ufa (see UMPO) and at MMPO ``Salyut'' (salute) in Moscow, Russia. As noted in the ZMKB introduction, Russia is assisting in D-436 funding.

The following are the principal changes in the D-436T1/T2:

Fan

Scaled from D-18T (D-436T2 with similar pointed spinner), higher rpm but diameter 1,373 mm (54.06 in). BPR (T1) 4.98, (T2) 4.89. Acoustic-panel area in duct doubled. Core booster stage added on fan shaft, so that without changing IP and HP compressors OPR reaches (T1) 22.17, (T2) 24.2.

Combustion Chamber

Modified for reduced emissions, with increase in TGT to (T1) 1,247ºC, (T2) 1,277ºC.

HP Turbine

3-D aerodynamics for NGVs and rotor, and improved cooling. TET (T1) 1,197ºC; (T2) 1,252ºC.

IP Turbine

Improved NGV aerodynamics.

LP Turbine

Disc strengthened for higher rpm.

Control System

Engine fitted with sensors to allow a FADEC to be developed. D-436TP has `electronic control and diagnostics'.

Dimensions

As D-36, except T2 has long pointed spinner.

Weight, Dry

D-436K, T	1,124 kg (2,478 lb)
D-436T1, T2, TP	1,450 kg (3,197 lb)
D-436T3	1,550 kg (3,417 lb)

Performance Ratings

T-O, S/L:	
D-436K, T	73.53 kN (16,535 lb st)
D-436T1, TP	75.0 kN (16,865 lb st) to 30ºC
D-436T2	81.96 kN (18,430 lb st) to 30ºC
D-436T3	91.66 kN (20,613 lb st)
Cruise (8,000 m; 26,250 ft at M0.75):	
D-436K	18.63 kN (4,189 lb)
Cruise (11,000 m; 36,090 ft at M0.75):	
D-436T	12.75 kN (2,866 lb)
D-436T1, TP	14.71 kN (3,307 lb)
D-436T2	16.67 kN (3,748 lb)
D-436T3	19.6 kN (4,409 lb)

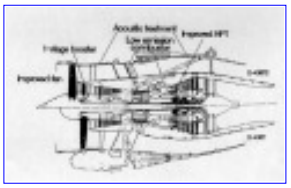
Specific Fuel Consumption

Cruise, as above:	
D-436K	18.4 mg/Ns (0.65 lb/h/lb)
D-436T	17.8 mg/Ns (0.63 lb/h/lb)
D-436T1, T2, TP	17.43 mg/Ns (0.617 lb/h/lb)
D-436T3	14.4 mg/Ns (0.51 lb/h/lb)

UPDATED

D-436K

D-436T2



Longitudinal section through D-436T (lower half) and D-436T2

D-436T2
(1998)

D-436TP
(1998)

D-436TP
(1999)

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AERO-ENGINES - TURBOPROP, UKRAINE

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Jane's Aero-Engines 10

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

AI-30

Under this designation the Ivchenko Progress design team has derived from the TV3-117VMA-SB2 (see under Klimov, Russia) an upgraded turboprop with the above designation. It is the baseline power plant of the An-140 short-haul airliner, and is also a candidate engine for the Il-114, MiG-110 and Tu-130. In the An-140 it drives the Stupino (Aerosila) AV-140 six-blade propeller. Production engines are to be manufactured at the associated Motor Sich plant. Differences from the original Russian engine are minor and in the latest ZMKB literature the original `VMA-SB2' designation is used.

Performance Ratings

Maximum contingency OEI	2,088 kW (2,800 shp)

Specific Fuel Consumption

Maximum cruise power	71.75 µg/J (0.425 lb/h/shp)

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Date Posted: 12 April 2001

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IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

AI-25

This small turbofan was produced to power the Yak-40 three-engined STOL local-service transport. It was accordingly designed under A G Ivchenko to be robust, simple and reliable in the world's harshest climates whilst maintained with simple tools. No attempt was made to achieve outstanding performance. Two versions were produced, the production factory being Motor Sich.

AI-25

Baseline engine, designed in 1965, first tested in 1966 and cleared for production at Zaporozhye in late 1967. Powers Yak-40 (a few military Yak-40 were powered by the AI-25TL). Service life 16,000 hours. More than 6,300 of this version were delivered. By mid-1999, when the engine designation had become AI-25 Series 2E, accumulated operating time was 56.3 million hours.

AI-25TL

Fitted with a zero-stage on the compressor and cooled rotor blades on the HP turbine. First tested in 1970. More than 2,650 of this version were delivered. Powers all production versions of Czech L-39. Selected for Aeroprogress (ROKS) T-910 Kuryer. From 1997 engines in the Czech Republic are being upgraded by LOM Praha. By mid-1999 accumulated operating time was 5.7 million hours.

AI-25TL Series 2

This engine has been developed to offer an indigenous power plant for the MiG-UTS twin-engined trainer. Featues include separate bypass and core nozzles, modified accessories (including an electric starter controlled by an automatic starting and flight control system) and drive gearbox, full anti-icing and a lubrication system for sustained inverted flight.

Type

Two-shaft turbofan.

Intake

Fabricated from sheet titanium. Central bullet, outer ring and 25 inlet guide vanes de-iced by HP bleed air.

Fan

Three-stage axial. Titanium-alloy blades pinned into grooved rims of discs bolted together. Casing and fan duct of magnesium alloy. Pressure ratio 1.695 at 10,560 rpm. Mass flow 44.8 kg (98.8 lb)/s. Bypass ratio 2.2. AI-25TL and AI-25TL Series 2, 46.8 kg (103.2 lb)/s, bypass ratio 1.98.

Compressor

Eight-stage axial. All-titanium rotor, with blades held in dovetail grooves in discs connected by two sets of bolts (see section drawing). Casing aluminium and magnesium. Pressure ratio 4.79 at 16,640 rpm. OPR 8.1. AI-25TL, zero stage added. Pressure ratio 5.66, giving OPR 9.6.

Combustion Chamber

Annular, with nickel-alloy flame tube fabricated to incorporate 12 forward cones each surrounding a fuel nozzle with centrifugal stabiliser vanes.

HP Turbine

Single stage, with air-cooled NGVs. AI-25, solid shrouded rotor blades held by fir-tree roots in cooled disc. Maximum TET 908ºC. AI-25TL, air-cooled blades, maximum TET 993ºC; AI-25TL Series 2, 955ºC.

LP Turbine

Two-stage, with shrouded blades held in cooled discs bolted together.

Jetpipe

Plain convergent nozzles for core and fan duct, no mixer or reverser.

Control System

Hydromechanical, with main and emergency standby system.

Oil System

Self-contained, pressure circulating. AI-25TL, designed for sustained inverted flight.

Starting

SV-25 pneumatic starter supplied from ground hose, cross-bleed or (Yak-40) AI-9K APU. AI-25TL has automatic-control electric starter.

Fuel Specifications

T-1 or TS-1 to GOST 10227-62, equivalent to DERD.2494 or MIL-F-5616.

Oil Specifications

Under fan duct, driven off front of HP spool. Equipment includes comprehensive ice and fire protection, oil chip detector and vibration monitor.

Dimensions

Length:	
AI-25	1,993 mm (78.46 in)
AI-25TL with jetpipe	3,358 mm (132.2 in)
AI-25TL Series 2	2,322 mm (91.42 in)
Width:	
AI-25	820 mm (32.28 in)
AI-25TL	985 mm (38.78 in)
AI-25TL Series 2	861.5 mm (33.92 in)
Height:	
AI-25	896 mm (35.28 in)
AI-25TL	958 mm (37.72 in)
AI-25TL Series 2	931 mm (36.65 in)

Weight, Dry

AI-25	312 kg (688 lb)
AI-25TL, AI-25TL Series 2	350 kg (772 lb)

Performance Ratings

T-O, S/L:	
AI-25	14.71 kN (3,307 lb st)
AI-25TL, AI-25TL Series 2	16.87 kN (3,792 lb st)
Cruise (6,000 m; 19,685 ft at 550 km/h; 342 mph):	
AI-25	4.34 kN (976 lb)
Cruise (8,000 m; 26,250 ft at 550 km/h; 342 mph):	
AI-25TL, AI-25TL Series 2	5.05 kN (1,135 lb)

Specific Fuel Consumption

Cruise, as above:	
AI-25	22.51 mg/Ns (0.795 lb/h/lb)
AI-25TL	23.08 mg/Ns (0.815 lb/h/lb)
AI-25TL Series 2	22.85 mg/Ns (0.807 lb/h/lb)

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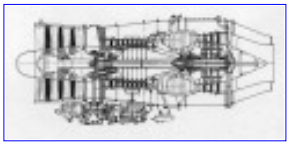
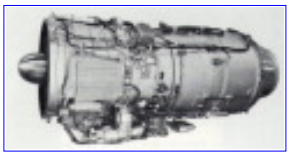
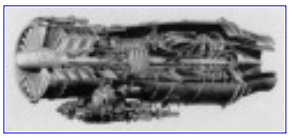


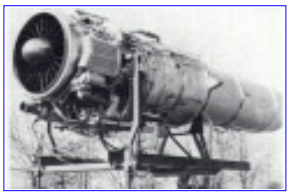
Longitudinal section through AI-25



AI-25 turbofan



Cutaway drawing of AI-25



AL-25TL

AI-25TL Series 2 (1999)

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AERO-ENGINES - TURBOPROP, UKRAINE

Date Posted: 12 April 2001

Jane's Aero-Engines 10

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

AI-20

This turboprop was the first gas-turbine engine designed by the Zaporozhye collective, headed by Ivchenko. The AI-20 was developed in 1955-57 as a competitor of the NK-4 by Kuznetsov design bureau for the Il-18 aircraft, using the experience of Mikulin design bureau in developing the AM-3M engine, and the expertise of CIAM. Mass production of AI-20 began in 1958 for Il-18 and An-10A.

The following variants were developed:

AI-20K

For Il-18V, An-10A and An-12. Rated at 2,983 ekW (4,000 ehp) at S/L, ISA. Produced at Perm Motors, and by SMPMC in China as WJ6.

AI-20A

Rated at 2,983 ekW (4,000 ehp). Qualified 1961 and fitted to An-12A.

AI-20M

Known as AI-20 Series 6. Initial T-O rating 3,126 ekW (4,192 ehp), later increased to 3,169 ekW (4,250 ehp). Fitted to An-12BK and derivatives and Il-18/20/22/38. Differs from AI-20 by shrouded turbine blades.

AI-20DK

Known as AI-20D Series 3. Rated at 3,124 ekW (4,190 ehp), navalised for An-8, An-12M and Beriev Be-12.

AI-20DM

Known as AI-20D Series 4. Rated at 3,863 ekW (5,180 ehp). Fitted to An-8, An-12M, Be-12 and Be-14.

AI-20D Series 5

Rated at 3,863 ekW (5,180 ehp), to operate from -60 to +55ºC with automatic variation of propeller pitch. Series 5 and 5E engines power the An-32.

AI-20DME, AI-20DMN

Versions used in vehicle-mounted ground power units, with output 1,600 or 2,500 MW.

The principal production factory was Motor Sich (which see), which delivered more than 13,800 AI-20 engines of several versions. Service life and TBO of different modifications depend on service conditions and are: service life from 6,000 hours (AI-20DM) to 24,000 hours (AI-20M); TBO 7,000 hours (AI-20M). By mid-2000 total operating time had reached 90.5 million hours.

The following description refers to the AI-20M:

Type

Single-shaft turboprop.

Intake

Inner and outer cones connected by six radial struts. Outer casing carries accessories and front mountings. Centre casing carries reduction gear.

Compressor

Axial, 10 stages, with four bypass valves, which are used at starting and transient ratings. Spool carried by forward extension shaft in roller bearing and bolted rear shaft in ball thrust bearing. Pressure ratio from 7.6 at T-O (ground) to 9.2 (cruise). Mass flow 20.7 kg (45.6 lb)/s. Stator casing of sheet stainless steel in upper/lower halves.

Combustion Chamber

Annular with 10 burner cones and two pilot burners and igniter plugs. The casing is one of the load-carrying elements of the engine.

Turbine

Axial, three stages. Rotor blades shrouded at inner and outer ends and installed in pairs in slots of air-cooled discs. Assembly overhung on cantilevered shaft running in roller bearing in tapered cone of combustion chamber, splined to compressor drive. First guide vanes and discs are cooled by secondary air from combustion chamber. Maximum entry temperature is 900ºC at S/L (937ºC for AI-20DM, 927ºC for AI-20D Series 5). Rotor speed 12,300 rpm, except 10,400 at ground idle.

Jetpipe

Fixed-area type with five radial struts. Nozzle area 0.225 m^2 (2.42 sq ft).

Output

Planetary type, two-stage, incorporating a six-cylinder torquemeter and type IKM negative-thrust transmitter. Reduction ratio 0.08732.

Accessories

Engine and airframe accessories driven off compressor front extension shaft, via radial shafts at 6 and 12 o'clock. Full ice protection and fire extinguishing systems.

Starting

Two electric starter/generators. Type STG-12-TMO-1000, supplied from ground or TG-16 or AI-8 APU.

Fuel Specifications

T-1, TS-1, T2, RT, to GOST-10227-86 (DERD.2492, JP-1 to MIL-F-56616).

Oil System

Pressure-feed type with full recirculation.

Oil Specifications

75 per cent MK-8 or MK-8P to GOST-6457-66 (DERD.2490 or MIL-O-6081B) and 25 per cent MS-20 or MK-22 to GOST 21743-76 (DERD.2472 or MIL-O-6082B).

Dimensions

Length	3,096 mm (121.89 in)
Width	842 mm (33.15 in)
Height	1,180 mm (46.46 in)

Weight

Dry	1,040 kg (2,293 lb)

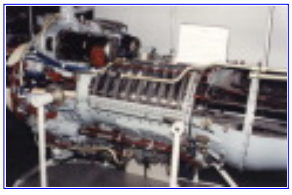
Performance Ratings

T-O (S/L, static)	3,169 ekW (4,250 ehp)
Max cruise (340 kt; 630 km/h; 391 mph at 8,000 m:	
26,250 ft)	1,986 ekW (2,700 ehp)

Specific Fuel Consumption

T-O	89.42 µg/J (0.529 lb/h/ehp)
Cruise, as above	73.3 µg/J (0.434 lb/h/ehp)

UPDATED



Close-up of cutaway AI-20

Longitudinal section through AI-20M

AI-20D Series 5
(2001)

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AERO-ENGINES - PROPFAN, UKRAINE

Date Posted: 12 April 2001

Jane's Aero-Engines 10

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

D-236

This propfan demonstrator was designed on the base of the D-36 core, but it incorporates a reduction gearbox to drive the SV-36 contrarotating propfan. Design began in 1979. The aim was to create a demonstrator for studying the geared propfan. Bench tests began in 1985, and the tests in the Il-76 flying testbed began in 1987.

The Yakovlev design bureau has been working on propfan integration since 1987. The Yak-42E-LL flying testbed with a D-236 replacing one D-36 first flew in March 1991.

Type

Three-shaft propfan.

Compressor Turbine

Generally as D-136.

Power Turbine

Three-stage, shrouded rotor blades.

Output

Planetary, single-stage, ratio 5.67; contrarotating output shafts, ratio of outer to inner shaft torque is 41 to 59 per cent. Drives an SV-36 propfan by Stupino Design Bureau of Machine-Building. Blades of composites without metal elements. Front propeller eight, rear propeller six blades. Rotation speed, T-O 1,000 rpm, cruising ratings 950 rpm, ground idle 500 to 600 rpm. Front and rear propeller speeds maintained equal at all ratings. Blade angle

control, digital electronic with hydraulic back-up. At maximum cruise rating (as below) propeller efficiency 0.87; at take-off rating thrust to power ratio 0.94. Propfan diameter 4.20 m (165.4 in).

Control System

D-36 type, hydromechanical with electronic unit limiting gas temperature and rotor speeds. Independent pneumonic (bubble memory) system controlling compressor blow-off valves.

Performance Ratings

T-O (S/L, ISA)	8,090 kW (10,850 shp)
Max cruise (11,000 m; 36,090 ft, M0.7)	4,730 ekW (6,340 ehp)

Specific Fuel Consumption

T-O, as above	77.09 µg/J (0.456 lb/h/shp)
Cruise, as above	59.98 µg/J (0.354 lb/h/shp)

UPDATED

D-236 development engine

D-236 installed in YAK-42E-LL (Paul Jackson)

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AERO-ENGINES - TURBOSHAFT, UKRAINE

Date Posted: 12 April 2001

Jane's Aero-Engines 10

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

D-136

This turboshaft engine was designed for the Mi-26 transport helicopter. Bench testing started in 1977, and series production in 1982. According to Mil, by 1999 a total of 280 Mi-26 helicopters had flown, though in 2001 ZMKB `Progress' gave the total deliveries of D-136 engines as only 470. Engines are made by Motor Sich.

The D-136 is composed of 10 modules, each of which (except for the main module) can be removed and replaced without disturbing neighbouring modules on an installed engine. Five gas-generator modules are identical with those of the D-36.

Type

Two-spool free turbine turboshaft.

Compressor

As D-36 but with redesigned intermediate case between LP and HP compressors (no fan duct). Speed (maximum): LP 10,950 rpm, HP 14,170 rpm. Mass flow 36 kg (79.4 lb)/s. OPR 18.3.

Compressor Turbines

Both single stage. TGT (normal) 1,205ºC, (max) 1,243ºC.

Power Turbine

Two-stage with uncooled nozzle guide vanes and tip-shrouded blades, with air-cooled discs. Combined as separate module with support housing, outer and inner casings connected by 11 wide load-bearing struts. Inner

casing contains front roller and rear ball bearings of free turbine. Struts carry oil pipes and speed sensor drive.

Output

Flexibly mounted shaft at rear, transmits torque from free turbine rotor to helicopter transmission by splines.

Jetpipe

Curved to side.

Accessories

Mounted on remote gearboxes above and below intermediate case, driven off HP shaft.

Starting

By air turbine starter.

Control System

Hydromechanical speed governor of free turbine with power synchroniser of both Mi-26 engines. Free turbine speed maintained at 8,300 rpm, changed by pilot command in range of ±300 rpm. Electronic control of gas temperature and speed of free turbine and gas generator rotors. Compressor blow-off valves controlled by self-contained pneumonic system. On Mi-26, engine is equipped with particle separator.

Dimensions

Length	3,964 mm (156.06 in)
Width	1,382 mm (54.4 in)
Height	1,124 mm (44.25 in)

Weight

Dry	1,050 kg (2,315 lb)

Performance Ratings

(ISA, S/L)

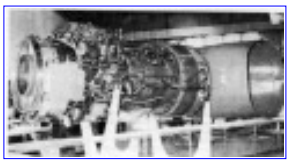
Max contingency	8,500 kW (11,399 shp)
T-O	7,457 kW (10,000 shp)
Max cruise	6,338 kW (8,500 shp)

Specific Fuel Consumption

T-O rating	73.7 µg/J (0.436 lb/h/shp)

UPDATED

Longitudinal section through D-136 (jetpipe moved 90º to discharge downwards)



D-136, jetpipe to left side

D-136, right side

D-136
(2001)

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AERO-ENGINES - TURBOFAN, UKRAINE

Date Posted: 12 April 2001

Jane's Aero-Engines 10

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

D-36

The D-36 was the first Soviet engine with a high-bypass ratio to fly. It was designed for the Yak-42, and is a three-shaft engine composed of 12 modules. Bench tests began in 1971, flight tests in 1974, and series production in 1977.

D-36 Series 1

Baseline engine, in service with Yak-42.

D-36 Series 3A

In production by Motor Sich for An-72 and An-74. Other D-36 versions power several types of twin-engined *Ekranoplans* (surface-effect ships).

Series D-36 engines have been produced by Motor Sich (which see). By September 1996 D-36 engines installed in the Yak-42 and An-72/-74 aircraft had flown almost 3 million hours, and this rose to 5.95 million by January 2000. By 2001 deliveries had tapered off at `more than 820'. According to Dr Vyacheslav Boguslayev, Director-General of Motor Sich, they have `an outstanding record of reliability'.

The D-36 is the base engine for a whole family: the **D-136** turboshaft; **D-236** propfan demonstrator; **D-336** industrial engine, and **D-436K/T**, **D-436T1/T2** turbofans. The D-36 was also used as a model for attaining design objectives of the D-18T.

Type

Three-shaft turbofan.

Fan

Single-stage, transonic, diameter 1,373 mm (54.06 in); 29 titanium blades with part-span shrouds; 48 outlet guide vanes (the number of blades and vanes chosen for minimum noise). Blade containment by winding Kevlar fibre on the fan casing. Short fan duct on Yak-42, common nozzle for fan and core on An-72 and An-74. Bypass ratio 5.6. Maximum speed 5,400 rpm. Mass flow (Series 1) 255 kg (562 lb)/s; (Series 3A) 260 kg (573 lb)/s.

IP Compressor

Six stages with inlet guide vanes adjusted on the bench, then fixed in position. Three blow-off valves. Discs and rotor blades of titanium, stator vanes of steel. Maximum speed 10,500 rpm.

HP Compressor

Seven-stage, with adjustable inlet guide vanes. Three blow-off valves. Rotor blades and discs of two aft stages of steel. Maximum speed 14,170 rpm. OPR 20 (T-O), 18.7 (cruise).

Combustion Chamber

Annular, with 24 burners and two igniters. Integral with HP turbine nozzle guide vanes (they form a single module). Combustion chamber case made by explosion stamping. Flame tube elements rolled and welded into one unit.

HP Turbine

Single stage. Maximum TGT (Series 1) 1,177ºC; (Series 3A) 1,211ºC. Rotor blades tip-shrouded, convective-film cooling system, attached by fir-tree with two blades in each groove. Nozzle vanes have convective cooling.

IP Turbine

Single stage. Uncooled, tip-shrouded, rotor blades. Nozzles guide vanes cooled by third-stage HP compressor air. The support housing is a module located between the HP and IP turbine rotors and combined with wide IP turbine nozzle vanes. Outer casing attached to inner casing by spokes passing through wide hollow nozzle vanes. Inner casing carries rear bearings of HP and IP rotors. Oil supplied through hollow nozzle vanes.

LP Turbine

Three-stage with tip-shrouded blades, air-cooled discs.

Jetpipe

Consists of rear-bearing housing and main duct nozzle. Rear engine mount attached to housing.

Reverser

Can be installed downstream of bypass duct. Not considered part of engine.

Mounting

Universal: under and over the wing, in the fuselage and on both sides without changing engine. Front mount attached to intermediate case cast in magnesium alloy.

Accessories

Mounted on gearbox under intermediate case joining IP and HP compressors, driven off HP shaft.

Starting

By air-turbine starter, on accessory gearbox, from ground source, APU or operating engine. In-flight starting can be assisted by starter.

Control System

Hydromechanical, with inner redundancy and electronic unit controlling gas temperature and rotor speeds. Compressor blow-off valves controlled by independent pneumonic (bubble memory) system located near the valves. Engine compressor air is the operating medium. The D-36 is provided with sensors sufficient for FADEC control. Testing with DSIC FADEC began August 1992.

Dimensions

Series 1:	
Length	3,470 mm (136.6 in)
Diameter	1,373 mm (54.06 in)
Series 3:	
Length	3,192 mm (125.66 in)
Width	1,541 mm (60.67 in)
Height	1,711 mm (67.38 in)

Weight, Dry

Series 1	1,109 kg (2,445 lb)
Series 3A	1,124 kg (2,478 lb)

Performance Ratings

(ISA)

T-O static	63.74 kN (14,330 lb st)
Max cruise at 8,000 m (26,250 ft) at M0.75	15.7 kN (3,527 lb)

Specific Fuel Consumption

T-O	10.195 mg/Ns (0.360 lb/h/lb st)
Cruise, as above	18.41 mg/Ns (0.650 lb/h/lb st)
Max cruise, as above	18.4 mg/Ns (0.65 lb/h/lb)

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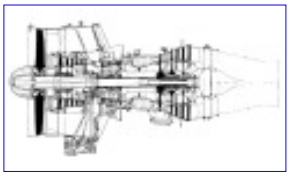
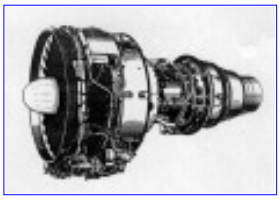
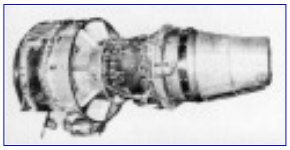


Longitudinal section through D-36



D-36



D-36

D-36 Series 3A
(1998)

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AERO-ENGINES - TURBOFAN, UKRAINE

Date Posted: 12 April 2001

Jane's Aero-Engines 10

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

D-18T

The D-18T was the first large high-BPR turbofan in the Soviet Union to be certificated and cleared for production. It was designed by a team led by Lotarev in order to power the Antonov An-124 Ruslan. It incorporates several design features originally introduced with the smaller D-36, and is likewise a three-shaft engine.

D-18T design began in 1977, and a single-spool core engine was tested in 1979. A two-spool core engine was developed, and the first run of a full-scale engine was accomplished in September 1980. First flight on the Il-76T flying testbed was carried out in March 1982. On 24 December 1982 the An-124 made its first flight. The D-18T also powered the An-225.

D-18T (Series 3)

Current production engine, made by Motor Sich. Operators of the An-124, such as Volga-Dnepr, had for several years been considering replacement by Western engines. Progressive Series 3 upgrades have made this unlikely. In 2000, TBO reached 4,000 hours, and 8,000 hours is considered possible `in the near term'. Fully certificated by MAK. By 2001, total deliveries of D-18T engines had exceeded 184, of which 55 were Series 3 engines.

Type

Three-shaft turbofan. Each of the three rotors is carried on two bearings (total 6). The bearings are resilient and resilient/damping. Oil cavities of the fan and IPC bearings are provided with labyrinth seals, others with radial-face contact seals. The engine comprises 17 modules.

Fan

Single-stage, supersonic, with 33 titanium blades with part-span shrouds. Stator has 56 carbon-glass composite plastic vanes, with epoxide matrix and inner and outer titanium shrouds. Leading-edge protected by stainless-steel strip. Containment of separated blade provided by winding Kevlar-type fibre on the inside surface of the case. Blades attached by fir-tree roots in disc slots. Speed (maximum) 3,450 rpm. Mass flow 765 kg (1,687 lb)/s. Bypass ratio 5.6, (Series 3) 5.7.

IP Compressor

Seven-stage, transonic, with variable inlet guide vanes and eight blow-off valves on case. Titanium blades and steel vanes. Speed (maximum) 5,900 rpm.

HP Compressor

Seven-stage with adjustable inlet guide vanes. Blades of first four stages of titanium, remainder of steel. Speed (maximum) 9,100 rpm. Overall pressure ratio (T-O) 25, (cruise) 27.5.

Intermediate Case

Intended to form transition path from IPC to HPC and fan duct as well as attachment of HPC bearing and IPC and HPC cases. Drive from HP rotor to accessory gearbox in lower part of case. Front engine mounts attached to inner case. Aluminium outer shell riveted. Inner shell and struts of titanium.

Combustion Chamber

High temperature, annular, with 22 main fuel nozzles and two igniters. Integrated with HPT NGVs (a separate module). Case consists of outer and inner shells. Cooling air (for IPT) passes between them. Combustion chamber specially modified for low emissions.

HP Turbine

Single-stage with tip-shrouded blades with convection-film air cooling. Blades mounted by fir-tree roots (a pair in each slot). Turbine support housing with IP and HP rotor bearings. Vanes air-cooled. TET (Series 3) 1,337ºC at T-O, 1,122ºC cruise (as below).

IP Turbine

Single stage with tip-shrouded blades with convective air cooling.

LP Turbine

Four-stage with uncooled blades with tip shrouds. Rotor of drum-disc type. Outer case cooled by air from fan duct.

Jetpipe

Comprises rear support case with LPT bearing and rear engine mount, and core nozzle. Struts shaped to untwist turbine outlet flow.

Reverser

Attached to rear flange of intermediate case, with 12 doors pulled inwards and blocking fan duct by axial

movement of translating cowl section, which simultaneously opens peripheral cascade rings. Control and drives are hydromechanical, using engine oil.

Accessories

Accessory gearbox drives constant-speed generator and two hydraulic pumps.

Control System

Similar to D-36. Self-contained pneumonic system controls blow-off valves and HPC inlet guide vanes. Engine control provides automatic starting and maintaining given rating over complete operating range.

Oil System

Self-contained continuous circulation under pressure.

Dimensions

Length	5,400 mm (212.6 in)
Fan diameter	2,330 mm (91.73 in)
Width	2,792 mm (109.9 in)
Height	2,937 mm (115.6 in)

Weight, Dry

Series 3	4,083 kg (9,001 lb)

Performance Ratings

T-O (S/L, ISA+13ºC)	229.8 kN (51,660 lb st)
Max cruise (11,000 m; 36,090 ft, M0.75, ISA)	47.67 kN (10,716 lb)

Specific Fuel Consumption

Series 3:	
T-O	9.770 mg/Ns (0.345 lb/h/lb st)
Cruise, as above	16.142 mg/Ns (0.570 lb/h/lb)

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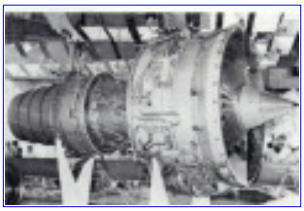
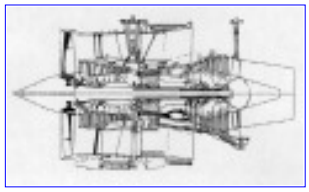


Progress D-18T

D-18T `exploded'



Longitudinal section through D-18T

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AERO-ENGINES - TURBOPROP, UKRAINE

Date Posted: 12 April 2001

Jane's Aero-Engines 10

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

AI-24

This turboprop was designed in 1960 to power the An-24 and its derivatives. There are four main versions:

AI-24 (Series 2)

Rated at 1,875 kW (2,515 shp). Installed on An-24A, An-24B, An-24V, An-24T and An-24PV. Rotor 15,100 rpm except 13,900 at ground idle. Production began in 1964.

AI-24A

Rated at 1,901 kW (2,550 shp), maintained by water injection. Further developed in China (see DEMC).

AI-24P

Multifuel version for Ekranoplan cruise propulsion. Rated at 1,814 kW (2,432 shp). One powers the SM-6 series and two power Meteor 2.

AI-24T

Powers the An-24A, -24B and -24T with water injection. T-O rating 2,074 kW (2,780 shp). Rotor 15,800 rpm except 14,050 at ground idle. Production from 1966.

AI-24VT

Same power as AI-24T. Service life 15,000 hours. Fitted to the An-26 and An-30. Production began in 1970.

AI-24UBE

Flat-rated to deliver 588.5 kW (789 shp) to 10,500 m (34,440 ft), driving accessories including 480 kVA main electric generator of Beriev (Ilyushin-derived) A-50.

AI-23

Designation of derived engines used in ground power installations.

Since 1961 the main source of production AI-24 engines has been Motor Sich (see later in Ukraine) which delivered more than 11,700 of all series versions. The AI-24 is flat-rated to maintain its nominal output to 3,500 m (11,500 ft). TBO was 3,000 hours in the Spring of 1966; by 1968 the later AI-24T had reached 4,000 hours. Service life, 22,000 hours.

Type

Single-shaft turboprop.

Intake

Large magnesium alloy casting, comprising inner and outer cones joined by four radial struts. Carries accessories, reduction gear, front mountings and compressor inlet guide vanes.

Compressor

Ten-stage axial. Stainless steel rotor, comprising rigidly connected discs carrying dovetailed blades. Front shaft runs in roller bearing and is bolted to propeller driveshaft of reduction gear; rear shaft runs in ball thrust bearing and is splined to turbine shaft. Welded steel casing in bolted left and right halves, with welded front and rear connecting flanges. Pressure ratio, 6.40 (AI-24T 7.05, AI-24VT 7.65). Mass flow 13.1 kg (28.9 lb)/s (AI-24T, AI-24VT, 14.4 kg, 31.7 lb/s).

Combustion Chamber

Annular, of spot welded heat resistant steel, with eight simplex burners inserted into swirl vane heads. Contains two starting units, each comprising a body, pilot burner and igniter plug.

Turbine

Three-stage axial. Three discs carry solid blades in fir-tree roots, and are automatically centred on each other when connected by stay-bolts to the extended flange at the rear of the turbine shaft. Shaft splined to compressor rear shaft and held by tie-rod; runs in roller bearing ahead of first turbine disc. Three stator diaphragms through-bolted together and to combustion chamber casing. First nozzle diaphragm cooled by secondary air from combustion chamber. Rotor/stator sealing effected by soft inserts mounted in grooves in nozzle assemblies. TET 797~C. Peak exhaust temperature during starting 750~C.

Jetpipe

Fixed-area type. Inner and outer rings connected by three hollow struts carrying 12 thermocouples.

Output

Planetary type, incorporating hydraulic torquemeter and electromagnetic negative-thrust transmitter for propeller auto feathering. Magnesium alloy casing. Front flange of propeller shaft has end splines and 12 stud holes for type AV-72 propeller of 3,900 mm (153.54 in) diameter turning at 1,245 rpm (AI-24T drives AV-72T propeller, max

1,305 rpm). Ratio 0.08255.

Accessories

Mounted on front casing are starter/generator, alternator, propeller speed governor and centrifugal breather. Below casing are oil unit, air separator and removable box containing lP and HP fuel pumps and drives to hydraulic pump and tachometer generators. Also on front casing are an aerodynamic probe, ice detector, negative-thrust feathering valve, torque transmitter and oil filter. AI-24T and AI-24VT have inflight vibration monitoring.

Starting

Electric STG-18TMO starter/generator supplied from ground power or from TG-16 APU.

Control System

Hydromechanical, with speed maintained by automatic variation of propeller pitch. Auto relief of torque overload, auto shutdown and feathering.

Fuel Specification

T-1, TS-1 to GOST 10227-62 (DERD.2494 or MIL-F-5616).

Oil System

Pressure circulation system.

Oil Specification

Mixture of 75 per cent transformer oil GOST 982-56 or MK-8 (DERD.2490 or MIL-O-6081B) and 25 per cent MS-20 or MK-22 (DERD.2472 or MIL-O-6082B).

Dimensions

Length overall	2,346 mm (92.36 in)
Width	677 mm (26.65 in)
Height	1,075 mm (42.32 in)

Weight

Dry	600 kg (1,323 lb)

Performance Ratings

T-O: See model listing	
Cruise rating at 243 kts (450 km/h; 280 mph) at 6,000 m (19,685 ft):	
AI-24A	1,156 ekW (1,550 ehp)
AI-24T	1,208 ekW (1,620 ehp)

Specific Fuel Consumption

T-O:	
AI-24VT	80.2 Ág/J (0.475 lb/h/shp)
At cruise rating:	
AI-24A	91.3 Ág/J (0.540 lb/h/ehp)

AI-24T	86.0 Ág/J (0.509 lb/h/ehp)
AI-24VT	89.0 Ág/J (0.527 lb/h/ehp)
Oil consumption	0.85 kg (1.87 lb)/h

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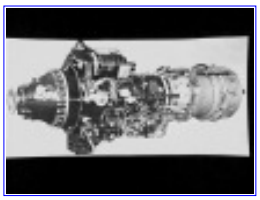

AI-24

Al-24T

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AERO-ENGINES - TURBOFAN, UKRAINE

Date Posted: 30 August 2000

Jane's Aero-Engines 08

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

D-18TM and D-18TR

The D-18TM was also intended to power the An-218. The D-18TR is a further modification. The two engines are of the same design, but have different ratings.

Fan
Single stage, with 35 shrouded titanium blades.

IP Compressor
Seven-stage, with variable inlet guide vanes and first- and second-stage stator vanes, and blow-off valves behind the third-stage.

HP Compressor
Seven-stage, with blow-off valves behind the fourth stage.

Control System
FADEC type.

Dimensions

Length	5,700 mm (224.5 in)
Width	2,982 mm (117.4 in)
Height	2,986 mm (117.6 in)

Weight, Dry

	4,750 kg (10,472 lb)

Performance ratings

T-O:	
D-18TM (Series 3) (S/L, +30ºC, 730 mm Hg)	248.1 kN (55,777 lb st)
D-18TR (Series 3)	269.7 kN (60,627 lb st)
Max cruise:	
D-18TM (Series 3) (11,000 m, MO.8, ISA)	51.49 kN (11,574 lb)
D-18TR (Series 3)	54.33 kN (12,213 lb)

Specific Fuel Consumption

Max cruise, as above	16.57 mg/Ns (0.585 lb/h/lb)

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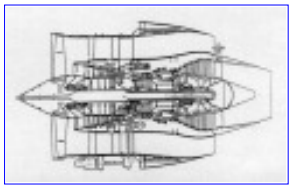

Longitudinal section through D-18TM turbofan (D-18TR similar)

D-18TM

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AERO-ENGINES - TURBOFAN, UKRAINE

Date Posted: 30 August 2000

Jane's Aero-Engines 08

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU

D-18T1

The D-18T1 was intended to power the An-218. It is a derivative of the production D-18T Series 3. Differences from the basic engine are:

The engine has a maximum contingency rating used in the event of failure of the other engine during take-off.

FADEC type control is provided.

Some units (including accessory gearbox, thrust reverser, engine aircraft attachment panels, pipes and electric connections) have been modified.

Dimensions, Weight

As D-18T.

Performance Ratings

Max contingency (S/L, 30ºC, 730 mm Hg)	255.0 kN (57,320 lb st)
T-O (S/L, 30ºC, 730 mm Hg)	214.2 kN (48,148 lb st)
Max cruise (11 000 m, M0.8, ISA +10ºC)	51.58 kN (11,596 lb st)

Specific Fuel Consumption

Max cruise	17.42 mg/Ns (0.615 lb/h/lb)

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AERO-ENGINES - MANUFACTURER, UKRAINE

Date Posted: 17 September 1999

Jane's Aero-Engines 06

ZAPOROZHYE MOTOR-BUILDING COMPLEX

MANUFACTURER DETAILS

This is an alternative name for jsc Motor Sich, see above.

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AERO-ENGINES - MANUFACTURER, UKRAINE

Date Posted: 17 September 1999

Jane's Aero-Engines 06

JUPITER - SMNPP *Yupiter* (Jupiter)

MANUFACTURER DETAILS

330066 Zaporozhye
Tel: (+380 612) 61 46 72/61 49 62
Fax: (+380 612) 65 31 40

Adjacent to Motor Sich, this recently formed enterprise is engaged in the design and development of high-technology products, such as participation in the TV3-117VMA-SBM engine.

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AERO-ENGINES - TURBOFAN, UKRAINE

Jane's Aero-Engines 03

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU - IVCHENKO PROGRESS ZMKB

D-727

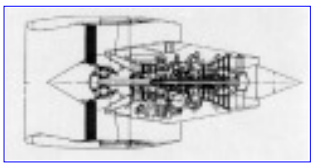
The D-727 turbofan with super high-bypass ratio is intended to power long-range passenger and cargo aircraft. It is being developed on the base of the D-27 propfan two-spool gas generator. The D-727 fan has wide-chord unshrouded blades, and is driven by a three-stage turbine through a reduction gear. Such a configuration allows a bypass ratio of 13.

When designing the D-727 substantiated technical solutions were used which minimised risk and shortened development time. The engine is of modular design and has automatic FADEC control.

A modification is being developed with a T-O rating of 113.0 kN (25,397 lb st), with a maximum contingency rating of 131.0 kN (29,453 lb st). The increased thrust is achieved by modifying the LP compressor.

PERFORMANCE RATINGS:

T-O, S/L	98.0 kN (22,046 lb st)



Longitudinal section through D-727

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AERO-ENGINES - TURBOSHAFT, UKRAINE

Jane's Aero-Engines 03

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU - IVCHENKO PROGRESS ZMKB

D-627

This projected engine is schemed as a high-BPR turbofan, with reverser, based on the core of the D-27. The T-O rating would be 115.7 kN (26,014 lb st). It was selected as the engine for the initial version of Yak-46.

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AERO-ENGINES - TURBOSHAFT, UKRAINE

Jane's Aero-Engines 03

IVCHENKO PROGRESS ZAPOROZHYE MACHINE-BUILDING DESIGN BUREAU - IVCHENKO PROGRESS ZMKB

D-127

The D-127 turboshaft engine is intended to improve the performance of the Mi-26 helicopter and its modifications (beginning with the Mi-26M), as well as to power new advanced heavyweight helicopters. It is a two-spool core engine with a free power turbine. The core is based on the gas generator of the D-27 propfan, but the power turbine has only three stages.

The D-127 is of modular design and has automatic FADEC control.

PERFORMANCE RATING:

T-O	10,700 kW (14,350 shp)

Longitudinal section through D-127

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AERO-ENGINES - TURBOFAN, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE TRENT

Detailed design of this engine began in 1988 to meet the propulsion requirements of the Airbus A330 and, with a larger fan, the Boeing 777. Subsequently, using the inherent flexibility of the three-shaft layout to match the two sizes of fan with independently scaled cores, a further range of engines is being developed to meet the needs of the A340-500/600, A380 and all possible growth versions of 747, 767 and 777. Relationships between the various Trent engines are shown in the diagram below. According to Rolls-Royce, ``Since 1997 the Trent family has gained a market-leading 57 per cent share of all wide-body engine orders''.

Trent designation	Take-off thrust	Application
553	235.8 kN (53,000 lb st)	A340-500
556	249.0 kN (56,000 lb st)	A340-600
758	257.9 kN (58,000 lb st)	A330
768	300.3 kN (67,500 lb st)	A330
772	316.3 kN (71,100 lb st)	A330
875	331.9 kN (74,600 lb st)	777
877	343.4 kN (77,200 lb st)	777
884	377.9 kN (84,950 lb st)	777
892	407.5 kN (91,600 lb st)	777
895	422.6 kN (95,000 lb st)	777

8104	462.68 kN (104,000 lb st)	demonstrator
900	302.5-373.7 kN (68,000-84,000 lb st)	A380

By this means, Rolls-Royce has created a family of technologically advanced engines with all the benefits of derivative design. The use of existing Trent fan diameters leads to a lower cost, shorter and lower-risk development programme. The development of a family of advanced scaled engine cores allows the engine thrust and performance to be optimised for each aircraft application. No other engine covers such a wide range of thrust (see table above).

Trent 700

The Trent 700 first ran in August 1990, and during the development programme routinely achieved thrust levels in excess of 355.9 kN (80,000 lb st). Engine certification was achieved in January 1994, followed by flight testing and subsequent certification on the A330 in December 1994. Before entry into service in March 1995, 90-minutes ETOPS approval was achieved, and this was extended to 120 minutes in December 1995 and 180 minutes in May 1996.

By September 2001, Trent 700 time on-wing exceeded 1.4 million hours with 16 operators in service (Cathay Pacific, Dragonair, Garuda Indonesia, Air Transat, Monarch, Emirates, Air Canada, Air Sri Lankan Airlines, Airtours International, Corsair, Gulf Air, Premiair, SAS, British Midland, TAM and Edelweiss). The 1 million-hour mark was passed in early 2001. The engine ratings available are the Trent **768**, flat-rated to 30ºC at 300.3 kN (67,500 lb st), the Trent **772** flat-rated to 30ºC at 316.3 kN (71,100 lb st) and the Trent **772B** flat-rated to 37ºC at 316.3 kN (71,100 lb st). The Trent 772B speeds and temperatures have been cleared up to 334.1 kN (75,000 lb st). Since the start of 1997, the Trent 700 has captured 43 per cent of all new A330 customer business, extending its overall share of the market to 36 per cent for 22 customers.

The Trent 772 has demonstrated the best take-off for the A330, the lowest overall emissions, and up to 5 dB noise advantage.

Type

Three-shaft turbofan.

Fan

Single-stage with 26 wide-chord snubberless blades. Each blade is manufactured by diffusion bonding and superplastic forming to create a hollow blade with integral canted spars from root to tip. Composite conical spinner. Mass flow (768) 898 kg (1,981 lb)/s, (772) 919 kg (2,027 lb)/s. BPR (768) 5.1, (772) 5.0.

IP Compressor

Eight stages, with variable IGVs and first two stator stages.

HP Compressor

Six stages, with improved tip clearance control based on V2500 technology. IP and HP case fitted with Dunlop bleed and solenoid valves. Mass flow (768) 325 lb/s, (772) 338 lb/s. OPR (768) 33.7, (772) 35.5.

Combustion Chamber

Annular, with 24 burners and two igniters. Liner of nickel alloy materials with thermal barrier coating with improved geometry for NO_X control. Design optimised for reduced emissions.

HP Turbine

Single stage, fitted with internally cooled rotor blades of CMSX-4 third-generation single-crystal material with thermal-barrier coating.

IP Turbine

Single stage with uncooled single-crystal CMSX-4 rotor blades.

LP Turbine

Four stages with 3-D blades.

Jetpipe

Combined fan/core mixer nozzle.

Reverser

Four-door type, with hydraulic operation, supplied by Hurel-Dubois.

Mounting

Hung by two attachments on the core.

Accessories

Single-piece gearbox on the underside of the fan case, supplied by Hispano Suiza. Driven by radial shaft from front of HP spool and thence via step-aside gearbox and inclined shaft in 6 o'clock fan-case strut. Gears and bearings individually replaceable. Separate oil tank and filler.

Control System

Full-Authority Digital (FADEC). Dual-channel, fan-case mounted electronic engine controller. It is contained with a Power Control Unit (PCU) and an Overspeed Protection Unit (OPU) in a sealed box on antivibration mountings.

Oil System

A full flow recirculatory system supplying oil to all bearings, providing lubrication and removing unwanted heat at all operating conditions. Oil is scavenged, filtered, deaerated, cooled and passed back to the high-pressure supply pump.

Dimensions

Fan diameter	2,474 mm (97.4 in)
Overall height (on transportation stand)	3,048 mm (120 in)
Length	3,912 mm (154 in)

Weight, Dry

	4,785 kg (10,550 lb)

Performance ratings

T-O (S/L, ISA):	
Trent 772	316.3 kN (71,100 lb st) to 30ºC
Trent 772B	316.3 kN (71,100 lb st) to 37ºC
Cruise (10,670 m; 35,000 ft, to M 0.82 to ISA+10ºC)	
Both	53.51 kN (12,000 lb)

Specific Fuel Consumption

Cruise, as above	
Both	16.538 mg/Ns (0.584 lb/h/lb)

Trent 800

In 1990, Boeing launched an even larger twin-engined aircraft, the 777. All three engine manufacturers fought to get on this aircraft, to such a degree that leaders of both the US engine firms complained that such competition is financially disastrous. Rolls-Royce might say that it depends how big a share you get. Thanks to the Trent's three-shaft layout, Rolls-Royce were not only able to respond quickly with an engine precisely tailored to the 777, but the resulting Trent 800 is significantly shorter and lighter than either of its rivals. Comparative figures include:

	Length (in)	Dry weight (lb)
Trent 800	172	13,100
PW4098	194.7	16,500
GE90 (basic engine)	204	16,664

Apart from the LP system the Trent 800 family are almost identical to the 700. The LP system now comprises a five-stage turbine driving a fan with the diameter increased to 110 in. In contrast, the PW engines have a seven-stage turbine driving a 112 or 113 in fan, while the GE90 has a six-stage turbine driving a fan with a diameter of no less than 123 in (even larger in the GE90-115B). Fan airflow varies from 2,467 to 2,664 lb/s for the Trent versions in service in early 2000 (Trent 875 to 895), while much greater airflows have been achieved on test. For example, as early as January 1994 a Trent 800 achieved a thrust of 106,087 lb, and in December 1998 a Trent 8104 was tested at a level in excess of 114,500 lb, as noted below.

This engine is regarded as a demonstrator, not a product on offer. It was the first to be fitted with a modified (so-called 3-D design) HP compressor and a fan with a new design of blade. Called `sweptback', though `scimitar' might be more descriptive, the new fan not only increases mass flow, and thus thrust, but also gives higher cruise efficiency and improved resistance to birdstrikes. All Trent 800 versions have a nacelle incorporating a reverser of the sliding cascade type.

The Trent 800 began engine testing in September 1993, and demonstrated a world record of 471.96 kN (106,087 lb st) on 28 January 1994. Engine certification was achieved in January 1995, three months early, at 90,000 lb, the first in the world at this thrust level. The first Boeing 777 with Trent 800 engines flew in May 1995, and entered service on schedule in April 1996. The severe 3,000-cycle ETOPS test was completed in April 1996, and FAA approval for 180-minutes ETOPS was granted in October 1996. Phil Hopton, Director - Boeing Programmes, then said ``The Trent 800 has experienced an exemplary EIS, with very high dispatch reliability rates".

From early 1997, the standard engine has been the Trent 892, which received FAA ETOPS approval in April 1997, effective immediately. As a result of Boeing increasing the maximum take-off weight of the 777-200ER, and because Rolls-Royce had already increased the temperature margins on the current engine design, Rolls-Royce could offer the Trent 895, a more powerful thrust rating of 95,000 lb, to the same production standard as the Trent 892. According to John Cheffins, RR President - Civil Aerospace, ``The Trent is the only power plant available to offer thrusts from 75,000 to 95,000 lb with one engine standard. It gives the 777 unmatched revenue-earning ability". It also gives the 777 a weight advantage of up to 3.6 tonnes (8,000 lb).

In April 1997, a Malaysia Airlines 777-200ER broke the world commercial long-distance record by flying Seattle-Kuala Lumpur (19,012 km, 11,814 miles, 10,267 n miles); it then went on to break the eastbound circumnavigation record by reaching Seattle in 41 hours 51 minutes total time (at an average speed of 890 km/h; 553 mph, 480 kt).

The Trent 800 has been selected worldwide on the Boeing 777. In September 1998, British Airways selected

the Trent 895 to power its additional order of up to 32 777s, instead of the GE90 which powers its previous 777s. Current Trent 800 operators include Thai Airways International, Emirates, American, Delta, Cathay Pacific, Singapore and Malaysia. Apart from BA other customers include ILFC and SALE, and most recently the Trent was chosen by El Al of Israel for 2001 deliveries.

In March 1999, Singapore Aero Engine Services Ltd (SAESL) opened to support Trent 800 engines in that region. Owned by SIA (50 per cent), RR (30) and HAESL (20), it will later handle other engines. HAESL itself supports Trents of Cathay Pacific at Hong Kong.

The inherent growth capability of the Trent 800 was emphasised by the fact that during qualification of the Trent 892 the thrust was frequently well above 100,000 lb, with significant temperature margins still in hand. Rolls-Royce has designed and built the Trent 8104, a high-thrust technology demonstrator. The Trent 8104 features the same fan diameter, engine mounts and architecture as the Trent 895, with improved aerodynamics and durability enhancements to provide 104,000 lb thrust.

At mid-2000, the Trent 800 was in an industry-leading position on the 777 with a 45 per cent market share (282 firm and option aircraft for 11 customers). Since 1997, the Trent has taken 80 per cent of all 777 orders, customers including American, Delta and (changing over from the GE90) British Airways. Rolls-Royce further comments ``The Trent has a 3.6 tonne (8,000 lb) installed weight advantage over the GE90, leading to the best payload/range performance on the 777. With the lowest deterioration rate among the competing powerplants and the best record of reliability, the Trent also delivers cost of ownership benefits. Environmentally the Trent 800 family has the lowest emissions in its class, and beats even the toughest community noise legislation by large margins."

By August 2001, total Trent 800 flight time exceeded 2.3 million hours. Rolls-Royce describe it as ``one of the most reliable aero engines in the world."

The following are differences compared with the Trent 700:

Fan

Single-stage with 26 blades made in the same way but larger. Mass flow (875) 1,119 kg (2,467 lb)/s, (877) 1,134 kg (2,500 lb)/s, (884) 1,175 kg (2,590 lb)/s, (892) 1,200 kg (2,645 lb)/s, (895) 1,208 kg (2,664 lb)/s, (8104) 1,243 kg (2,740 lb)/s. BPR (875) 6.2, (892) 5.8, (895) 5.8, (8104) 5.4. FPR (895) 1.81.

IP Compressor

Similar to Trent 700 but increased capacity from greater FPR.

HP Compressor

Core mass flow (892) (388 lb), (895) 178.2 kg (393 lb)/s, (8104) (428 lb). OPR (892) 40.8, (895) 41.6, (8104) 45.0.

Combustion Chamber

Modified for increased durability.

HP Turbine

Similar to Trent 700, also in CMSX-4®.

IP Turbine

Similar to Trent 700 except for third-generation RR3000 alloy.

LP Turbine

Additional fifth stage.

Reverser

Cascade type with hydraulically-actuated axially-translating cowl. Supplied by Boeing.

Mounting

Hybrid mount including tie strut from compressor case front mount, carrying engine thrust to pylon at rear core mount. Engine is also fan-case supported to carry weight and aerodynamic loads.

Dimensions

Fan diameter	2,794 mm (110.0 in)
Overall height (on transportation stand)	3,076 mm (121.1 in)
Length	4,369 mm (172.0 in)

Weight, Dry

All versions	5,942 kg (13,100 lb)

Performance Ratings

T-O (S/L, ISA)

875	331.9 kN (74,600 lb st)
877	343.4 kN (77,200 lb st)
884	377.9 kN (84,950 lb st)
892	407.5 kN (91,600 lb st)
895	422.6 kN (95,000 lb st)
8104	462.68 kN (104,000 lb st)
Typical cruise (10,670 m; 35,000 ft, M 0.83, to ISA+10ºC)	60.05 kN (13,500 lb)

Specific Fuel Consumption

Cruise, as above	16.283 mg/Ns (0.575 lb/h/lb st)

Trent 500

The third sub-family of Trent engines to be developed are tailored to the stretched Airbus A340-500 and -600. Though sharing the same fuselage cross-section as all the earlier Airbus wide-bodies, these aircraft are significantly larger and heavier, and include the longest-ranged aircraft ever offered to the airlines. A market for such aircraft began to be discerned by 1990, but Airbus spent more than seven years confirming the specification.

From the outset the A340, the first Airbus to have four engines, had been powered by the Franco/US CFM56-5C family of engines, with thrust levels around 32,000 lb. The stretched aircraft was clearly going to need significantly more power, and until 1997 it was expected that its engine would be whatever emerged from CFMXX studies, with thrust in the region of 45,000 lb. Unfortunately for CFM the potential customers kept demanding more, and eventually the Paris/Cincinnati partnership had to admit they could no longer meet the thrust required.

Thanks to the Trent's three-shaft layout, and to Rolls-Royce's exceptional computerised design capabilities, the British company was able to offer the Trent 500 family precisely tailored to the bigger A340. The new aircraft is now in production in two versions. The A340-500 carries up to 440 passengers over sectors up to 9,781 miles. The 340-600 has an even longer fuselage to carry up to 485 passengers up to 8,630 miles.

Currently bottom of the thrust range of the Trent family, this engine matches the Trent 700 fan diameter and external dressings with a Trent 800 core scaled down to about 0.8 size and a new LP turbine matched to the reduced core size. Other features are shown in the accompanying diagram. The result is an engine rated at 235.8 to 275.8 (53,000 to 62,000 lb st), with a BPR of about 8.5, with outstanding fuel efficiency and with a noise

signature dramatically below Stage 3.

In June 1997, the Trent 500 was selected by Airbus as the sole engine for the A340-500 and A340-600. The engine was precisely tailored to this application. Airbus Managing Director Jean Pierson said ``We have selected the Trent 500 because of its low noise, excellent fuel efficiency and low operating costs.'' An additional advantage for the operator is provided by commonality of the engines between the ratings, with only a data entry plug change between the Trent 553 rated at 235.8 kN (53,000 lb st) for the A340-500 and the Trent 556 rated at 249.0 kN (56,000 lb st) for the A340-600. The Trent 500 was certificated at 266.9 kN (60,000 lb st) in December 2000. As it has been tested at 302.5 kN (see below), its temperature margins will be exceptional. Compared with redline the hot-day T-O margin for the Trent 556 will be 70ºC and for the Trent 553 it will be up to 90ºC, making for very long on-wing life.

The Trent 500 is the first engine to incorporate the tiled Phase 5 combustion chamber from the outset. As described and illustrated in the preliminary feature Civil Engines, this combusion chamber has achieved on test a combination of performance qualities unmatched by any other combustor, including those with dual annular rings of burners.

Nacelles for the Trent 500 are in production by Aérospatiale Matra (intake), CASA of Spain (outer panels) and Aircelle (a joint venture of Snecma subsidiary Hispano-Suiza Aerostructures and Airbus). The first reverser was shipped to Airbus in June 1999 and, fitted to a Trent 500 development engine, has carried out a series of simulated landings at Rolls-Royce's outdoor test facility, and the first complete nacelle flew in June 2000 as noted below.

By August 2001 the Trent 500 had won firm and option orders to power an initial 127 aircraft for 10 customers. Bench testing of the Trent 500 began ahead of schedule on 29 May 1999, with over 300 successful hours logged by the end of the year, when six Trent 500 engines had gone on test. Performance has been above prediction; the 60K certification-thrust level was reached within the first 3 hours, and by 10 June 1999 this initial development engine had run at 302.5 kN (68,000 lb st). By the end of the year seven Trent 500s had run, and by June 2000 slightly more hours (1,700) had been logged than on the Trent 800 at the same development stage. Everyone involved in the programme considers this to be an engine of exceptional promise. For example, the engine subjected to bird ingestion lost 2.5 per cent power, one-tenth of the permitted value.

On 20 June 2000, flight testing began with an engine in the port inboard position on an A340-300. Certification was achieved two weeks ahead of schedule on 15 December 2000. Certification was at the level of 266.73 kN (60,000 lb st) to give a large margin over the initial in-service ratings (see below). The first A340-600 was rolled out in March 2001, and made an outstandingly successful 5 hour 28 minute maiden flight on 23 April. EIS is due with Virgin Atlantic in June 2002. The A340-500 is to enter service later in that year, when total Trent time (all versions) should easily exceed six million hours. Over 100 Trent 500 engines are due for delivery within the first six months of service.

Following the first flight of the A340-600, powered by standard production engines, John Cheffins said ``The Trent 500 has been one of the smoothest engine-development programmes we have ever undertaken. It has been on or ahead of schedule ever since we committed to the programme four years ago''. By the end of 2001 four A340-600s were to be in flight test, to be joined by an A340-500 in early 2002. The four aircraft are planned to complete the development programme in 2,400 flying hours.

From the outset, Rolls-Royce has elected to treat the Trent 500 development programme as if it were intended for twin-engined aircraft. Thus, by aiming at full ETOPS qualification it is expected to achieve dispatch reliability of at least 99.95 per cent from the start of service. The following are differences from the Trent 700:

Fan

Wide-chord blades with revised aerodynamics. Mass flow (553) 838 kg (1,847 lb)/s, (556) 858 kg (1,892 lb)/s. BPR 8.5.

IP Compressor

Entire core is basically an 80 per cent linear scale of that of the Trent 892, but incorporating 3-D viscous-flow technology derived from the 8104. Both compressor spools have double casings to separate the load-bearing outer casing from the gas path to maintain very tight blade clearances.

HP Compressor

See IP Compressor. Total OPR (both versions) 36.7.

Combustion Chamber

New Phase 5 design with 20 burners and liner of individually replaceable refractory tiles. Sized to match scale mass flow.

HP Turbine

Matched to reduced core flow. NGVs and rotor blades of aircooled CMSX-4 third-generation single-crystal material. Provided with control system to optimise all tip clearances. Disc fitted with a front-face coverplate to deflect the hottest gas and provide pre-conditioned cooling air for the rotor blades, which are shrouded for total performance retention.

IP Turbine

NGVs of CMSX-4. Rotor blades shrouded.

LP Turbine

Five stages as 800 but with reduced blade height. Blading of new high-lift form.

Dimensions

as 700.

Weight, Dry

	4,717 kg (10,400 lb)

Performance Ratings

T-O (S/L, ISA):	
553	235.8 kN (53,000 lb st)
556	249.0 kN (56,000 lb st)
Cruise (10,667 m; 35,000 ft, M 0.82, to ISA+10ºC)	47.71 kN (10.700 lb)

Specific Fuel Consumption

Cruise, as above	16.08 mg/Ns (0.568 lb/h/lb)

Trent 600

For several years Rolls-Royce has been studying a Trent 600 engine matched to the requirements of possible future upgraded versions of the Boeing 747. Compared with current 747 engines it would offer major reductions in fuel burn and noise. A specification published in Issue 9 of this product no longer applies, and Rolls-Royce wish to give no further details at this time (late 2001).

Trent 900

In October 1996, an MoU was signed between Rolls-Royce and Airbus to supply the Trent 900 engine for the Airbus A3XX, which on go-ahead on 19 December 2000 became the A380. The Trent 900 is being developed within Rolls-Royce to meet the Airbus programme using the Trent family philosophy and three-shaft concept as

the basis. The new engine will match a reduced-weight version of the swept fan of the Trent 8104 to allow use of common accessories, with a scaled-up core from the Trent 500 to match the core flow requirement of the increased thrust.

The Trent 900 meets or exceeds all the demanding payload/range, noise and cost targets set by Airbus. It will be certificated initially at 362.5 kN (81,500 lb st). All ratings up to this level will be achieved with one bill of material. The Trent 900 is capable of growing, with no changes to basic architecture, to 373.7 kN (84,000 lb st). In September 2000 it became the first A380 engine to be specified by a customer (ILFC). A month later this was followed by the first airline customer to select an engine (Singapore Airlines). In consequence the Trent 900 is the lead engine on the A380, and it will be the first into service. Other customers who have selected this engine include Virgin Atlantic and Qantas. By end-September 2001 the Trent 900 had gained 81 per cent of orders and options announced for A380 engines.

Fan

Very similar to Trent 8104, but reduced weight (hence lighter disc and containment). Transportable in 747 freighter without the need to split from the core. Mass flow 1,125 kg (2,481 lb)/s. BPR 7.9.

HP Compressor

Entire core based on Trent 500 and 8104 with 3D viscous-flow technology. OPR 38.2.

Dimensions

Fan diameter: 2,946 mm (116.0 in)

Length: 4,546 mm (179.0 in) .

Weight, Dry

Not stated

Performance ratings

T-O (S/L, ISA)	302.5-373.7 kN (68,000-84,000 lb st)
Cruise (10,670 m; 35,000 ft, M 0.85, ISA	63.2 kN (14,200 lb)

Contract Price

Trent 700 series, November 1997, Trent 772 engines for 19 A330-300, US$450 million; April 1998, engines for six A330-200, US$150 million. Trent 800 series, November 1997, combined orders for 133 777s, US$3.5 billion, including Trent 892 for 11 777s, US$290 million; April 1998, for ten 777s, US$260 million, and for two 777s, US$51.9 million; August 1998, for 15 777 US$390 million; September 1998, for 32 777, US$930 million; July 2000, for three 777, US$90 million; Trent 500 series, May 1998, for 10 A340-500, US$400 million; September 1998, for 16 A340-500, US$640 million; total at June 2000 for 124 A340-500/600, US$5.8 billion.

By September 1998, orders and options for the Trent significantly exceeded 1,200, ``worth approximately US$14 billion''. Risk- and revenue-sharing partners across the Trent family include BMW Rolls-Royce, Hispano-Suiza, Ishikawajima Harima Heavy Industries (IHI), Industria de Turbo Propulsores (ITP), Kawasaki Heavy Industries (KHI), LucasVarity, GE Celma, Denel Aviation, Marubeni, Fokker Elmo and, for the Trent 500, FiatAvio and Hamilton Standard. Together these companies have from 20 to 24 per cent participation in each programme.

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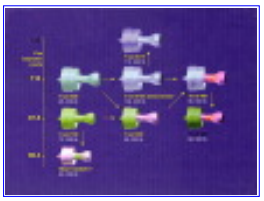
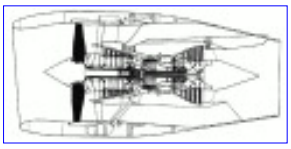
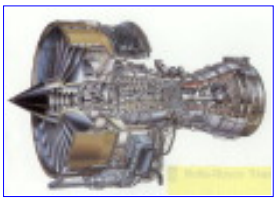


The family of large Rolls-Royce turbofans (1999)

Trent 700



Trent 700 nacelle cross-section (2001)

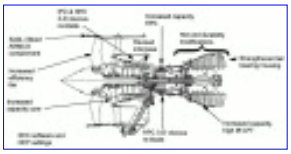


Cutaway drawing of Trent 700

Trent 800

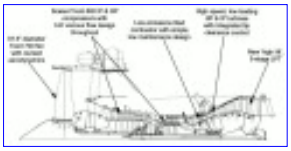
Cutaway drawing of Trent 800

Trent 800 cross-section (2001)



Trent 8104 changes relative to Trent 892 (1998)

Trent 8104 (2000)



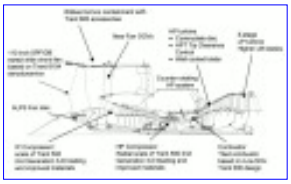
Features of the Trent 500 (1998)

About to mate the fan to the first Trent 500 (2000)

The first Trent 500 on test (2000)

Cutaway drawing of Trent 500 (1999)

Cutaway drawing of Trent 900 (2002)



Features of the Trent 900 (2000)

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AERO-ENGINES - TURBOFAN, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE 535

This engine is also often referred to as the RB211-535. It was derived from earlier versions of the RB211.

535C

This was the launch engine for the Boeing 757. It has an HP module based on the RB211-22B, six-stage IP compressor without variable stator vanes, and a scaled-down -524 fan. Fan air flow is 18 per cent lower than that of the -22B, and core air flow 12 per cent lower. The 535C powered the first 757 on its first flight on 19 February 1982 and entered service with Eastern on 1 January 1983.

535E4

Advanced version offering increased thrust, together with reduced fuel consumption relative to the 535C. The E4 allowed 757s powered by it to be cleared for 120-minute ETOPS in December 1986, extended to 180 minutes in 1990.

535E4B

Mechanically similar to E4 but uprated, enabling the 757 to carry heavier payloads from noise-sensitive airports. EIS August 1989; cleared for 180-minute ETOPS. With minor changes, this engine was selected in 1991 as an alternative power plant for the Tupolev Tu-204. The sixth aircraft of this type was completed as a Tu-204-120, with the British engines, and began its flight testing on 14 August 1992. This programme was

launched by Bermuda-based Sirocco Aerospace International, owned by Kato Aromatic of Cairo and Cairo Bank. Kato itself ordered 13 aircraft, with 17 on option. The first engine shipset for a production Tu-204-120 was delivered to the Aviastar factory at Ulyanovsk in July 1996. Aircraft certification by ARMAK was achieved in January 1997. The Tu-204-120 is now equipped with an English-language cockpit, and is expected to complete JAA certification in late 2002. The launch customer is Egypt's Air Cairo. Two Chinese operators have selected the Tu-204-120C cargo version, with an order for five firm and 10 option aircraft.

The Rolls-Royce engine has been selected by over 80 per cent of 757 customers, and the market share continues to increase. Over 1,200 engines have been sold to power more than 550 aircraft. The E4B has been selected by 86 per cent of customers for the stretched 757-300, the first of which was delivered in March 1999. The 535E4B has built a reputation for industry-leading reliability, with the world's lowest combination of SVR and IFSD. An engine of Icelandair holds the world record of over 40,000 hours on-wing, 81 engines have exceeded 9,000 cycles between shop visits, 50 per cent of engines have exceeded 11,000 hours between shop visits, and over 400 engines have set first-run lives in excess of 15,000 hours. Total flight time by July 2001 was 28.4 million hours.

Over the past 20 years it has been suggested that the 535E4 with minor changes could replace the pairs of existing engines in the USAF's Boeing B-52 bombers. Boeing resurrected the proposal in July 1996, calculating that over the planned remaining life of these aircraft such a change would save nearly US$2 billion, whilst improving reliability. This proposal was dealt a blow by the decision of the British MoD and RAF to lease Boeing C-17 aircraft without switching to the 535E4.

Type

Three-shaft turbofan.

Fan

Single stage, with rotating conical spinner but without inlet guide vanes. 535C has 33 titanium blades with part-span snubbers. Diameter 1,859 mm (73.2 in). Mass flow 518 kg (1,142 lb)/s. Bypass ratio 4.4. 535E4 and E4B have 22 wide-chord blades without the need for snubbers, with activated diffusion-bonded titanium skins on a titanium honeycomb core. Fan case of Rohrbond with Kevlar containment. Diameter 1,892 mm (74.5 in). Mass flow 522 kg (1,150 lb)/s. Bypass ratio 4.3.

IP Compressor

Six stages of controlled diffusion design. No variable vanes.

HP Compressor

Six stages of end-bend blading, with stage 4, 5 and 6 discs in titanium superalloy. Low-expansion casing for improved tip clearance control. Overall engine pressure ratio (535C) 21.1, (535E4) 25.8, (535E4B) 27.1.

Combustion Chamber

Annular, 18 airspray nozzles (24 in low-NO_x Phase 5 combustor, which was certificated in August 1998 and entered service in 1999), flexible tiled liner mountings, heatshields and thermal barrier coatings. See diagram in preliminary feature Civil Engines.

HP Turbine

Single stage. Rotor blades, directionally solidified, cast with HP leading-edge cooling, HP and LP internal air cooling passages both with triple pass system. Nozzle guide vanes with curved stacking and advanced cooling.

IP Turbine

Single stage. Cooled NGVs with multilean stacking for improved air flow onto high-aspect ratio blades.

LP Turbine

Three stages. All turbine casings double wall and cooled.

Jetpipe

Core and bypass flows mixed in integrated nozzle.

Reverser

Fan reverser only. Jacks move translating cowl to rear, blocker deals seal fan duct and uncover cascade vanes. Over- expansion reduces core thrust.

Accessories

Mounted on gearbox under fan case, driven from HP spool. The gearbox is now produced by Denel Aviation of South Africa, which delivered its 475th unit in September 2001.

Dimensions

Length: 535C	3,010 mm (118.5 in)
535E4	2,995 mm (117.9 in)
Fan diameter: 535C	1,859 mm (73.2 in)
535E4	1,882 mm (74.1 in)

Weight, Dry

535C	3,309 kg (7,294 lb)
535E4, E4B	3,295 kg (7,264 lb)

Performance Ratings

(note: flexible T-O ratings involving considerable derating are used in operation)

T-O (S/L, ISA): 535C	166.4 kN (37,400 lb st)
535E4	178.4 kN (40,100 lb st)
535E4B	192.0 kN (43,100 lb st)
Max climb (10,670 m; 35,000 ft, M0.80):	
535C	40.1 kN (9,023 lb)
535E4	41.4 kN (9,300 lb)
Max cruise (10,670 m; 35,000 ft, M0.80):	
535C	37.6 kN (8,453 lb)
535E4	38.7 kN (8,700 lb)

As above, installed, ISA + 10ºC:	
535C	35.9 kN (8,064 lb)
535E4	37.1 kN (8,345 lb)

Specific Fuel Consumption

(cruise, as above)

535C	18.30 mg/Ns (0.646 lb/h/lb)

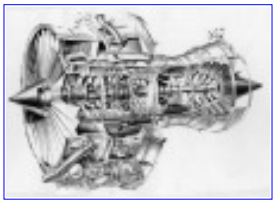
535E4	16.94 mg/Ns (0.598 lb/h/lb)

Contract Price

August 1996, five shipsets for 757, US$95 million, 30 shipsets for Tu-204-220, US$450 million. April 1997, 16 shipsets for 757 with eight-year support US$240 million; September 1997, engines for five 757, US$75 million; September 1998, engines for two 757-200 plus one spare engine, US$36 million, and for two firm, two option, 757-300, up to US$60 million.

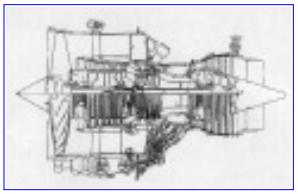
UPDATED

Rolls-Royce 535C

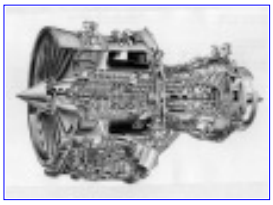


Cutaway drawing of Rolls-Royce 535C

Rolls-Royce 535E4



Longitudinal section and cutaway of Rolls-Royce 535E4



Cutaway drawing of Rolls-Royce 535E4

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AERO-ENGINES - TURBOFAN, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

V2500

Rolls-Royce has a 30 per cent share in this civil turbofan. It is described under IAE in the International section.

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AERO-ENGINES - TURBOFAN, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

TF41

This engine, nearing the end of its service life, was produced jointly with Allison, which is today Rolls-Royce Corporation. It is described under that company in the International section.

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AERO-ENGINES - TURBOSHAFT, UNITED KINGDOM

Date Posted: 11 January 2002

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ROLLS-ROYCE plc

RTM322

Rolls-Royce has a 47 per cent share in this three-nation turboshaft. Details are given under Rolls-Royce Turbomeca in the International section.

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ROLLS-ROYCE plc

RB199

Rolls-Royce has a 40 per cent share in this military turbofan. Details are given under Turbo-Union in the International section.

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ROLLS-ROYCE plc

Olympus

The Concorde engine is described under Rolls-Royce SNECMA in the International section.

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ROLLS-ROYCE plc

MTR390

Rolls-Royce has a 20 per cent share in this three-nation turboshaft. Details are given under MTR in the International section.

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Date Posted: 11 January 2002

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ROLLS-ROYCE plc

JSF

Rolls-Royce, especially Rolls-Royce Corporation, is a major partner in the propulsion of the Lockheed Martin X-35, as outlined under JSF in the International section. The largest single contributions are the STOVL version's shaft-driven fan and three-bearing nozzle.

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ROLLS-ROYCE plc

EUROJET TURBO EJ200

Rolls-Royce has a 33 per cent share in this military turbofan. It is described under Eurojet Turbo in the International section.

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Date Posted: 11 January 2002

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ROLLS-ROYCE plc

BR700 series

Rolls-Royce produces these engines at a plant near Berlin. Details are given under Rolls-Royce Deutschland in the German section.

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ROLLS-ROYCE plc

Adour

Rolls-Royce has a 50 per cent share in this military turbofan, which remains in production. Details are given under Rolls-Royce Turbomeca in the International section.

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Rolls-Royce plc employs 43,700 people worldwide, with more than half in the aerospace businesses which serve airline, corporate and regional airline, helicopter and international defence customers. It has around 56,000 engines in service with more than 500 airlines, 2,400 corporate and utility operators and more than 100 armed forces.

As in 2000, more than 80 per cent of sales were to customers outside the UK. In calendar year 2000 more than 1,000 civil engines were delivered. This is almost three times as many as the level in the mid-1990s. Overall, the order book (backlog of unfulfilled signed contracts) increased to £13.1 billion.

In the calendar year 2000 turnover (sales) increased to £5,864 million, underlying profit before taxation increased to £436 million. Civil-engine sales in 2000 totalled £3,150 million, a 31 per cent share of the global market, while military sales amounted to £1,403 million. The latter figure is expected to grow significantly, not least because of the Eurofighter and JSF programmes.

In 1995, Rolls-Royce acquired the Allison Engine Company, whose civil and military programmes are listed under Rolls-Royce Corporation in the USA section. These are now incorporated into the Rolls-Royce product range with engines available for more than 50 civil and military fixed-wing and rotary-wing aircraft programmes.

The former BMW Rolls-Royce will be found in the German section as Rolls-Royce Deutschland GmbH.

Rolls-Royce has a number of partnerships, joint ventures and collaborative programmes. These include International Aero Engines, Eurojet, Turbo-Union, Rolls-Royce Turbomeca and MTR GmbH. It also has a number of risk- and revenue-sharing partners on civil programmes, mainly the Trent engine. Other partnerships are expanding the repair business, major additions in 1998 being TAESL with American Airlines at Fort Worth and IECO with Singapore Airlines Engineering.

In 2001 Rolls-Royce launched Total Care, ``an innovative power management system tailored to the requirements of each customer, providing them with through-life support for their engines''. It provides a 24-hour online service, called aeromanager.com, underpinned by the company's global overhaul and repair network established at 16 locations on four continents. Already, in 1999-2001 Rolls-Royce doubled its share of civil-engine support to more than 50 per cent worldwide. Total Care includes arrangements for predictive maintenance, power-by-the-hour contracts and options such as engine leasing and asset management. Total Care agreements signed in 2001 included one for US$1 billion to support the RR535 engines of the Boeing 757s of American Airlines, and one for US$100 million covering the AE3007 engines of South African Airlink's Embraer fleet.

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AERO-ENGINES - TURBOJET, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE VIPER

This turbojet was originally designed by Armstrong Siddeley in 1950 as a cheap expendable engine for missiles and targets. It first ran at a rating of 7.29 kN (1,640 lb st), in April 1951, by which time Chief Engineer John Marlow was working on versions for manned applications. Astonishingly, small numbers were still being made in 2000. Production exceeds 5,600, with flight time of 15 million hours.

Current versions are:

Viper 8 (Mk 100 Series)

Small numbers of Mk 102 engines are flying in the Jet Provost T.3 and the Mk 104 in the Jindivik 102B, in each case rated at 7.78 kN (1,750 lb st). Single-stage turbine drives seven-stage compressor with mass flow 14.06 kg (31.0 lb)/s, pressure ratio 3.8.

Viper 11 (Mk 200 series)

Mass flow increased to 20 kg (44 lb)/s. Mass flow 20 kg (44 lb)/s. Pressure ratio 4.8. Powers Jindivik Mk 3 drone, Jet Provost T.4 and 5, SOKO Galeb and HJT-16 Kiran Mk I/IA. Viper 202s from Jet Provosts have been converted to Mk 210C to power 18 new Jindivik targets for the UK Ministry of Defence from 1998. A Viper 11 version, the 22-1, was built under licence in Italy by Piaggio and HDH of Australia for the Aermacchi MB-326.

Viper 20 (Mk 500 Series)

Development with zero stage on compressor. Mass flow 23.9 kg (52.7 lb)/s. Pressure ratio 5.6. Major applications include early HS 125 (Mks 521, 522) and PD-808 executive aircraft (Mk 526), and BAe 167 Strikemaster (Mk 535), MB-326 (Mk 540) and Jastreb (Mk 531) training and light combat aircraft. Mk 540 built under licence by Piaggio in Italy and by Atlas in South Africa under Piaggio sub-licence.

Viper 600 Series

All current production Vipers are of this type, which introduced an eight-stage compressor handling a considerably increased air flow, driven by a two-stage turbine. The basic civil Viper 601 powers the Raytheon Hawker 125-600. The 632-41, made by Turbomecanica of Romania and Orao of former Yugoslavia, powers the IAR-93A and IJ/INJ-22. The 632-43, made by Piaggio, powers the MB-339. The 632-46, formerly made by Orao, powers the G-4 Super Galeb. The 632-41R, 633-44 and 633-47 all have an afterburner of the two-gutter type, with hot-streak ignition and a multiflap variable nozzle. These engines were made by Turbomecanica and Orao to power the IAR-93B and J-22. The 680-43, giving increased dry thrust, powers the MB-339C and MB-339FD.

The following details apply to the Viper 600 Series:

Type

Single-shaft axial turbojet.

Compressor

Eight-stage. Steel drum type rotor with disc assemblies. Magnesium alloy casing with blow-off valve. Stator blades mounted in carrier rings slotted into casing. All stator blades and 1st, 2nd and 8th stage rotor blades of steel; remainder aluminium alloy, Zero-stage and first stage rotor blades attached by fir-tree roots; stages 3-8 riveted. Mass flow (632) 26.5 kg (58.4 lb)/s, (680) 27.2 kg (60.0 lb)/s. Pressure ratio (601) 5.8, (632, 633) 5.9, (680) 6.8.

Combustion Chamber

Short annular type with 24 vaporising burners and six starting atomisers. Electric ignition from two high-energy plugs with separate supplies.

Turbine

Two-stage axial. Shrouded blades attached to discs by fir-tree roots and locking strips. Discs cooled by compressor bleed. Entry gas temperature (632-43) 967ºC.

Accessories

Gearbox driven from front of compressor by bevel gear. Pads for starter/generator, tachometer generator, fuel system and oil pump.

Starting

24 V starter/generator.

Control System

Hydromechanical, with variable-stroke pump, barometric control and air/fuel control.

Oil System

Self-contained recirculatory system. Military version fully aerobatic.

Dimensions

Length (flange to flange):	
Viper 531, 535, 632, 680	1,806 mm (71.1 in)
Viper 633	4,440 mm (174.8 in)
Inlet diameter	490 mm (19.3 in)
Max width	749.3 mm (29.5 in)
Height	901.7 mm (35.5 in)

Weight, Dry

Viper 11	281 kg (620 lb)
Viper 531, 535, 540	358 kg (790 lb)
Viper 601, 632	376.5 kg (830 lb)
Viper 633	421 kg (928 lb)
Viper 680	379 kg (836 lb)

Performance Ratings (T-O)

Viper 11	11.12 kN (2,500 lb st)
Viper 531	13.9 kN (3,120 lb st)
Viper 535, 540	14.9 kN (3,360 lb st)
Viper 601	16.7 kN (3,750 lb st)
Viper 632	17.66 kN (3,970 lb st)
Viper 633 (afterburner)	22.38 kN (5,030 lb st)
Viper 680	19.13 kN (4,300 lb st)

Specific Fuel Consumption

(T-O)

Viper 11	30.3 mg/Ns (1.07 lb/h/lb st)
Viper 500 series	28.3 mg/Ns (1.00 lb/h/lb st)
Viper 601	26.9 mg/Ns (0.95 lb/h/lb st)
Viper 632	27.5 mg/Ns (0.97 lb/h/lb st)
Viper 633	49.05 mg/Ns (1.73 lb/h/lb st)
Viper 680	27.8 mg/Ns (0.98 lb/h/lb st)

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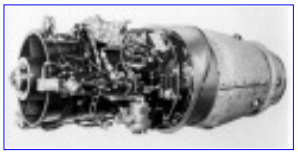
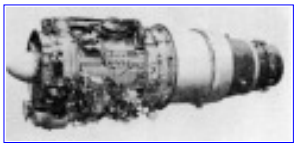


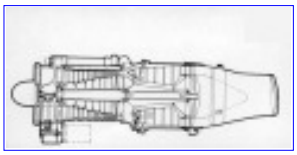
Viper 200 series



Viper 522 series

Cutaway drawing of Viper 680

Viper 680



Longitudinal section through Viper 500 series

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AERO-ENGINES - TURBOPROP, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE TYNE

Originally known as the RB109, the Tyne is a twin-spool turboprop which, in its day, set new standards of pressure ratio and fuel economy. The engine first ran in April 1955, and the Mk 506 (RTy.1) entered service at 3,568 ekW (4,785 ehp) in 1961 as the engine of the V951 Vanguard. The following are current versions:

Mk 515/10 (RTy.12)

Powers CL-44D-4 and CL-44J.

Mk 515/101W (RTy.12)

Water injection. Powers Belfast.

Mk 21 (RTy.20)

Powers Atlantic and Atlantique. Assembled by SNECMA and MTU under licence.

Mk 22 (RTy.20)

Powers Transall C-160. Assembled by SNECMA and MTU under licence.

Mk 801 (RTy.20)

Powers Aeritalia G222. Produced in partnership with the European consortium.

About 1,000 engines remain in aircraft service. The Tyne is also important in warship and hydrofoil propulsion and electricity generation.

The following details refer specifically to the Tyne RTy.20, but are generally applicable to all versions:

Type

Two-spool axial-flow turboprop.

Intake

Annular intake surrounds reduction gear housing. Integrally cast in magnesium alloy, with seven hollow support struts. Oil tank of annular form made up by rear wall of air intake casing and fabricated steel shell. Anti-icing by hot oil circulated through struts and by hot air tapped from HP compressor. Electrical de-icing of cowling surrounding intake.

LP Compressor

Six-stage axial-flow type. Made up from six steel discs, of which first (stage 0) disc is integral with shaft and remaining five discs splined to shaft. The 216 light alloy rotor blades are unshrouded and fixed to discs by single pin fixing. Inlet guide vanes and stage 0 stator blades of fabricated hollow construction to provide de-icing by means of air bled off HP compressor. The 431 aluminium-alloy stator blades are secured in casing by tongue and groove location. Steel LP casing in one piece. LP compressor mounted on front roller bearing and rear roller bearing. Intermediate casing between HP and LP compressors is aluminium alloy casting housing internal wheelcase. Bleed valve mounted on top of casing operates under approach conditions when LP and HP speeds are unmatched.

HP Compressor

Nine-stage axial-flow type. Made up from nine steel discs: stages 1 and 2 attached to front shaft by bolts, remainder splined to rear shaft. Total of 575 rotor blades of titanium (first seven stages) and steel (last two stages). The 734 steel stator blades are fixed in rings by tongue and groove location. Stator blades are unshrouded. HP casing of centri-cast steel supports stator drum by bolted flanges. HP compressor mounted on front roller bearing and rear ball bearing. Rear bearing also takes thrust from intershaft ball bearing mounted slightly ahead of it. Pressure ratio 13.97:1. Air mass flow 21.1 kg (46.5 lb)/s.

Combustion Chamber

Ten flame tubes of Nimonic sheet mounted within annular chamber. Combustion system casings of steel. Flame tubes contain double twin-flow coaxial burners. Flame tubes 3 and 8 (on engine horizontal centreline) contain high-energy igniter plugs.

HP Turbine

Single-stage. Steel disc attached to HP shaft by ten taper bolts. 121 Nimonic blades attached by fir-tree roots. Blades air-cooled and tip-shrouded. Centri-cast steel casing, containing four stators of nozzle guide vanes. The 50 HP vanes are air-cooled. HP turbine carried on roller bearing ahead of turbine. Shaft splined to HP compressor shaft. TGT 1,000ºC.

LP Turbine

Three-stage. Steel discs bolted to each other and to forward LP shaft by ten bolts. Stage 3 disc integral with rear LP shaft. All blades of Nimonic, secured by fir-tree roots. Stage 1 has 101 tip-shrouded blades. Stage 2

has 106 tip-shrouded blades. Stage 3 has 61 tip-shrouded blades. The three LP stators have 60 hollow blades each. 20 thermocouples are fitted in the leading-edges of the first-stage LP nozzle guide vanes. LP turbine is carried on roller bearings at rear and supported by plain bearing in HP shaft at front. Shaft splined to LP compressor shaft. Gas temperature after turbine 453ºC.

Jetpipe

Fabricated construction, supporting tail bearing from nozzle box outer casing by 10 struts with streamlined fairings.

Mounting

Four mounting feet on engine vertical and horizontal centrelines, located immediately aft of oil tank.

Output

Double reduction gearing by compound epicyclic train. High-speed pinion driven from forward end of LP shaft with final drive through planet wheel carrier integral with propeller shaft. Ratio 0.064:1. Shaft carried in one set of ball bearings and one set of roller bearings. Fixed annulus. Propeller diameter: Mk 512, 4.42 m (14 ft 6 in); Mks 515, 801, 21, 4.88 m (16 ft 0 in); Mk 22 5.49 m (18 ft 0 in).

Accessories

Internal wheelcase houses two drives. Port wheelcase drive from LP shaft, to accessory gearbox, LP tachometer and LP shaft governor. Starboard wheelcase drive from HP shaft to fuel pumps, HP tachometer, oil pumps and breather, mounting for starter motor.

Starting

Mk 515/101W, electric; other versions, air starter located on HP wheelcase.

Control System

Single multiplunger variable-stroke HP pump delivers fuel to burners via flow control unit. Unit incorporates filter, throttle valves, shut-off cock and barometric pressure sensing device. Operation is a function of air intake pressure and throttle valve pressure drop. Hydromechanical governors control overspeeding of LP and HP sections of engine. Fuel anti-icing is by an oil-heated fuel heater between tank and LP pump. Max fuel supply pressure 87.88 kg/cm^2 (1,250 lb/sq in). Water/methanol injected into LP compressor through holes in stage 0 disc.

Fuel Specification

MIL-J-5624D, Grade JP-4.

Oil System

Dry sump type with one pressure and six scavenge spur-gear pumps driven from HP shaft. Tank capacity 26.2 litres (6.9 US gallons, 5.76 Imp gallons) including 9.1 litres (19.2 US pints, 16 Imp pints) for feathering reserve. Thermostatically controlled air-cooled oil cooler. Oil supply pressure 3.16 kg/cm^2 (45 lb/sq in).

Oil Specification

DERD.2487.

Dimensions

Length overall	2,762 mm (108.72 in)
Max diameter	1,400 mm (55.0 in)

Weight, Dry

Mk 515	995 kg (2,193 lb)
Mk 21	1,098 kg (2,420 lb)
Mk 22	1,105 kg (2,436 lb)

Performance Ratings

(T-O, S/L, ISA)

Mk 512	4,135 ekW (5,545 ehp)
Mk 515	4,273 ekW (5,730 ehp)
Mks 21, 22	4,552 ekW (6,105 ehp)
Mk 801	4,086 ekW (5,480 ehp)

Specific Fuel Consumption

(T-O as above)

Mks 21, 22	81.9 µg/J (0.485 lb/h/ehp)

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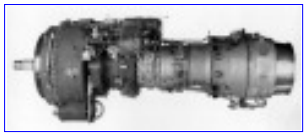
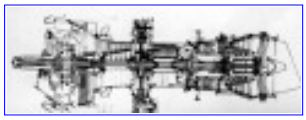
Tyne Mk 22



RB109 prototype

Cutaway drawing of Tyne Mk 21



Longitudinal section through Tyne Mk 21

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AERO-ENGINES - TURBOFAN, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE TAY

The Tay turbofan is designed around the core and external gearbox of the Spey Mk 555. The LP system has been tailored to complement this by maintaining core inlet and outlet conditions similar to those of the original engine. The wide-chord fan and three-stage IP compressor are driven by a three-stage LP turbine which uses the latest proven technology. The cold bypass air and hot exhaust are combined in a forced mixer. The bypass duct is carbon fibre composite. The engine is completely modular.

The first engine (then called Mk 610-8) went on test in August 1984. The initial versions received certification from the CAA in June 1986. By September 1999, over 1,700 Tay engines had accumulated 12 million hours in service. Certification of the Tay 650 was achieved ahead of schedule in June 1988.

The Tay easily meets all current emission standards, and enables aircraft to comply with FAR Pt 36 Stage 3 noise requirements with large margins. In 13 years of operation the Tay has been outstandingly reliable.

Tay 611

Selected for Gulfstream IV. EIS June 1987. In March 1998, Gulfstream placed a US$700 million contract for engines for GIV SP aircraft through 2003.

Tay 620-15

Selected for Fokker 70 and 100. EIS Fokker 70, October 1994; Fokker 100, April 1988.

Tay 650-15

Small increase in fan diameter and advanced HP turbine, giving 9 per cent increase in T-O rating and 15 per cent

more climb, max cont and cruise. Specified for higher performance versions of Fokker 100. EIS October 1989.

Tay 651-54

Selected to re-engine 727-100. Similar to Tay 650 except small changes to external gearbox and accessories. EIS December 1992. By October 1999, 52 aircraft had been re-engined.

Type

Two-shaft turbofan.

Fan

Single-stage with wide-chord blades. Mass flow (611, 620) 186 kg (410 lb)/s, (650) 189.6 kg (418 lb)/s, (651) 193.2 kg (426 lb)/s. Bypass ratio (611, 620) 3.04, (650) 3.06, (651) 3.07.

LP Compressor

New design with three stages on fan shaft, with curvic coupling. Titanium and steel drum, titanium blades, steel stator vanes.

HP Compressor

Twelve-stage axial (as Spey Mk 555). Steel construction. Overall engine pressure ratio: (611, 620) 15.8, (650) 16.2, (651) 16.6.

Combustion Chamber

Tubo-annular with 10 flame tubes, each with one burner. Flame tubes made of Transply porous sandwich fabricated from chem-milled sheets brazed together. Advanced combuster EIS 1995.

HP Turbine

All Mks except 650 and 651, two stages as Spey Mk 555. Tay 650 and 651, advanced two-stage design. TET: (611, 620) 1,041ºC, (650) 1,127ºC, (651) 1,140ºC.

LP Turbine

New design with three stages.

Jetpipe

Forced deep chute mixer with 12 deep lobes. Carbon fibre composite bypass duct.

Reverser

(611, 620, 650) Grumman target type, (651) Dee Howard.

Accessories

Radial drive from front of HP shaft to gearbox on left underside of bypass casing. In January 2001 a contract was signed with Denel Airmotive of South Africa for future gearboxes. Prospective contract value was announced as R30 million (US$4.2 million) per year.

Control System

As Spey Mk 555 but with improved fuel control unit.

Dimensions

Length: Tay 611, 620, 650, 651	2,405 mm (94.7 in)

Fan diameter: Tay 611, 620	1,118 mm (44.0 in)
Tay 650, 651	1,138 mm (44.8 in)

Weight, Dry

Tay 611	1,422 kg (3,135 lb)
Tay 620	1,445 kg (3,185 lb)
Tay 650	1,515 kg (3,340 lb)
Tay 651	1,533 kg (3,380 lb)

Performance Rating

(T-O)

Tay 611, 620-15	61.61 kN (13,850 lb st) to 30ºC
Tay 650	67.17 kN (15,100 lb st) to 30ºC
Tay 651	68.51 kN (15,400 lb st) to 28ºC

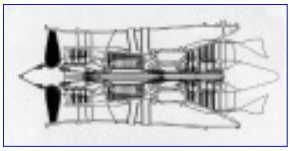
Specific Fuel Consumption

Cruise:	
Tay 611 (13,100 m; 43,000 ft, M0.8)	20.1 mg/Ns (0.71 lb/h/lb)
Tay 620 and 650 (9,145 m; 30,000 ft, M0.73)	19.5 mg/Ns (0.69 lb/h/lb)
Tay 651 (10,669 m; 35,000 ft, M0.78)	19.5 mg/Ns (0.69 lb/h/lb)

VERIFIED

Tay 620-15

Cutaway drawing of typical Tay



Longitudinal section through Tay 650

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AERO-ENGINES - TURBOFAN, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE SPEY

Design of the RB163 Spey began in September 1959, and the first engine ran at the end of December 1960. Flight testing of two Speys in a Vulcan began on 12 October 1961, and prototype flight trials of the Spey-engined Hawker Siddeley Trident began on 9 January 1962. Including the RB183, the Spey Mk 555 (originally called the Spey Junior), over 2,600 civil Speys were built, while production of military engines in the United Kingdom exceeded 2,800. Total flight time in mid-2001 was 9 million hours military and 56 million civil. Military Speys are licensed for manufacture in Italy, Brazil, the USA and China.

Several organisations are currently offering or developing Stage 3 hushkits for Speys fitted to One-Eleven and Gulfstream aircraft.

Military versions:

Mk 101

The Mk 101 powered the Buccaneer, a restored example of which was in 2001 about to begin a civil career at Thunder City in South Africa. This is the only version apart from the Mk 555 and Mk 807 to have a four-stage LP compressor. Thrust rating is 49.06 kN (11,030 lb st).

Mks 202/203

Stressed for flight to M2.5, with robust shaft/disc LP compressor and large afterburner and variable nozzle. Plessey gas-turbine starter. Powered Phantom FG.1 (one still used for calibration at Boscombe Down) and FGA 2. Produced under licence 1975-79 by Xian Aero-Engine Corporation in China as WS9. Max ratings (dry) 54.5

kN (12,250 lb st), (with afterburner) 91.25 kN (20,515 lb st).

Mks 250/251

Marinised versions of civil Mk 512, driving large alternators for Nimrod. T-O rating 53.35 kN (11,995 lb st). The number of Nimrod MR.1 aircraft delivered to the RAF was 46. Of these 35 were upgraded to MR.2 and in mid-2000 a total of 26 was regarded as being `in service'. Of these, 21 are being remanufactured to bring them up to MRA.4 standard, which among other things replaces the Spey engines by the BR710 Mk.101. Despite this, in May 2000, the UK Ministry of Defence signed a contract valued at more than £50 million for the support of the Spey 250/251 in RAF Nimrods. This is to run for *12 years*. This was described as ``smart procurement in action''. Spey support is provided by Rolls-Royce East Kilbride.

Mk 807

Basically matches Mk 101 rotors within the structure of the Mk 511. Produced for the AMX under licence in Italy (prime contractor FiatAvio, assisted by Alfa Romeo and Piaggio) and Brazil (prime contractor CELMA). T-O rating 49.06 kN (11,030 lb st).

Mk 807A

Mk 807 uprated to the 57.8 kN (13,000 lb st) class as candidate engine for future variants of AMX, but in June 1999 it was reported that any future AMX would be more likely to have a different engine (candidates said to be unaugmented EJ200 or F414).

Civil versions:

Mk 511-5, 511-8 and 511-14

T-O rating of 50.7 kN (11,400 lb) at 12,390 rpm. Mk 511-8 for Gulfstream II and III, and Mk 511-14 for One-Eleven.

Mk 511-5W and 511-14W

As 511-5 and 511-14 but with water injection to maintain rating to 35ºC.

Mk 512-5 and 512-14

T-O rating 53.37 kN (12,000 lb st) at 12,390 rpm in Mk 512-14 for One-Eleven.

Mk 512-14DW

T-O rating 55.82 kN (12,550 lb st). Similar to basic Mk 512 but with T-O rating increased by raising limiting compressor delivery pressure at T-O, with turbine entry temperature held at existing level by water injection.

Mk 555

Simplified engine for F28 Fellowship. Four-stage LP compressor, Mk 555-15N and -15P fitted with 10-lobe mixer nozzle. T-O rating 44.03 kN (9,900 lb st) to 29.7ºC. Also known as RB183.

Type

Two-spool axial flow turbofan.

Intake

Annular type at front of engine, with bleed-air thermal anti-icing.

Compressor

Axial flow two-spool type, with five-stage (four-stage on Mks 101 and 555) low-pressure (LP) and 12-stage high-pressure (HP) compressors. First-stage HP stator vanes are of variable-incidence type. LP compressor is of the drum type, built of aluminium alloy and pinned to shaft. HP compressor is of the disc type, built of aluminium alloy and with first stage bolted to shaft, remaining stages splined to shaft. All stator blades are of aluminium. LP compressor has aluminium blades. HP compressor has blades of aluminium, steel and titanium. Stator blades are slotted into casing. Rotor blades are attached by pins or dovetailed slots. LP compressor has one-piece aluminium alloy casing. HP compressor has two-piece steel casing. Pressure ratio (101) 16.5, (512, 250) 21.0, (555) 15.4. Mass flow (101) 91.63 kg (202 lb)/s, (512, 250) 94.35 kg (208 lb)/s, (555) 90.27 kg (199 lb)/s, (807) 91.6 kg (202 lb)/s. Bypass ratio (typical) 0.71, (807) 1.0.

Combustion Chamber

Can-annular type of steel with 10 Nimonic sheet flame tubes. Duplex downstream burners, one per chamber. High-energy igniters in chambers 4 and 8.

HP Turbine

Two-stage axial flow, first HP stage is air-cooled. Turbine discs of creep-resisting ferritic steel, bolted to shaft. Nickel-based alloy blades are attached by fir-tree roots.

LP Turbine

Two-stage.

Jetpipe

Fixed-area type with outer wall and cone of stainless steel sheet. Civil engines available with various noise-suppressing nozzles to meet Chapter 3 (see Environmental Factors). The latest such offering is that of Quiet Technology Gulfstream, of Opa Locka, Florida. This is based on a One-Eleven nozzle, and should be certified in first-quarter 2002.

Reverser

Internal clamshell type reverser on Mk 250/251 and civil engines.

Mounting

Two trunnions, two saddle mountings and one rear mounting.

Accessories

Two accessory drives. Port gearbox, driven from LP rotor, carries LP governor and LP tacho. Starboard gearbox, driven from HP rotor, carries LP and HP fuel pumps, fuel regulator, main oil pumps, air flow control rpm signal transmitter, starter and HP tacho. Provision also made (for starboard gearbox) for aircraft ancillaries such as hydraulic pump, constant-speed drive and generator.

Starting

In most civil engines, Rotax CT1013 air turbine starter.

Control System

Plessey LP fuel pump, feeding through fuel-cooled oil cooler and Marston Excelsior fuel heater to LP fuel filter at inlet, to Lucas GD-type fuel pump. High-pressure fuel metered by Lucas fuel regulator unit, embodying

combined speed and acceleration control and fed through Lucas LP governor and shut-off valve to Duple fuel spray nozzles. Maximum fuel pressure 126 kg/cm^2 (1,800 lb/sq in). Water injection (applicable to all engines bearing `W' suffix): Water supplied by aircraft-mounted Lucas turbopump, fed by air from two or three engines (dependent on aircraft) through engine-mounted automatic shut-off valve to injector passages in fuel spray nozzles. Water sprays into primary air flow through flame tube swirlers. Fuel regulator unit incorporates automatic reset, to increase fuel flow and so restore flame temperature reduced by injection of water.

Fuel Specification

DERD 2482 or 2486.

Oil System

Self-contained continuous circulation system. Single pressure pump draws oil from tank, feeds it through fuel-cooled cooler and HP filter to gearboxes and shaft bearings. Front bearing has supplementary pressure and scavenge pumps in nose bullet to ensure positive feed at start-up. Five main scavenge pumps, in casing with pressure pump, return oil from all bearing housings and gearboxes to tank. Total system capacity 13.6 litres (3.6 US gallons, 3.0 Imp gallons). Usable oil 5.1 litres (9 Imp pints). Normal oil supply pressure 2.5 to 3.5 kg/cm^2 (35 to 50 lb/sq in).

Oil Specification

DERD 2487.

Dimensions

Length, less tailpipe:	
Mks 101, 512	2,911 mm (114.6 in)
Mk 202, 203	5,204 mm (204.9 in)
Mk 250, 251	2,972 mm (117.0 in)
Mk 555	2,780 mm (109.4 in)
Mk 807	2,456 mm (96.7 in)
Diameter (basic)	825 mm (32.5 in)

Weight, Dry

Mk 101	1,121 kg (2,471 lb)
Mk 202, 203	1,857 kg (4,093 lb)
Mk 250, 251	1,243 kg (2,740 lb)
Mk 512	1,168 kg (2,575 lb)
Mk 555	1,024 kg (2,257 lb)
Mk 807	1,096 kg (2,417 lb)

Performance Ratings

(T-O) see list of variants

Specific Fuel Consumption

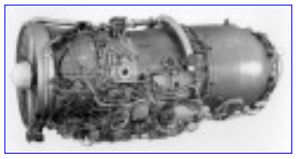
(T-O rating)

Mk 101	16.18 mg/Ns (0.57 lb/h/lb)
Mk 202, 203 (afterburner)	52.39 mg/Ns (1.85 lb/h/lb)

Mk 250, 251	16.53 mg/Ns (0.58 lb/h/lb)
Mk 512	17.33 mg/Ns (0.612 lb/h/lb)
Mks 555, 807	15.85 mg/Ns (0.56 lb/h/lb)

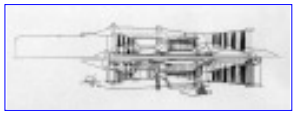
UPDATED

Spey 511



Spey 555

Cutaway drawing of Spey 807



Longitudinal sections of Spey 202 without afterburner (upper) and Mk 512 (lower)

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AERO-ENGINES - TURBOFAN, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE RB211

The designation RB211 applies to a family of three-shaft turbofans of high bypass ratio and high pressure ratio, ranging in thrust from 166.4 kN (37,400 lb st) to over 266.9 kN (60,000 lb st). Over 2,300 engines are in service, logging over 84 million hours. The derived 535 and Trent are described separately. For all applications Rolls-Royce retains responsibility for the complete propulsion system.

RB211-22C

Original version for the first Lockheed L-1011 services, rated at 186.8 kN (42,000 lb st) to 18.9ºC and 172.35 kN (38,750 lb st) to 28.9ºC. Certificated in February 1972 by the CAA and in April 1972 by the FAA.

RB211-22B

Improved engine, fitted to L-1011-1 and -100, rated at 186.8 kN (42,000 lb st) to 28.9ºC. Fan with 33 blades of solid titanium with part-span snubbers. Pod-mounted reverser, but jetpipe with thrust spoiler. Mass flow 626 kg (1,380 lb)/s. Overall pressure ratio 25. Bypass ratio 4.89. Certificated April 1973. All Dash-22 engines brought to this standard.

RB211-524

Redesigned family eliminating all previous deficiencies, with new fan handling greater mass flow at significantly greater pressure ratio and new core. IP compressor handling 9 per cent greater airflow (bypass ratio reduced to 4.4) and redesigned HP spool with skewed blading. Improved HP turbine for higher TGT with reduced metal temperatures, enlarged jetpipe with short afterbody. Core thrust spoiler omitted. First engine run 1 October 1973. Later models still in production with over 1,200 delivered.

RB211-524-02

Initial Dash-524 version for L-1011-200, entered service 1977 with TGT 1,252ºC rated at 213.49 kN (48,000 lb st) to 28.9ºC.

RB211-524B-02

For L-1011-200 and -500, interrupted-spigot HP turbine disc, PS1 bleeds and inlet guide vane control, improved fan and core nozzles, new 103 Mk 8 FFR (fuel-flow regulator). Mass flow 671.3 kg (1,480 lb)/s. Bypass ratio 4.4. Overall pressure ratio 29.0. Rated at 222.4 kN (50,000 lb st) to 28.9ºC.

RB211-524B2-19

For 747-200 and 747SP, larger cabin offtake and different bleed system, modified reverser. Rated at 222.4 kN (50,000 lb st) to 28.9ºC.

RB211-524C2-19

For 747-200, PS1 control, additional combustion chamber offtake, HP3 air used to cool HP turbine, HP/IP turbine housing with Waspaloy bolts, 102 Mk 6 FFR. Rated at 229.1 kN (51,500 lb). Entered service 1980.

RB211-524B3-02

For L-1011-500. Rebladed fan of 2,180 mm (85.8 in) diameter, moved 73.7 mm (2.9 in) forward, fitted with composite nosecone, surrounded by longer Armco containment and with lighter decambered outlet guide vanes. Modified IP spool, new fuel system with 104 Mk 1 FFR. New pod with short inlet, improved noise attenuation and composite access doors. Certificated May 1980 at 222.4 kN (50,000 lb st) with 3.5 per cent lower sfc.

RB211-524B4-02

For L-1011-500. High-capacity IP/LP turbine module and HP3 pre-swirl, new turbine suspension. Certificated February 1981 at 222.4 kN (50,000 lb st) with sfc 4.8 per cent better than 524B.

RB211-524B4 Improved

1987 revision for L-1011-250. Polished fan with tighter tip clearance, smoothed outlet guide vanes with polyurethane coating, new intermediate-case bearings, new combustor locating dogs, HP turbine with improved disc and blades, new IP turbine (as D4 Upgrade), new sealing strips and ceramic cement

throughout to give sfc 6.7 per cent better than 524B.

RB211-524D4-19

For 747-200. Increased-efficiency fan with mass flow 702 kg (1,548 lb)/s, new IP compressor with modified guide vanes and thicker IP1/IP2 discs, thicker outlet case to combustor, lightweight reverser, acoustic lining to core nozzle, increased-capacity gearbox, oil cooler and 105 Mk 2 FFR with new amplifier. Rated at 235.75 kN (53,000 lb st) with sfc 4.8 per cent lower than 524C2-19.

RB211-524D4 Upgrade

For 747 versions. Polished fan blades with snubbers (clappers) de-scissored, skewed HP spool with polished aerofoils and improved abradable lining, tougher combustion chamber with low-pollution liner and no diffuser offtake, new IP turbine with multilean guide vanes and variable-work blades. Rated at 235.75 kN (53,000 lb st) with sfc 2.5 per cent better than D4.

RB211-524D4-B19

For 747, as 524D4 Upgrade but with new intermediate-case bearings and new frequency limit on HP6 blading.

RB211-524G-19

For 747-400 and 767. New fan of 2,192 mm (86.3 in) diameter with 24 wide-chord hollow blades (same technology as 535E4 engine), new spinner, disc, shaft, annulus fillers and case with 56 supercritical outlet guide vanes; mass flow 728 kg (1,605 lb)/s. Titanium torsion box, new front bearing (front flange 68.6 mm, 2.7 in, forward), new rear case, new IP spool with optimised guide vanes, fewer drain holes, improved abradable lining and surface finish. New HP spool with improved HP1 and HP6 blading, modified offtakes and smoothed abradable lining leading to strengthened combustor outer case and outlet guide vanes. New HP turbine with optimised tip clearance, MARM 002 nozzles, and disc labyrinth seal. New IP/LP turbines with similar improvements and blanked firtree roots. Lightweight reverser with translating cowl of carbon-fibre composite. Multilobe mixer to common nozzle of titanium with new tailcone. Completely redesigned pod and pylon. FADEC control with option of on-condition monitoring. Uprated gearbox and 90 kVA generator. Delivered from May 1988 rated at 258 kN (58,000 lb st). Cruise sfc 8.8 per cent better than 524D4.

RB211-524H-36

For 747-400 and 767-300. As 524G-19 but with improved combustor and HP turbine cooling. Overall pressure ratio raised from 33 to 34.5. Certificated November 1989 at 269.6 kN (60,600 lb st), flat-rated to 30ºC (86ºF).

RB211-524G/H-T

This is a significantly improved engine, incorporating the complete core of the Trent 700. Mass flow rises to 738.5 kg (1,628 lb)/s, BPR is 4.1 and OPR 34.5. The hybrid G/H-T offers customers a broad range of advantages on both new production and as a retrofit package. It can be operated at either G or H rating. It gives operators a 2 per cent improvement in fuel burn. It also saves approximately 109 kg

(241 lb) weight per engine, and gives increased temperature margins, improved performance retention and longer on-wing life, besides meeting all current and planned environmental legislation, including the lowest NO_x. The G/H-T was certificated on 11 May 1997 and engine/airframe certification followed in April 1998. By 2000, six airlines had committed to this engine, which entered service with Cathay Pacific in April 1998. One order, for 108 upgrade kits for British Airways, has a value of US$200 million.

The **RB211-535** is described as the **535** at the start of Rolls-Royce entries.

The following description relates basically to the RB211-524G-19:

Type

Three-shaft axial turbofan.

Fan

Single-stage overhung, driven by LP turbine. Composite nosecone, 24 hollow wide-chord blades in titanium alloy, controlled diffusion outlet guide vanes. Aluminium casing, with Armco containment ring. Mass flow 728 kg (1,605 lb)/s. Bypass ratio 4.3.

IP Compressor

Seven-stage, driven by IP turbine. Two drums, one of titanium discs welded together and the other of welded steel discs, bolted to form one rotor, carrying titanium blades. Aluminium and steel casings carry steel stator blades. Single-stage titanium variable inlet guide vanes.

HP Compressor

Six-stage, driven by HP turbine. Welded titanium discs, single steel disc and welded nickel alloy discs bolted together carrying titanium, steel and nickel alloy blades. Steel casing carries steel and Nimonic stator blades. Overall pressure ratio 33.

Combustion Chamber

Fully annular, with steel outer casings and nickel alloy combustor. Downstream fuel injection by 18 airspray burners with annular atomisers. High-energy igniter plugs in Nos. 8 and 12 burners.

HP Turbine

Single stage, with directionally solidified nickel alloy rotor blades, both convection- and film-cooled, mounted in nickel alloy disc by fir-tree roots.

IP Turbine

Single stage, with directionally solidified nickel alloy rotor blades fir-tree-mounted in nickel alloy disc.

LP Turbine

Three-stage, with nickel alloy rotor blades fir-tree-mounted in steel discs.

Jetpipe

Integrated nozzle with deep-chute forced mixer.

Accessories

Radial drive from HP shaft to gearbox on fan casing. Accessories include integrated drive generator and

aircraft hydraulic pumps.

Oil System

Continuous circulation dry sump system supplying oil to four bearing chambers with a combination of ball and roller bearings. 27 litre (7.1 US gallon, 5.9 Imp gallon) oil tank integral with gearbox.

Dimensions

Length overall:	
RB211-22B, -524C2	3,033 mm (119.4 in)
RB211-524B4, -524D4	3,106 mm (122.3 in)
RB211-524G, -524H, -524G/H-T	3,175 mm (125.0 in)
Fan diameter:	
RB211-22B, -524C2	2,154 mm (84.8 in)
RB211-524B4, -524D4	2,180 mm (85.8 in)
RB211-524G, -524H, -524G/H-T	2,192 mm (86.3 in)

Weight, Dry

RB211-22B	4,171 kg (9,195 lb)
RB211-524B4, B4 Improved	4,452 kg (9,814 lb)
RB211-524C2	4,472 kg (9,859 lb)
RB211-524D4, D4 Upgrade	4,479 kg (9,874 lb)
RB211-524G/H	4,386 kg (9,670 lb)
RB211-524G/H-T	4,277 kg (9,429 lb)

Performance Ratings

T-O: see model listings	
Cruise at 10,670 m (35,000 ft) and M0.85 (uninstalled):	
RB211-22B	42.2 kN (9,495 lb st)
RB211-524B4, B4 Improved	48.9 kN (11,000 lb st)
RB211-524C2	51.1 kN (11,490 lb st)
RB211-524D4 (all models)	50.0 kN (11,230 lb st)
RB211-524G, -524H, 524G/H-T	52.5 kN (11,813 lb st)

Specific Fuel Consumption

(Cruise)

RB211-22B	17.79 mg/Ns (0.628 lb/h/lb)
RB211-524B4	17.56 mg/Ns (0.620 lb/h/lb)
RB211-524B4 Improved	17.16 mg/Ns (0.606 lb/h/lb)
RB211-524C2	18.21 mg/Ns (0.642 lb/h/lb)
RB211-524D4	17.48 mg/Ns (0.617 lb/h/lb)
RB211-524D4 Upgrade (1987)	17.02 mg/Ns (0.601 lb/h/lb)
RB211-524G, -524H	16.15 mg/Ns (0.570 lb/h/lb)
RB211-524G/H-T	15.83 mg/Ns (0.560 lb/h/lb)

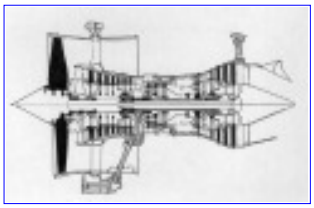
Contract Price

June 1997, engines for three 767-300s: US$55 million.

UPDATED

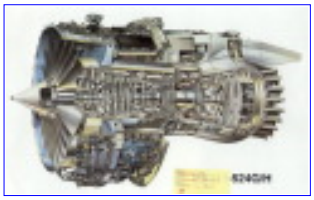
RB211-22B

RB211-524D4

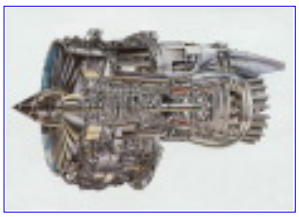


Longitudinal sections of RB211-524G (upper half) and RB211-524D4 (lower)

RB211-524G



Cutway drawing of RB211-524G



Cutaway of RB211G/H-T (2000)

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AERO-ENGINES - TURBOFAN, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE PEGASUS

USMC designation: F402

Originally designed by Bristol Siddeley, the Pegasus is a two-shaft turbofan designed for short take-off/vertical landing (STOVL) applications. It powers all versions of the Harrier attack and training aircraft. The Pegasus provides both lift and propulsive thrust through four swivelling exhaust nozzles which vector the engine's thrust from horizontal, for conventional (wingborne) flight, to vertical, for jetborne flight, and beyond the vertical for rapid inflight deceleration. To minimise aircraft control problems in jetborne flight, thrust is divided between the engine nozzles to ensure that the resultant thrust passes through a fixed point irrespective of nozzle angle. The LP and HP spools rotate in opposite directions, to minimise gyroscopic effects, and HP bleed air is used for aircraft stabilisation in hovering flight.

Pegasus 11

Pegasus entered service in the BAe Harrier in 1969 as the Pegasus 6 Mk 101, progressing through Pegasus 10 Mk 102 in 1971 to Pegasus 11 Mk 103 by 1974.

Pegasus 11-21

This engine was developed for the McDonnell Douglas/BAe Harrier II. Developed from the Pegasus 11, the 11-21 provided much improved reliability, a substantially reduced maintenance burden and offered a small thrust increase. Delivery of Pegasus 11-21 began in December 1984. Since 1986 Pegasus 11-21 engines for the Harrier II AV-8B and GR.5/7 have been fitted with a DECU (Digital Engine Control Unit).

Pegasus 11-61

This was the final production version, offering up to 15 per cent greater thrust than the 11-21 at high ambient temperatures. It entered service in 1990 with the US Marine Corps. To reduce the cost of engine ownership, the 11-61 has twice the overhaul life of the 11-21. Maintenance activity is minimised through use of proven digital engine control and engine monitoring systems, improved inspection facilities and modular construction. This engine had long been considered as an upgrade option for the UK Harrier fleet. At last, in December 1999 it was announced that the Ministry of Defence had signed a contract for £120 million for the upgrade of 40 of the Mk 105 engines fitted to the Harrier GR.7 to Mk 107 standard. This work is to be completed by the start of 2004. The deal includes an option, worth an additional £230 million, to upgrade a further 86 engines. Tim Jones, Executive V-P, Customer Business for Defence (Europe), described the contract as ``a tremendous success for Smart Procurement, a ground-breaking contract.'' Nothing has been announced regarding any upgrade for the Mk 106 engines of Royal Navy Sea Harriers.

Total Pegasus experience exceeds 1.6 million hours on over 1,100 engines. Engines are supported by Rolls-Royce East Kilbride. The following are the principal in-service versions:

Engine Mark	Operator	Designation
Pegasus 11	Royal Air Force	Mk 103
	Royal Navy	Mk 104
	Royal Thai Navy	Mk 150
	Indian Navy	Mk 151-32
Pegasus 11-21	Royal Air Force	Mk 105
	Royal Navy	Mk 106
	US Marine Corps	F402-RR-406A
	Spanish Navy	Mk 152-42
Pegasus 11-61	US Marine Corps,	F402-RR-408A & B
	Spanish Navy and Italian Navy	
	Royal Air Force	Mk 107

Type

Two-shaft vectored-thrust turbofan.

Fan

Three-stage, overhung ahead of front bearing. Titanium alloy blades with advanced circumferential snubbers. Mass flow (11) 196 kg (432 lb)/s; (11-21) 205 kg (454 lb)/s; (11-61) 209 kg (461 lb)/s. Pressure ratio (11-21) 2.5, (11-61) 2.7. Bypass ratio (11) 1.38, (11-21) 1.4, (11-61) 1.2.

HP Compressor

Eight-stage with titanium alloy rotor blades and discs. Pressure ratio (11) 14, (11-61) 16.0.

Combustion Chamber

Annular with `T-shaped' fuel vaporisers.

HP Turbine

Two-stage cooled HP turbine incorporating single-crystal blading.

LP Turbine

Two-stage uncooled LP turbine with single-crystal material in first stage.

Nozzle

Two fabricated steel zero-scarf front (cold) nozzles and two Nimonic rear (hot) nozzles rotated simultaneously by bleed-air driven air motor under pilot command. The complete assembly is part of the airframe.

Starting

Gas-turbine starter/APU driving through engine gearbox.

Control System

FADEC comprising duplicated digital engine control units (DECU) and a hydromechanical fuel metering unit (FMU); Sea Harrier engines are entirely hydromechanical. The FMU includes a cockpit-selectable manual fuel control system.

Oil System

Self-contained with fuel-cooled oil cooler.

Dimensions

Width, including nozzles	2,510 mm (98.8 in)
Length, including nozzles	3,485 mm (137.2 in)
Diameter (inlet and fan):	
11	1,220 mm (48.0 in)
11-61	1,222 mm (48.1 in)

Weight, Dry

(including nozzles)

103	1,412 kg (3,113 lb)
104	1,442 kg (3,179 lb)
105	1,470 kg (3,240 lb)
11-61	1,570 kg (3,461 lb)

Performance Ratings

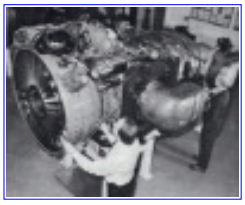
(T-O, S/L, ISA)

11	95.64 kN (21,500 lb st)
11-21	96.75 kN (21,750 lb st)
11-61	106.0 kN (23,800 lb st)

UPDATED

Assembly of Pegasus 105

Cutaway drawing of Pegasus 11-61



Pegasus 11-61 (F402-RR-408)

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AERO-ENGINES - TURBOFAN, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE CONWAY

This was the world's first large turbofan. Its initial production versions were the Mk 508 (RCo.12) for the 707-420 and the Mk 509 (RCo.12) for the DC-8-40. The Mk 201 at RCo.17 rating of 91.64 kN (20,150 lb st) powered the Victor B.2 bomber and K.2 tanker.

RCo.43 Mk 301

This engine powers all marks of VC10 and Super VC10 in service with the RAF. In December 1998 it completed two million hours in RAF service. The following description refers to this version. A total of 904 Conway engines was produced.

Type

Two-spool axial-flow turbofan.

Intake

Annular, fabricated steel assembly in two circumferentially separate pieces. Forward casing carries nose fairing supported by 17 fixed-incidence hollow steel sheet inlet guide vanes. The casing carries the LP compressor front bearing housing, supported by 31 hollow zero-stage LP stator blades of fabricated steel. Anti-icing is provided by hot air bled from HP compressor outlet casing and fed into inlet guide

vanes and zero-stage stators.

LP Compressor

Eight-stage axial-flow type, consisting of a drum carrying first four (fan) stages of blades, bolted to rear section of shaft to which are splined the four intermediate discs. All blades are unshrouded, and are attached to drum or discs by steel pins. The LP compressor casing consists of three steel rings shrouding the first three rotor stages and an axial split two-piece aluminium casing housing the later stages. All stator blades are of aluminium located by tongue and groove arrangement, and are shrouded at their inner ends. Shrouds carry interstage seals. LP shaft carried on two roller bearings; axial location by ball thrust bearing inside HP shaft. Mass flow 170.1 kg (375 lb)/s. Pressure ratio 15.77. Bypass ratio (S/L, ISA) 0.56. Intermediate casing is a magnesium casting housing air flow duct, entry to bypass and internal wheelcase. Inner casing connected to outer by eight hollow webs carrying concentric fairing at rear edges to provide air flow separation into bypass. LP outlet guide vanes located at mouth of air flow duct and HP inlet guide vanes located at rear between inner casing and bypass intake fairing.

HP Compressor

Nine-stage axial-flow type, consisting of nine steel discs on two-piece stainless steel shaft. Two sections of shaft connected by bolted flanges. Stage 1 disc located by bolts to flanges of shafts. Stage 2 disc located by shaft connecting bolts. Stages 3-9 discs splined on to shaft. Rotor blades of titanium (stages 1-7) and steel (stages 8-9), attached to discs by hollow steel pins. HP casing of fabricated steel carrying stages 1-8. HP stator blades located by rings attached to casing by set screws. Stator blades of stainless steel. HP shaft carried in front roller bearing and rear ball thrust bearing. Outlet is a fabricated steel casing forming annular air flow duct. Two bell-shaped housings joined by 10 webs. HP compressor outlet guide vanes located at front (small) end of casing and 10 stainless steel flame tube scoops located at rear (large) end.

Combustion Chamber

Can-annular steel combustion chamber carrying 10 Nimonic flame tubes, interconnected for initial flame spread. Duplex downstream burners, one per can. High-energy igniter plugs situated in Nos 4 and 7 flame tubes.

HP Turbine

Single-stage. Ferritic steel disc secured to HP shaft by taper bolts. Blades of Nimonic alloy air-cooled, located by fir-tree roots. Roller bearing ahead of turbine disc. Stainless steel nozzle casing, supporting inner cone by 10 hollow struts. 10 Nimonic flame tube discharge nozzles at forward end. Three sets of cast nozzle guide vanes. Stage 1 vanes (HP), located between inner and outer casings, are air-cooled.

LP Turbine

Two-stage. Stage 1 disc of steel bolted to LP shaft, extended rearwards to carry steel Stage 2 disc which is located by spigots and supported in roller bearing at tail. Blades of Nimonic alloy located by fir-tree roots. Second and third stage (LP 1 and 2) vanes are located in outer casing and secured by ring at inner ends.

Jetpipe

Outer conical member supports LP turbine bearing via 10 radial struts. All structural members shielded by 10 segments joined by 10 streamlined fairings. Entire structure of stainless steel sheet. The bypass duct is a two-piece assembly of titanium separated circumferentially. Extends from rear of compressor

intermediate casing to end of exhaust unit. Located by flange at forward end and by a serrated ring at rear.

Reverser

Located immediately aft of exhaust assembly. Consists of two ram-operated clamshell doors which, when closed, direct jetstream toward front via cascade vanes in reverser walls. Reverse thrust equals 50 per cent normal forward thrust. No noise suppressor fitted to RAF VC10.

Mounting

Saddle mounts above compressor intermediate casing. Single suspension point above nozzle box.

Accessories

Drives in compressor intermediate casing, through bottom three webs to port, starboard and lower wheelcases. Port drive from HP shaft to oil pumps, scavenge pumps, dual fuel pumps, alternator drive and centrifugal breather. Starboard drive from HP shaft to air starter, HP tachometer generator, fuel backing pump, hydraulic pump and scavenge pump. Lower drive from LP shaft to LP tachometer generator.

Starting

Rotax CT 1012 air turbine starter.

Control System

Backing pump delivers LP fuel through heater (hot air) and filter to dual variable-stroke multiplunger HP pumps. Thence through flow control unit to burners. Lucas flow control unit gives proportional flow and is governed by throttle opening and air intake pressure. Acceleration control unit prevents surge during violent throttle handling. Max fuel pressure 119 kg/cm^2 (1,700 lb/sq in).

Fuel Specification

DERD 2482, US MIL F-5616-Avtur, or DERD 2496, US MIL F-5624C-Avtag.

Oil System

Wet sump type. Oil circulated by one pressure and seven scavenge pumps.

Oil Specification

Aeroshell ASTO.390, Castrol 3C or 325, OX-7, all to DEngRD.2498. Sump capacity 13.6 litres (3.6 US gallons; 3 Imp gallons). Normal oil supply pressure 2.8 kg/cm^2 (40 lb/sq in).

Dimensions

Length overall	3,912 mm (154.0 in)
Diameter (max)	1,717 mm (67.6 in)

Weight, Dry

	2,420 kg (5,335 lb)

Performance Ratings

T-O (S/L, ISA)	100 kN (22,500 lb st)
Max continuous	79.6 kN (17,900 lb st)
Cruise (11,000 m; 36,090 ft, 880 km/h; 547 mph)	23.78 kN (5,345 lb)

Specific Fuel Consumption

T-O	17.69 mg/Ns (0.625 lb/h/lb st)
Max continuous	17.10 mg/Ns (0.605 lb/h/lb st)
Cruise, as above	23.14 mg/Ns (0.819 lb/h/lb)

Oil Consumption

	0.51 litres (0.9 Imp pint)/h

UPDATED

Two views of RCo.43 Mk 301

Two views of RCo.43 Mk 301

Twin RCo.43 in VC10

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AERO-ENGINES - TURBOJET, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE ORPHEUS

The Orpheus is a single-spool turbojet of particularly simple design. It was initiated in December 1953 by what later became Bristol Aero-Engines as a private venture, to meet the requirement for an engine in the medium-thrust range for applications where low weight and small frontal area are of prime importance.

The initial BOr.1 version first ran on 17 December 1954, and completed a type test in spring 1955. It first flew in the prototype Gnat on 18 July 1955. The following variants are still in operation.

BOr.2

This subseries was first type-tested in November 1956, at a rating of 20.04 kN (4,520 lb st). The Mk 701 powered the HAL Gnat, no longer in IAF service. The Mk 701-1, derated to 15.07 kN (3,400 lb st), was made under licence by HAL in India and powers the HAL Kiran II trainer. The fully rated Mk 703 powers surviving HF-24 Marut fighters and trainers.

BOr.3

Engines of this type were first type-tested in May 1957. The fully rated Mk 803 was made under licence in Germany and Italy (see below) to power the Aeritalia G91. The Mk 805, derated to 17.74 kN (4,000 lb st), powers the Japanese Fuji T-1A trainer. None of these aircraft remain in military service, but examples of the G91 and T-1A are airworthy.

BOr.4

Similar to the BOr.3, this series was derated to reduce fuel burn and extend life. Fully anti-iced, it powers a surviving Gnat T.1 trainer.

The Orpheus was developed under Ministry of Aviation contract in connection with the joint Mutual Weapons Development Programme agreement between the United States and the United Kingdom. It was the only engine selected for development under this programme for the NATO lightweight strike fighter.

The Orpheus was manufactured under licence by HAL at Bangalore, India, to power the Gnat, Kiran II and Marut, in Italy by Fiat for the G91 and in Germany by Klöckner-Humboldt-Deutz, also for the G91.

The following data apply specifically to the Orpheus 803 (BOr.3):

Type

Axial flow lightweight turbojet.

Intake

Fabricated from light alloy, with cartridge starter housed in centrebody. Casting in ZRE magnesium-zirconium alloy, housing front bearing and having integral gear chamber for accessory drives.

Compressor

Seven-stage axial flow type. Steel and aluminium alloy rotor blades attached by fir-tree roots to seven rotor discs separated by spacer rings, the assembly being secured by through bolts. Fixed-incidence inlet guide vanes mounted in air intake casing. Six rows of stator blades mounted in dovetail grooves in two-part ZRE stator casing, all but two rows of stator blades being of aluminium alloy. Delivery via aluminium alloy casting having inner and outer portions joined by seven radial struts, between which are mounted the seven burners. The delivery casing houses the compressor outlet guide vanes, and also carries the two igniter plugs and air tapping connections.

Combustion Chamber

Can-annular type, with inner and outer casings of heat-resistant steel containing seven flame tubes. Each flame tube comprises an integral turbine stator (NGV) nine-vane segment, adjacent segments being bolted together to form complete annulus.

Turbine

Single-stage axial with 125 unshrouded blades, mounted in disc rim by fir-tree fixings. NGVs, see under Combustion Chamber.

Jetpipe

Welded steel construction, secured to exhaust cone flange. Surrounded by Refrasil heat-insulating blanket and fitted with thermocouple bosses. Jetpipe and exhaust cone are surrounded by light alloy shroud forming annular cooling air space.

Mounting

Three-point suspension, with main trunnions on each side of delivery casing and tie link from forward face of gear casing.

Accessories

Gear chamber integral with lower portion of air intake casing provides drives for fuel pump, two hydraulic pumps, electric generator, tachometer and oil pump. Drive from main rotor via bevel gears and shaft

through lower vertical strut in intake casing.

Starting

Twin-breech cartridge starter, the breeches being mounted on the right side of the compressor casing and the starter mechanism within the nose bullet of the intake duct.

Control System

Comprises Lucas variable-stroke pump, combined control unit including a unitary throttle and shut-off cock, flow distributor and barometric pressure control, together with air/fuel ratio control and overspeed governor. Duplex burners.

Oil System

Comprises main pressure pump, metering pump and two scavenge pumps. Oil from pressure pump is led to front bearing and metering pump, the latter supplying accessory drives and rear bearing, oil from which is exhausted into jet efflux. Scavenge oil from front bearing and accessory drives is returned to tank.

Dimensions

Length:	
BOr.2	1,854 mm (73.0 in)
BOr.3	1,916 mm (75.45 in)
BOr.4	1,919 mm (75.5 in)
Diameter	823 mm (32.4 in)

Weight, Dry

BOr.2	358 kg (790 lb)
BOr.3	374 kg (825 lb)
BOr.4	408 kg (900 lb)

Performance Rating

(T-O, S/L, ISA)

BOr.2	21.57 kN (4,850 lb st)
BOr.3	22.24 kN (5,000 lb st)
BOr.4	18.82 kN (4,230 lb st)

Specific Fuel Consumption

(T-O)

BOr.2	29.93 mg/Ns (1.057 lb/h/lb st)
BOr.3	30.59 mg/Ns (1.08 lb/h/lb st)
BOr.4	29.38 mg/Ns (0.964 lb/h/lb st)

UPDATED

Orpheus 803 (BOr.3)

Orpheus 701-1 made by HAL

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AERO-ENGINES - TURBOSHAFT, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE NIMBUS

The Nimbus is a free-turbine turboshaft, originally derived from Turbomeca designs by Blackburn Engines of the Hawker Siddeley Group, with the designation A.129. Via Bristol Siddeley it became a Rolls-Royce engine, but in 2001 all rights to it were transferred to H+S Aviation of Portsmouth, Hampshire. It is retained here for convenience.

It is basically an Artouste with two axial flow compressor stages added forward of the centrifugal compressor and one axial turbine stage added to the gas generator. The shaft-drive is taken via a free turbine and a two-stage gearbox. The Nimbus was first run as a turbojet in July 1958, and as a turboshaft in the following month. In flat-rated form, the Nimbus powers the Westland Scout and Wasp helicopters. For the Wasp the engine has been 'marinised' to combat the effects of salt-water ingestion.

Type

Free-turbine turboshaft engine.

Intake

Annular aluminium-alloy casting with three radial struts supporting front ball-thrust bearing.

Compressor

Two-stage axial-flow compressor, followed by single-sided centrifugal stage. Axial stages have integrally machined blades and integral stub-shafts and are bolted together. Shaft supported at front in high-speed ball

bearing and at rear in high-speed roller bearing. Cast stator blades in inner and outer retainer rings. Stainless-steel centrifugal compressor. Pressure ratio 6.5. Air mass flow 5.0 kg (11 lb)/s.

Combustion Chamber

Annular type. Fuel is injected centrifugally into flame zone of combustion chamber through radial holes in hollow mainshaft. Starting is by torch igniter.

Compressor Turbine

Two-stage axial, with integrally machined blisks.

Power Turbine

Single-stage free turbine of Nimonic, with integrally machined blisks.

Jetpipe

Bifurcated type.

Output

Helical spur reduction gear.

Accessories

Engine-driven auxiliaries and accessories mounted on taper flanges around air intake. Driven by spur gear train from compressor.

Control System

Gear-type pump supplies fuel metered by a mechanically governed control unit.

Oil System

Gear type compound pressure and scavenge pump, full-flow filter and system of oil strainers.

Dimensions

Installed overall length	1,854 mm (73.0 in)
Width	980 mm (38.6 in)
Height	868 mm (34.2 in)

Weight, Dry

Dry, less gearbox, approx	177 kg (390 lb)

Performance Ratings

Max (5 min up to ISA + 30ºC)	529 kW (710 shp)
1 h (to ISA + 27ºC)	511 kW (685 shp)
Max continuous	447 kW (600 shp)

Specific Fuel Consumption

At max continuous rating	150.4 µg/J (0.89 lb/h/shp)

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Nimbus installed in Scout

Nimbus installed in Wasp

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AERO-ENGINES - TURBOJET, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE NENE

This simple centrifugal turbojet first ran in October 1944. It swiftly reached 2,268 kg (5,000 lb) thrust, making it the most powerful aircraft engine in the world. Though little used in Britain, it was produced in vast numbers in the USA (as the Pratt & Whitney J42), in the Soviet Union (as the RD-45), in France (as the Hispano-Suiza Nene), and in China (as the WP5). It was also produced by Rolls-Royce in Canada and Australia.

All these versions are still extant, and the T-33AN Silver Star remains in service with the Canadian Armed Forces. The only Nene-family engines flying in large numbers are the derived ex-Soviet VK-1 and the Chinese WP5. These are described in the Russian section under Klimov.

Mk 102 (RN.2)

Baseline UK production engine, to which the main description refers. Powered Attacker and various other aircraft but no longer active.

Mk 10 (RN.2)

Similar to Mk 102 but a larger wheelcase is provided to mount and drive aircraft accessories. Engine adapted for installation in Canadair Silver Star (Lockeed T-33) trainer.

Mk 103 (RN.6)

Similar to Mk 102 but with thrust increased. Minor structural changes to accommodate increased working temperatures. Cartridge turbo-starter. Powers Sea Hawk.

Mk 105

Derived from Mk 102, with mass flow increased to 41.0 kg (90.2 lb)/s. Fitted with French accessories. Produced by Hispano-Suiza for Dassault Ouragan.

Type

Single-shaft turbojet.

Intake

Air rammed in at the aircraft inlet is fed to an airtight plenum chamber surrounding the engine. From here it is sucked in to the front and rear of the compressor via fine mesh screens.

Compressor

Single-stage double-entry centrifugal compressor with double-sided impeller. Impeller has 29 vanes per side with separate forged aluminium rotating guide vanes machined all over. Compression ratio (static) 4.45:1. Air mass flow 40 kg (88 lb)/s. Rotor assembly, consisting of impeller, turbine and shafts, supported on three bearings. End bearings of roller type and centre bearing a deep-grooved ball bearing. Mounted on shaft between compressor and centre bearing is a fan which directs cooling air on to and under centre and rear bearings and face of turbine disc.

Combustion Chamber

Nine straight-flow combustion chambers each consisting of an outer casing, a perforated flame tube and duplex burner. Flame igniters in Nos 3 and 8 air casings.

Turbine

Single-stage axial flow turbine comprising solid steel disc and 54 blades in Nimonic 80 nickel chromium alloy. Blades secured in periphery of turbine disc by fir-tree type serrated roots. Wheel and shaft bolted together. Jetpipe temperature 690ºC approx.

Jetpipe

Consists of jetpipe and propelling nozzle. Exhaust cone of fixed length but jetpipe varies in length according to installation requirements of aircraft. These parts mainly double-walled, with space between packed with Alfol heat-insulating material. Air heating jackets can be arranged around jetpipe for gun or cabin heating. Maximum jetpipe temperature 734ºC.

Mounting

Range of standardised brackets designed to suit various types of installation with six alternative combinations of attachment points.

Accessories

Wheelcase on front of engine, driven from forward impeller shaft, houses drives for aircraft accessory gearbox, tachometer generator and two fuel pumps. Alternative drive positions for aircraft accessory gearbox, upper horizontal drive and upward inclined drive, suiting Rotol SG1 series of gearboxes. Speed of

gearbox drives 0.421 engine speed. Cabin supercharger driven from accessory gearbox. 24 V electric starter motor mounted on port side of wheelcase.

Control System

Twin pumps of oscillating multiplunger type with built-in overspeed governors draw fuel through filter mounted under wheelcase and deliver to burners via throttle control valve, high-pressure shut-off cock and pressurising valve. Barometric pressure control acts on servo mechanism in high-pressure pump to vary pump delivery according to altitude requirements. Acceleration control unit retards fuel flow to prevent over-fuelling during acceleration periods.

Fuel Specification

Aviation Turbofuel (DERD 2482).

Oil System

Continuous circulation system, with main bulk of oil contained in sump mounted on lower part of wheelcase. Sump houses pressure and scavenge oil-pumps, two gauze scavenge oil filters. Purolator high-pressure filter, pressure relief valve and de-aerator. Capacity of oil sump 5.7 litres (9 Imp pints). Specification DEF.2001 oil is used.

Dimensions

Length (to exhaust flange, typical)	2,440 mm (96.0 in)
Diameter	1,258 mm (49.5 in)

Weight, Dry

Mk 10	735 kg (1,620 lb)
Mk 105	730 kg (1,606 lb)

Performance Rating

(T-O, S/L)

Mk-10	22.24 kN (5,000 lb st)
Mk 103	24.02 kN (5,400 lb st)
Mk 105	22.69 kN (5,100 lb st)

Specific Fuel Consumption

(T-O, as above)

Typical	30.03 mg/Ns (1.06 lb/h/lb st)

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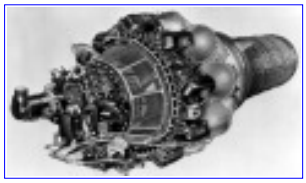

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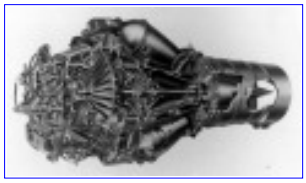

Cutaway Nene 103

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AERO-ENGINES - TURBOSHAFT, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE GNOME

This engine is a development of the General Electric T58, made (initially by de Havilland Engines, which was merged into Bristol Siddeley) under licence. The first Gnome engine ran on 5 June 1959, since when more than 2,300 engines have been delivered. Of these about 1,170 remain in service. Total flight time in 2001 was 7.3 million hours.

H.1000

Initial version. Power plant for military Whirlwind HAR Mk 9, HAR Mk 10 and HCC Mk 12, civil S-55 Series 3 and Agusta-Bell 204B.

H.1200

Used in Agusta-Bell 204B, Boeing Vertol 107 and some Kawasaki KV-107/II-5s.

Coupled Gnome H.1200

This engine, for the Wessex Mks 2, 5, 50 and 60 series, comprises two H.1200s driving through a coupling gearbox designed and manufactured by Rolls-Royce. Maximum potential output is 1,864 kW (2,500 shp), but the Wessex transmission is limited to 1,156 kW (1,550 shp) at the rotor head. Should either power section be shut down, the other will automatically increase power to the required output, up to the standard maximum H.1200 emergency rating of 1,007 kW (1,350 shp). The ratio of the Wessex coupling gearbox is 7.476:1. This version is expected to pass out of service in 2002.

H.1400

Based on the H.1200, with modified compressor to increase air flow. Turbine diaphragm cooling redesigned to increase temperature capacity and life. Dimensions unchanged. Mounting pads identical to those of H.1200. Produced for the Westland Sea King and Commando.

H.1400-1

Uprated H.1400, with increased gas generator speed and improved gas generator turbine-blade material. Produced for Sea King and Commando.

H.1400-1T

As H.1400-1 plus addition of a power-turbine inlet temperature limiter. Turbine nozzle adjusted to improve performance in hot climates.

The following description refers specifically to the H.1400 turboshaft version:

Type

Axial-flow free-turbine turboshaft engine.

Intake

Annular forward-facing. Centre housing carrying front main bearing supported by four radial struts. Struts and inlet guide vanes anti-iced with hot compressor bleed air and oil drainage.

Compressor

Ten-stage axial. Controlled variable incidence for inlet guide vanes and first three rows of stator blades. Integral spool-type rotor assembly with rotor blades secured in dovetail root fittings. Rotor splined to shaft which is carried on roller bearings at front and ball bearing at rear. Main steel casing split along horizontal centreline, with stator blades brazed in carrier rings. Mass flow, H.1200, 5.7 kg (12.55 lb)/s; H.1400, 6.22 kg (13.7 lb)/s. Pressure ratio, H.1200, 8.12; H.1400, 8.4. A short-length casing interposed between compressor and combustor has radial vanes across compressor outlet to carry main centre bearing.

Combustion Chamber

Straight-through annular chamber with outer casing split along horizontal centreline. Sixteen Simplex-type fuel injectors, eight on each of two sets of manifolds. One Lodge capacitor-discharge high-energy igniter plug.

Compressor Turbine

Two-stage, coupled to compressor shaft by conical shaft. Extended-root blading with fir-tree attachments. A short-length intermediate casing interposed between gas-producer and power turbines carries power-turbine nozzle guide vanes.

Power Turbine

Single-stage free turbine. Extended-root blading with fir-tree attachments. Rotor disc integral with output shaft and overhung from rear on roller bearing on downstream face of disc and ball bearing at rear of shaft. Complete assembly mounted inside exhaust ducting.

Jetpipe

Curved exhaust ducting arranged to suit individual applications.

Mounting

Three forward mounting faces on intake casing. Two rear mounting faces on upper portion of primary gear casing. When no reduction gear fitted, rear mounting face on engine centreline at power-turbine output shaft housing.

Output

Optional double-helical gear providing reduction from nominal 19,500 rpm power turbine speed to 6,600 rpm at output shaft. Provision for power take-off to left or right.

Accessories

Quill shaft drive through lower intake strut. Fuel and lubrication systems mounted beneath compressor casing. Power take-off shaft up to 100 shp on primary reduction gear casing for separate accessories gearbox.

Starting

Rotax electric starter in nose bullet.

Control System

Lucas hydromechanical units, comprising variable-stroke multiplunger pump, flow control unit and throttle controlled by HSD electrical control computer and throttle actuator.

Fuel Specification

DERD.2453, 2454, 2486, 2494 and 2498 (NATO F34, F40, F35 and F44).

Oil System

Fully scavenged gear pumps. Serck oil cooler.

Oil Specification

Military, DEng RD 2487 and 2493, Castrol 205 GTO and Esso Turbo Oil 2380. Commercial, Aero Shell Turbine Oil 750, Esso Extra Turbo Oil 274, Castrol 98, Castrol 205 GTO and Esso Turbo Oil 2380.

Dimensions

Length:	
H.1000, H.1200, H.1400-1	1,392 mm (54.8 in)
Coupled H.1200 (Wessex)	1,747 mm (68.8 in)
Max height:	
H.1000, H.1200, H.1400-1	549 mm (21.6 in)
Coupled H.1200 (Wessex)	1,031 mm (40.6 in)
Max width:	
H.1000, H.1200 (ungeared)	462 mm (18.2 in)
H.1400-1 (ungeared)	577 mm (22.7 in)
Coupled H.1200 (Wessex)	1,059 mm (41.7 in)

Weight, Dry

H.1000 (ungeared)	134 kg (296 lb)
H.1200 (ungeared)	142 kg (314 lb)

H.1400-1 (ungeared)	148 kg (326 lb)
Reduction gearbox	52.6 kg (116 lb)
Coupled H.1200	422 kg (930 lb)

Performance Ratings

At power turbine shaft:	
Max contingency (2.5 min; multi-engine aircraft only):	
H.1200	1,007 kW (1,350 shp)
H.1400-1	1,238 kW (1,660 shp)
H.1400-1T	1,092 kW (1,465 shp) to 45ºC
Max one-hour (single engine):	
H.1000	783 kW (1,050 shp)
H.1200	932 kW (1,250 shp)
H.1400-1	1,145 kW (1,535 shp)
H.1400-1T	1,030 kW (1,380 shp) to 45ºC
Max continuous:	
H.1000	671 kW (900 shp)
H.1200	783 kW (1,050 shp)
H.1400-1	932 kW (1,250 shp)
H.1400-1T	783 kW (1,050 shp) to 45ºC

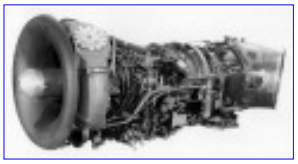
Specific Fuel Consumption

At max contingency rating:	
H.1200	104.4 µg/J (0.618 lb/h/shp)
H.1400-1	102.75 µg/J (0.608 lb/h/shp)
H.1400-1T (30ºC)	105.8 µg/J (0.626 lb/h/shp)

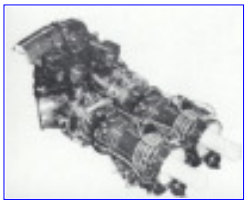
UPDATED

Cutaway drawing of Gnome H.1400-1

Coupled Gnome H.1200 in Wessex



Gnome H.1400



Coupled Gnome H.1200

Gnome H.1400 engines in Sea King

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AERO-ENGINES - TURBOSHAFT, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE GEM

The Gem was developed to power the Westland Lynx helicopter. Current applications are in the Westland Super Lynx and Agusta A 129. The Gem 41 series has been civil certified. In 2001 the Gem completed 25 years in service, and at the same time passed 3,000,000 hours in (mostly exceedingly arduous) operation. Half this time has been logged by the Gem Mk 42.

The choice of a two-spool gas generator gives fast response to power demand without the need for a complex control system. There are seven major modules, each of which is assembled, tested and released as an interchangeable unit.

The following versions are in use.

Gem 2 Series

Mk 1004 for A 129. Direct drive in place of reduction gearbox and electronic instead of hydromechanical control. Production under licence by Piaggio. Contingency rating 704 kW (944 shp).

Gem 41 Series

Modified compressor to increase mass flow by about 10 per cent plus small increase in TET. All converted to Gem 42 standard.

Gem 42 Series

Improved reliability and power retention over Gem 41. Current production standard for land-based Westland Super Lynx. Mks 204, 205 for British forces (new and conversion engines). Mk 1017 in service from new with two operators and following conversion with four more. Contingency rating 835 kW (1,120 shp).

Gem 43 Series

As Gem 41 with added electronic engine control. Current standard for Naval Lynx. Mk 1020 for Nigerian Navy. Rated at 846 kW (1,135 shp).

The following description relates to the Gem 42 Series:

Type

Free-turbine turboshaft.

Intake

Light alloy annular duct with flared outer case connected by five struts to hub.

LP Compressor

Four-stage axial. Mass flow (2) 3.22 kg (7.1 lb)/s, (42) 3.41 kg (7.52 lb)/s.

HP Compressor

Single-stage centrifugal. Overall pressure ratio (2) 11.3 to 11.8, (42) 12.0.

Combustion Chamber

Annular reverse flow with 17 air atomiser fuel sprays. High-energy ignition to four torch igniters.

HP Turbine

Single-stage close-coupled to HP impeller. Max entry gas temperature (2) 1,187ºC.

LP Turbine

Single-stage with shrouded blades.

Power Turbine

Two-stage axial with shrouded blades. Max speed 27,000 rpm.

Output

Single-stage double-helical reduction gear with rotating planet cage carried by ball bearing at front and roller bearing at rear.

Accessories

Bevel gear on front of HP shaft drives starter/generator, fuel pump, oil cooler fan, centrifugal breather, oil pump and (2, 42) hydromechanical control.

Control System

Plessey fluidics automatic control, and power matching for multi-engine installation. Hamilton Standard electronic control fitted to Mk 1004.

Oil System

Engine-mounted oil tank and cooler. Magnetic chip detectors. Oil filter in accessory wheelcase.

Dimensions

Length overall	1,099 mm (43.2 in)
Width overall	575 mm (22.6 in)
Height overall	596 mm (23.5 in)

Weight, Dry

Gem 1001, 2	178 kg (393 lb)
Gem 41, 42	183 kg (404 lb)
RR 1004	163 kg (360 lb)

Performance Ratings

See table

Specific Fuel Consumption

50 per cent max T-O:	110 µg/J (0.65 lb/h/shp)

GEM ENGINE RATINGS, kW (shp) ISA S/L STATIC

		One Engine Inoperative			Normal Twin Operation		
Designation	**Date in service**	**Emergency (20 s)**	**Max contingency (2.5 min)**	**Intermediate contingency (60 min)**	**Max (T-O) (30 min)**	**Max (T-O) (5 min)**	**Max continuous**
Gem 2, Mk 1001	1976	n/a	671 (900)	619 (830)	n/a	619 (830)	559 (750)
RR 1004	1990	759 (1,018)	777 (1,042)	657 (881)	657 (881)	n/a	615 (825)
Gem 41-1, Mk 1014	1978	n/a	835 (1,120)	790.5 (1,060)	n/a	746 (1,000)	664 (890)

Gem 42, Mk 1017	1987	n/a	835 (1,120)	790.5 (1,060)	n/a	746 (1,000)	664 (890)

n/a: not applicable

UPDATED

Cutaway drawing of Gem 42

A Gem 42 being serviced by the Royal Norwegian Air Force

RR 1004 for A129

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AERO-ENGINES - TURBOSHAFT, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE GAZELLE

The Gazelle is a free-turbine engine which can be mounted in a horizontal, vertical or inclined attitude. It was designed and developed by D.Napier & Son Ltd primarily as a helicopter engine, and is still used in some Westland Wessex helicopters. Design of the Gazelle began in June 1954, and the first unit ran on 3 December 1955. MoA helicopter type approval was received in September 1959.

Current versions of the Gazelle are as follows:

Gazelle NGA.22 Mk 165

Potential one-hour rating of 1,335 kW (1,790 shp) but flat rated. Fitted in Wessex Mk 3.

Gazelle NGa.22 Mk 165C

Identical to Mk 165, except for having generator instead of alternator. Used in Wessex Mk 31B helicopters, originally serving with Royal Australian Navy.

Type

Free-turbine axial-flow turboshaft engine.

Intake

Radial type magnesium alloy casting consisting of two discs joined by six radial struts. Anti-icing by hot engine oil circulated through five of the struts and underside of casting, and by hot air through sixth strut. Intake carries row of hollow variable-incidence inlet guide vanes made of beryllium-copper sheet, heated by compressor delivery air.

Compressor

Eleven-stage axial-flow type. DTD 197A aluminium-bronze rotor blades secured by fir-tree roots to discs splined to shaft. Discs for stages 1, 6, 7, 8, 9, 10 and 11 of Rex 448 stainless steel, remainder are made of aluminium alloy. Rotor shaft of S.106 stainless steel carried in front roller and rear ball bearings. Two-piece casing cast in DTD 5003 (Mks 101, 161 and 162) or RR.450 (Mks 165 and 165C) magnesium alloy, carrying DTD 197A stator blades mounted in half-rings bolted to casing. Pressure ratio 5.9:1. Mass flow 7.71 kg (17.0 lb)/s at 19,100 compressor rpm.

Combustion Chamber

Integral type chamber casing, with six flame tubes of Nimonic 75. Manufactured by Lucas. One Simplex upstream-injection burner in each chamber. High-energy surface discharge igniter plugs in two chambers.

Turbine

Two independent axial turbines. Two-stage turbine drives compressor; single-stage turbine drives the reduction gear mounted behind it. First-stage turbine blade material is G.67. Third-stage blade material is Nimonic 90. Blades attached by fir-tree roots. Turbine disc material is Rex 448 (compressor turbine) and Rex 535 (power turbine). Compressor turbine shaft is S.106 stainless steel. Power turbine disc and shaft are integral. One row of fixed cast guide vanes for each turbine stage. Material is X40 or G.67, depending on stage and rating.

Jetpipe

Fixed quadrupled jetpipes of Nimonic 75.

Mounting

Three struts to pads on main support plate between compressor and combustion chambers. Engine torque reaction mounting on reduction casing.

Output

Epicyclic reduction gear, incorporating hydraulic torque meter, driven from free turbine. Ball and roller bearings. Ratio 7:1.

Accessories

Through spur gears off lower end of compressor shaft in intake casing. Drive passes through one of radial struts to gear train on upper side of casing, for fuel pump, oil pumps, centrifugal breather, tachometer and hydraulic pump. Bevel gears drive the generator of the Mk 165C or the CSD and alternator of the Mk 165.

Starting

Rotax combustion/air starter type CT.0821, mounted adjacent to compressor.

Control System

Hydromechanical type, compensated for altitude with partial-authority rotor speed governor. Fully variable inlet guide vanes are operated from a fuel pump signal. Manufactured by Lucas.

Fuel Specification

Normally DERD.2488, 2482, 2494 and 2486. In emergency, DERD.2485, DEF 2401-C and DEF 2402-B may be used, except on Mk 165C.

Oil System

Dry sump type, with gear-type pumps. Normal oil supply pressure 5.62 kg/cm^2 (80 lb/sq in). Oil is fuel cooled.

Oil Specification

DERD.2487.

Dimensions

Length overall	1,905 mm (75.0 in)
Inlet diameter	1,080 mm (42.5 in)

Weight, Dry

	401 kg (884 lb)

Performance Ratings

(S/L, ISA, nominal)

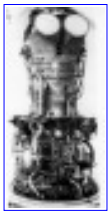
Intermediate, one-hour	1,193 kW (1,600 shp) flated rated, 2,570 output rpm
Max continuous	1,044 kW (1,400 shp)

Specific Fuel Consumption

One-hour rating	115.0 µg/J (0.68 lb/h/shp)

UPDATED

Early Gazelle 161



Gazelle 165C

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AERO-ENGINES - TURBOJET, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE DERWENT

This centrifugal turbojet was produced quickly in 1944 by scaling down the Nene. Small numbers are flying in surviving Meteors. The RD-500 was built without a licence in the Soviet Union, and was in turn produced under Soviet licence in China as the RD-500 and the RD-500K missile engine. Small numbers of both the Russian and Chinese versions remain operational. The following refers to the British Mk 8:

Type

Single-shaft turbojet.

Compressor

Single double-sided centrifugal rotor with diameter of 525 mm (24.5 in). Mass flow 29.7 kg (65.5 lb)/s. Pressure ratio 4.19. Rotor assembly supported on three bearings.

Combustion Chamber

Nine straight-flow combustion chambers with internal concentrically mounted flame tubes. Interconnectors between the chambers to equalise pressure and to ignite the fuel in adjoining tubes when starting up. Two torch igniters (Mk 8) or high-energy igniter plugs (in chambers 3 and 8).

Turbine

Single-stage axial-flow turbine with 54 blades. Direction of rotation anti-clockwise (viewed from rear).

Mounting

Two trunnions mounted on horizontal centreline of the compressor casing, and a torsionally-free diamond frame with link connections to two aircraft pick-up points.

Accessories

Engine accessories, including fuel and oil pumps, mounted on wheelcase at front of engine, together with an electric starter motor. Drive provided in wheelcase for accessory gearbox.

Control System

Engine-driven positive-displacement multiplunger swashplate pump, with built-in overspeed governor, draws fuel through a fabric-element low-pressure filter and delivers it to fixed orifice type burners, one in each combustion chamber, via a throttle control valve and ring manifold. Pump delivery pressure is variable and is controlled by a built-in servo system actuated by a separate barometric pressure control containing a capsule which is subjected to nacelle presssure.

Fuel Specification

Aviation Turbofuel (DERD.2482).

Oil System

Oil tank 12.5 litres capacity mounted on engine wheelcase. A gear-pump supplies pressure oil to main bearings and wheelcase. Front bearings drain into wheelcase, which is scavenged by upper unit of a twin geartype scavenge pump. Lower unit scavenges directly the centre and rear main bearings. Combined delivery from twin scavenge pump is discharged into the tank. No oil cooler fitted.

Oil Specification

DERD.2490.

Dimensions

Length	2,108 mm (83.1 in)
Diameter	1,092 mm (43.0 in)

Weight, Dry

	580 kg (1,280 lb)

Performance Ratings

(S/L, ISA)

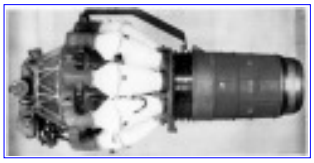
T-O	16.01 kN (3,600 lb st) at 14,700 rpm
Max continuous	13.74 kN (3,090 lb st) at 14,100 rpm

Specific Fuel Consumption

T-O, as above	29.18 mg/Ns (1.03 lb/h/lb st)

VERIFIED

Derwent 8



Derwent 5 (shared post-war Meteor with Mk 8)

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AERO-ENGINES - TURBOPROP, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE DART

The design of this pioneer turboprop began in 1945, and the first flight-cleared engine flew at a rating of 559 kW (750 shp) in October 1947. Subsequently this simple and robust single-shaft engine was developed to give powers up to 2,460 kW (3,300 shp). Production ended in 1987 with 7,100 engines delivered. They have flown 149 million hours. Rolls-Royce underestimated the size of the future market for turboprops in this power class, and so failed to produce a successor. The following are current versions:

Dart 21 (RDa.7)

Mass flow 9.75 kg (21.5 lb)/s. Pressure ratio 5.4. Gear ratio 0.093. Powers Alizé.

Dart 525 (RDa.7)

Mass flow 10.66 kg (23.5 lb)/s. Pressure ratio 5.6. Gear ratio 0.093. Powers Viscount.

Dart 529 (RDa.7)

Gear ratio still 0.093 but larger (3.5 m; 11 ft 6 in) propeller. Powers FH-227 and Gulfstream I.

Dart 536-2 (RDa.7)

Gear ratio 0.086. Powers 748.

Dart 536-7R (RDa.7)

Gear ratio 0.086. Powers F27.

Dart 552 (RDa.7)

Improved compressor and turbine aerodynamics to upgrade existing Mk 536 engines.

Dart 542 (RDa.10)

Mass flow 12.02 kg (26.5 lb)/s. Pressure ratio 6.35. Gear ratio 0.0775. Powers YS-11.

Dart 201 (RDa.12)

Water/methanol injection. Mass flow 12.25 kg (27.0 lb)/s. Gear ratio 0.0775. Powers Andover C.1.

Type

Single-shaft centrifugal-flow turboprop engine.

Intake

Circular intake with annular duct leading to impeller eye of first-stage compressor. Oil tank around intake is cast integral with casing. Secondary air intake supplies air to oil cooler mounted on top of casing.

Compressor

Two-stage centrifugal-flow compressor. Each impeller has 19 vanes and steel rotating guide vanes.

Combustion Chamber

Seven straight-flow combustion chambers. Flame tubes with fuel atomisers in front end of each tube for downstream injection. High-energy igniter plugs in Nos 3 and 7 chambers.

Turbine

Three axial stages. First and second discs are bolted together by five bolts and all three by further five, while the drive shaft is divided, with inner shaft connecting turbine with reduction gear and outer shaft with compressor. All blades of Nimonic alloy and secured on discs by fir-tree roots.

Jetpipe

Propeller thrust line coaxial with engine main shaft but exhaust unit has a slight inclination to suit installation. Unit comprises an outer shell which supports an inner cone on three struts enclosed in aerofoil-section fairings to reduce turbulence and straighten gas flow at nozzle. Maximum jetpipe temperature 650~C.

Mounting

Four feet are provided at 90~ on the horizontal and vertical centrelines of compressor casing, although only three need be used. Bottom foot for hot-air gate valve. No rear mounting is required, but jetpipe if used requires separate mounting in airframe.

Output

Double reduction gearing with helical high-speed train and final helical gear drive. The two gear trains connected by three layshafts. High-speed pinion driven by an inner shaft system bolted directly onto turbine discs. All gears and propeller shaft carried in roller or ball bearings. Bevel gears from one of the layshafts provide drives to fuel and oil pumps and propeller controller unit. Bevel gear and engaging mechanism on pinion shaft provide drive from starter motor.

Accessories

An accessory gearbox drive is taken from the main-shaft centre-coupling immediately behind the compressor through a train of gears to a housing on top of the intermediate casing.

Control System

Single multiplunger variable-stroke injection pump delivers fuel to burners through flow control unit, which incorporates a filter, throttle valve, shut-off cock and barometric pressure control. Operation of control unit is function of intake pressure and throttle valve pressure drop, thus determining fuel/air ratio for all engine operating conditions. RDa.10 and 12 have duple fuel system, with pressures of 22.85 kg/cm^2 (325 lb/sq in) at idle and 63.28 kg/cm^2 (900 lb/sq in) at T-O. In all other versions, fuel pressure at burners varies from 2.81 kg/cm^2 (40 lb/sq in) at idling speed to 70.3 kg/cm^2 (1,200 lb/sq in) at maximum power. Automatically progressive injection of water/methanol used to restore take-off power under high ambient temperature conditions. System interconnected mechanically with throttle lever to ensure that it can only be used at take-off rpm. Fuel filter de-icing by hot air from compressor. Hot-air gate valve fitted to bottom engine mounting. Only two cockpit controls, a throttle lever for varying power and a high-pressure cock for stopping engine. Throttle valve is interconnected with the propeller controller and high-pressure cock is linked with propeller feathering controls. Blades may be feathered by moving shut-off cock lever past the closed position; depression of an unfeathering button returns blades to fine pitch. Certain Viscount aircraft feature automatic selection into zero pitch, available with the aircraft weight on the landing gear. All other aircraft feature automatic cancellation of ability to come below flight fine pitch when the throttles are advanced to maximum power with gust locks removed.

Oil System

Entirely self-contained. Integral oil tank (total capacity 14 litres; 24 Imp pints) feeds engine via standpipe and feathering pump through tank base, to ensure feathering possible even after prolonged system oil leak. Gear pump supplies oil to all bearings and reduction-gear jets at nominal pressure of 2.10 kg/cm^2 (30 lb/sq in) and at nominal flow of 2,091 litres (552 US gallons; 460 Imp gallons) per hour. Combined delivery from four scavenge pumps returned to tank via oil-cooler on top of intake casing. Pressure and scavenge pumps in single housing and driven by common shaft.

Dimensions

Length, without jetpipe:	
Mk 21	2,500 mm (98.4 in)
Mks 525, 529, 536, 552	2,480 mm (97.6 in)

Mks 542, 201	2,527 mm (99.49 in)
Diameter (all)	963 mm (37.9 in)

Weight, Dry

Mk 21	575 kg (1,268 lb)
Mks 525, 529	560 kg (1,235 lb)
Mks 536, 552	561 kg (1,237 lb)
Mk 542	628 kg (1,384 lb)
Mk 201	675 kg (1,488 lb)

Performance Ratings

(T-O, S/L, ISA)

Mk 21	1,425 kW (1,910 shp)
Mk 525	1,342 kW (1,800 shp)
Mk 529	1,484 kW (1,990 shp)
Mk 536-2	1,551 kW (2,080 shp)
Mks 536-7R, 552	1,581 kW (2,120 shp)
Mk 542	2,051 kW (2,750 shp)
Mk 201	2,215 kW (2,970 shp)

Specific Fuel Consumption

Cruise, 6,096 m; 20,000 ft:	
Mks 525-536	97.68 Ág/J (0.578 lb/h/ehp)
Mks 542, 201	93.96 Ág/J (0.556 lb/h/ehp)

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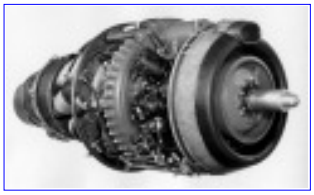
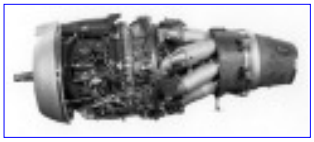


Dart 536

Dart 552 in F27



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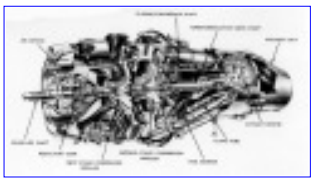

Cutaway drawing of Dart 525 or 536

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AERO-ENGINES - TURBOJET, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE AVON (MILITARY)

First run in 1946, this axial-compressor turbojet took many years to develop to an acceptable standard of handling and reliability. A significant number remain in service in many countries. Early 100-series engines have a 12-stage compressor and eight separate tubular combustion chambers. The more powerful 200- and 300-series have a can-annular combustion chamber and a compressor with 15 stages (200 series) or 16 stages (300).

Military Avon engines were made by Rolls-Royce, Bristol, Napier, Standard Motors, Commonwealth Aircraft (Australia) and Svenska Flygmotor (Sweden). A very few, including the Swedish-made RM6B (Lansen) and RM6C (Draken), have afterburners. Surviving Hunters have both 100 and 200 series engines. Canberras have 100-series, except for the PR.9, the version still operational with the RAF, which has the 200 series. Lightnings, flying privately in South Africa, have the 300 series.

Type

Axial-flow turbojet.

Intake

Annular casing surrounding starter bullet. Variable-incidence inlet guide vanes of DTD.171 weldable chrome-nickel austenitic steel, automatically-operated by hydraulic ram. Magnesium intake casing incorporates front bearing housing to which is bolted the starter reduction gear housing. All forward-facing surfaces have compressor bleed anti-icing.

Compressor

Single axial spool, (100 series) 12 stages, (200 series) 15 stages, (300 series) 16 stages. Rotor blades pinned to rotor discs which are splined to shaft. Discs of 100 series aluminium alloy, 200 and 300 series Rex 448 or Jessops

H 46 stainless steel. Rotor and matching stator blades of aluminium alloy (stages 1-8), stainless steel (stages 9-11) and titanium alloy (stages 12-15). First four stages of stator blades are tip-shrouded. Compressor shaft, on two bearings, one roller (front) and one ball (rear), in two parts with flanged joint forming part of 15th stage disc. Compressor casing in two parts, the forward casing, of cast magnesium, covering first seven compressor stages and rear casing of aluminium alloy. Cast compressor outlet casing directs air into eight ducts, hollow webs between each duct collecting hot compressor air for anti-icing and cabin pressurisation. Typical mass flow, (100 series) 56 kg (123.5 lb)/s, (200 series) 68.6 kg (151 lb)/s, (300 series) 77.1 kg (170 lb)/s. Pressure ratio, (100) 6.5, (200) 7.48, (300) 8.43.

Combustion Chamber

100 series, eight separate chambers; 200 and 300 series, eight flame tubes mounted within annular chamber, each fitted with Rolls-Royce duplex burner. Interconnecting pipes between each pair of flame tubes. High-energy igniters in Nos 3 and 6 tubes. Flame tubes of Nimonic 75. Double-walled combustion system casing of Fortiweld steel.

Turbine

Two-stage. Turbine-shaft coupled to compressor shaft by ball and socket coupling with splined sleeve to transmit torque and supported at rear end by roller bearing. High-pressure blades in 100 series are of Nimonic 95A, low-pressure blades of Nimonic 80A, secured to steel discs by fir-tree roots and locked; 200 and 300 series, higher-grade Nimonic alloys. The Mk 210 (RA.24R) was the world's first production engine with air-cooled rotor blades (1956). All blades shrouded at tips. Nozzle casing, of cast chrome steel, has two rows of guide vanes, one ahead of each turbine stage. All vanes of C.242 nickel-base alloy.

Jetpipe

Of Nimonic 75, flange-bolted to nozzle casing. Exhaust cone carried on four faired radial struts. Insulating material in space between two walls of outer casing. The RM6B (see under Volvo) has a Svenska Flygmotor afterburner, which in the RM6C is improved and fitted with a different design of nozzle. The Avon 302C, fitted to the Lightning, has an afterburner with hot-streak ignition, and a multiflap nozzle driven by screwjacks.

Accessories

Horizontal shaft drive to each side of engine from gear system, driven by main shaft from just aft of centre bearing. External wheel case on starboard side serves lubrication and fuel pump systems, drive on port side serves, through bevel drive, generators or alternators or ancillary gearbox mounted on airframe.

Starting

(100 series) triple-breech BTH cartridge or Rotax electric or iso-propyl liquid fuel. (200 series) electric starter motor or iso-propyl gas starter or kerosene/air turbo-starter, (300 series) Avpin (iso-propyl), all mounted in bullet in air intake.

Control System

(100 series) Low-pressure, Lucas-Twin-D pump and barometric control unit. (200 and 300 series) Controlled by variations of flow rather than variations of pressure. Proportion of main engine flow is fed from upstream of the throttle valve and bypassed to the pump outlet. This forms the proportional circuit which controls main flow under varying conditions of altitude, jetpipe temperature and engine speed.

Fuel Specification

DERD.2482 or 2486.

Oil System

Closed circuit system with combined oil tank and sump under compressor casing and fuel-cooled oil cooler on port side. Lubrication of three main bearings, starter reduction gear and ancilliary drive system by single pump.

Dimensions

Length overall:	
Mk 115	2,593 mm (102.1 in)
Mk 207	3,124 mm (123.0 in)
Mk 302C	3,505 mm (138.0 in)
Diameter:	
Mk 115	1,072 mm (42.2 in)
Mk 207	1,054 mm (41.5 in)
Mk 302C	1,118 mm (44.0 in)

Weight, Dry

Mk 115	1,143 kg (2,520 lb)
Mk 207	1,301 kg (2,869 lb)
Mk 302C	1,360 kg (2,998 lb)

Performance Ratings

(T-O, S/L, ISA)

Mk 115	35.8 kN (8,050 lb st) at 8,100 rpm
Mk 207	45.13 kN (10,150 lb st)
Mk 302C, afterburner	69.72 kN (15,680 lb st)
RM6C, afterburner	76.08 kN (17,110 lb st)
Mk 302C, dry	56.43 kN (12,690 lb st)

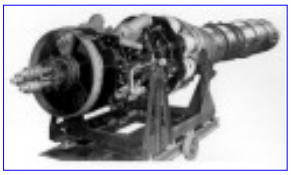
Specific Fuel Consumption

T-O as above:	
Mk 115	27.75 mg/Ns (0.98 lb/h/lb st)
Mk 207	24.64 mg/Ns (0.87 lb/h/lb st)
Mk 302C, dry	22.94 mg/Ns (0.81 lb/h/lb st)

UPDATED

Avon 200 series (RA.24)

Avon 300 series



Avon 100-series with afterburner and tripe-cartridge starter

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AERO-ENGINES - TURBOJET, UNITED KINGDOM

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE plc

ROLLS-ROYCE AVON (CIVIL)

In order to power the Comet 4 and Caravelle jetliners a family of civil Avon engines was developed from 1954. An exceptionally efficient three-stage turbine was designed, driving a compressor with a zero stage to handle increased air flow with higher pressure ratio. By 1960 the RA.29/6 had been developed with a `zero-zero' compressor stage, and the RA.29/6 Mk 533R is still in service with Caravelle VI aircraft. The following refers to this version.

Type

Single-shaft axial-flow turbojet.

Intake

Annular casing surrounding starter bullet. Variable-incidence inlet guide vanes automatically operated by hydraulic ram. Magnesium alloy intake casing incorporates front bearing housing supported on six tangential struts. Starter motor and reduction gear are mounted in centre of air intake ahead of front bearing. Inlet guide vanes and starter fairing have compressor bleed anti-icing.

Compressor

Seventeen-stage axial-flow. Rotor blades pinned to stainless steel discs which are splined to shaft. Stator blades of stainless steel (Stages 0 and 5-8), aluminium alloy (stages 1-4) and titanium or steel (stages

9-17). Rotor blades of aluminium alloy (stages 0-2) and titanium (stages 3-17). First five stages of stator blades (stages 0-4) are tip-shrouded. Compressor shaft mounted on roller bearing at front and ball bearing (rear). Compressor casing in two parts (front part magnesium, rear part aluminium), each split axially into two halves. Last five stages of compressor are housed in fabricated steel casing integral with compressor outlet casing, which directs air into eight ducts, hollow webs between each duct collecting hot compressor air for anti-icing and cabin pressurisation. Mass flow 83.92 kg (185.0 lb)/s. Pressure ratio 10.33.

Combustion Chamber

Eight flame tubes of Nimonic 75 mounted within annular chamber. Combustion system casing of Fortiweld steel.

Turbine

Three-stage. Turbine shaft, supported at rear end by roller bearing, connected to compressor shaft by screw coupling. Turbine blades of Nimonic alloys, all shrouded at tip and secured to steel discs by fir-tree roots. Nozzle box, of cast chrome steel, carries three rows of guide vanes, one ahead of each turbine stage. All vanes of C.242 cast nickel base alloy.

Jetpipe

Of nickel chromium steel, with exhaust cone carried on three faired radial struts. Insulating material in space between two walls of outer casing. Greatrex noise-suppressing multilobe nozzle, incorporating hinged flap to vary nozzle area for greater cruise fuel efficiency.

Accessories

Horizontal shaft drive to each side of engine from gear system driven by main shaft from just aft of centre bearing. External wheelcase on starboard side serves lubrication and fuel pump systems; drive on port side serves generator or ancillary gearbox according to installation.

Starting

By electric starter motor, mounted in bullet in air intake.

Control System

Controlled by variations of flow rather than variations of pressure. A proportion of the main engine flow is fed from upstream of the throttle valve and bypassed to the pump outlet. This forms the proportional circuit which controls the main flow under varying conditions of altitude, jetpipe temperature and engine speed.

Oil System

Closed circuit system with fuel-cooled oil cooler. Combined oil tank and sump under rear compressor casing. Lubrication of three main bearings, starter reduction gear and ancillary drive system by single pump. Five pumps scavenge bearing and drives.

Dimensions

Length	3,404 mm (134.0 in)
Diameter	1,067 mm (42.0 in)

Weight, Dry

	1,583 kg (3,490 lb)

Performance Rating

(T-O, S/L, ISA)

	56.03 kN (12,600 lb st)

Specific Fuel Consumption

(T-O, as above)

	22.23 mg/Ns (0.785 lb/h/lb st)

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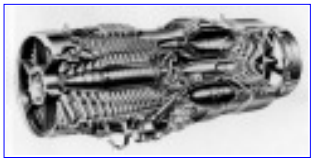
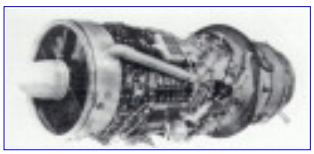


Cutaway drawing of RA.29/1 (16-stage compressor)



RA.29/6

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AERO-ENGINES - TURBOFAN, UNITED KINGDOM

Date Posted: 17 September 1999

Jane's Aero-Engines 06

ROLLS-ROYCE plc

ROLLS-ROYCE ALLISON TF41

See the International section.

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AERO-ENGINES - TURBOSHAFT, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOLOY CORPORATION

SOLOY ALLSTAR

In the mid-1980s Soloy developed and certificated a conversion package for the Eurocopter AS 350 AStar helicopter with the Allison (now Rolls-Royce) 250-C30M. The STC (Supplemental Type Certificate) was subsequently sold to Rocky Mountain Helicopters, where this conversion was known as the AllStar. Following close collaboration with Rolls-Royce North America, Soloy announced in January 2000 that it was re-entering the helicopter retrofit market with **Soloy AllStar**. This is the AStar AS 350D, B or BA retrofitted with the Rolls-Royce 250-C30M (see Rolls-Royce in this section). The Soloy STC is applicable to all AS350D, B and BA models.

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL INC - HONEYWELL INC

Chairman and CEO: Lawrence A Bossidy

In June 1999, AlliedSignal Inc and Honeywell Inc announced a definitive merger. Both are extremely strong in aerospace, as well as other fields of advanced technology. For the purposes of this product the key player is Honeywell Engines, described below. Honeywell is a world leader in control systems. In 1998, AlliedSignal employed 70,400 and had revenues of US$15.1 billion. Honeywell employed 57,000 and had 1998 revenues of US$8.4 billion.

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President and CEO: Robert D Johnson
Vice-President, Communications: Dennis Signorovich

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Fax: (+1 310) 512 24 90
e-mail: dennis.signorovich@honeywell.com
Web: http://www.honeywell.com

This is the largest single segment of the new Honeywell company, with annual revenues of about US$10 billion.

HONEYWELL ENGINES, SYSTEMS & SERVICES

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President, Engines, Systems & Services: Steve Loranger
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Programme Manager: David Moniz

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e-mail: davidf.moniz@honeywell.com

This company manages the plant at the above address making engines for regional and business aircraft and helicopters. Honeywell has important military helicopter business, and is a partner in ITEC's F124 fighter/trainer engine.

Current production engines have designations based on such names as Lycoming Fan or Turbo Prop Engine. Future engines are likely to have designations with the corporate prefix AS. By March 2000 there was no intention to change engine designations.

The Phoenix company was, in May 1998, awarded a cost-share contract (US$15,239,000 DoD, US$3,311,000 contractor) for the Joint Turbine Advanced Gas Generator (JTAGG) III. This is a joint Army/USAF search for new technologies for small and medium-size engines. Expected completion date is January 2004.

Significant growth is forecast for flexible lease arrangements for propulsion engines and APUs (auxiliary power units). In mid-1998, the company had a rotatable pool of 700 engines and 400 APUs as part of its OEM service to customers.

It is also a partner in two collaborative enterprises, LHTEC which produces the T800 helicopter engine and the CFE Co which produces the CFE738 turbofan, both of which are listed alphabetically in this section.

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

TELEDYNE CONTINENTAL MOTORS - TCM TURBINE ENGINES

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Vice-President and General Manager: Mark Celusniak

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TCM Turbine Engines is a subsidiary of Teledyne Continental Motors, itself a business unit of Teledyne Technologies of Los Angeles. It produces small gas turbine engines for training aircraft, cruise missiles and UAVs. Its engine for manned aircraft is the J69-T-25A.

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

GE AIRCRAFT ENGINES

GENERAL ELECTRIC CF6-80C2

US Military designation: F103-GE-102

This engine is a major redesign for higher thrust and improved sfc, based on the CF6-80A1/A3 but with a 2,362 mm (93 in) diameter fan. It has a four-stage LP compressor and LP turbine redesigned aerodynamically with 5.5 stages. The first CF6-80C2 ran in May 1982 and exceeded 276 kN (62,000 lb st) corrected thrust. Flight test on an A300 took place between August and December 1984, leading to certification on 28 June 1985. The engine entered revenue service on 5 October 1985.

Programme sharing agreements were signed with SNECMA of France, MTU of Germany, Volvo Aero of Sweden and FiatAvio of Italy. The following versions have been announced:

CF6-80C2A1

Rated at 257.4 kN (57,860 lb st) to 30ºC. Powers A300-600.

CF6-80C2A2

Rated at 233.4 kN (52,460 lb st). Powers A310-200Adv and A310-300.

CF6-80C2A3

Rated at 262.0 kN (58,950 lb st) to 30ºC. Powers A300-600.

CF6-80C2A5

Rated at 267.3 kN (60,100 lb st) to 30ºC. Powers A300-600R.

CF6-80C2A5F

Rated at 267.3 kN (60,100 lb st) to 30ºC. Powers A300-600F.

CF6-80C2A8

Rated at 257.4 kN (57,860 lb st) to 35ºC. Powers A310-300 and SATIC A300-600ST.

CF6-80C2B1

Rated at 249.0 kN (55,980 lb st) to 30ºC. Powers 747-200 and -300.

CF6-80C2B1F

Rated at 254.3 kN (57,160 lb st) to 32.2ºC. Powers 747-400.

F103-GE-102

Military version, rated at 252.3 kN (56,700 lb st). Powers VC-25A (Air Force One); in late 1999 engine flight hours 18,332 with no IFSD or unscheduled SVR. Selected to power USAF AL-1A (Airborne Laser) anti-missile defence aircraft. In 2000 the use of this engine to re-engine the USAF C-5 Galaxy was examined, and this led in February 2002 to a US$126 million contract for 14 engines to support a six-year System Development and Demonstration phase of the C-5 Reliability Enhancement and Re-engining Program (RERP). These engines will support a three-aircraft test programme due to begin in 2005, and options exist for four additional engines for an additional aircraft. The RERP is planned to provide up to an additional 490 engines to keep the USAF C-5 fleet viable until 2040.

CF6-80C2B1F1, B1F2

Rated at 267.0 kN (60,030 lb st) to 30ºC. Powers 747-400.

CF6-80C2B2

Rated at 229.5 kN (51,590 lb st) to 32.2ºC. Powers 767-200, -200ER.

CF6-80C2B2F

Rated at 231.4 kN (52,010 lb st) to 32.2ºC. Powers 767-300ER.

CF6-80C2B3F

Rated at 231.4 kN (52,010 lb st) to 32.2ºC. Powers 747-400D.

CF6-80C2B4

Rated at 254.4 kN (57,180 lb st) to 32.2ºC. Powers 767-300, -300ER.

CF6-80C2B4F

Rated at 254.8 kN (57,280 lb st) to 32.2ºC. Powers 767-300ER.

CF6-80C2B5F

Rated at 267.0 kN (60,030 lb st) to 30ºC. Powers 747-400.

CF6-80C2B6

Rated at 267.2 kN (60,070 lb st) to 30ºC. Powers 767-300ER.

CF6-80C2B6F, B7F

Rated at 267.0 kN (60,030 lb st) to 30ºC. Powers 767-300ER and -300F.

CF6-80C2B6FA

Rated at 267.0 kN (60,030 lb st) to 30ºC, driving two 150 kVA alternators. Powers 767 AWACS. Though military aircraft they are exports to Japan, so the engines do not have US military F103 designations.

CF6-80C2B7F

See note below under HP TURBINE. This turbine module has been overtaken by the B8F, see next.

CF6-80C2B8F

Rated in the 280.2 kN (63,000 lb st) class. In June 1999 this was the latest and highest-thrust member of the CF6-80C2 family. It incorporates the most advanced standard of HP turbine, with GE90-type architecture, R88 rotors and improved cooling. Certificated in March 1999, with EIS on 767-400ER in May 2000 with Delta and July 2000 with Continental. The new turbine module can be retrofitted to all previous CF6-80C2 engines.

CF6-80C2D1F

Rated at 270 kN (60,690 lb st) to 30ºC. Powers MD-11.

By 30 April 1999, a total of 761 aircraft was in service, powered by 2,500 CF6-80C2 engines. These had then flown 48.6 million hours, with the high-time engine on 37,908 hours and the high-cycle engine

having accumulated 17,496 cycles. Engine-caused SVR was 0.099, equivalent to one per 10,101 hours. Dispatch reliability was 99.96 per cent, and IFSD 0.005 (one per 200,000 hours). All versions have been granted FAA 180-minutes ETOPS on the 767; the DGAC has granted 180-min ETOPS on the A300 and A310, and the Japanese CAB has granted 120-minutes on the 767. In November 1997, an engine was removed from a 767-200ER of US Airways after 28,747 hours; in March 1998, the other engine had logged 30,015 hours.

By June 1999 the CF6 family was maintaining its announced reputation as ``the world's most reliable family of aircraft engines''. In service with over 150 customers, the overall CF6 family had then logged more than 180 million flight hours. Overall dispatch reliability was 99.94 per cent, and IFSD rate 0.008.

The CF6-80C2 differs from earlier CF6 engines in the following features:

Fan

Single stage, with 38 blades and four-stage booster (LP compressor). Mainly titanium except for steel mid-fan shaft, aluminium spinner and blade-containment shroud of layers of Kevlar around aluminium case. Eighty composite exit guide vanes canted for better aerodynamic efficiency. Mass flow, typically 802 kg (1,769 lb)/s; bypass ratio 4.98 to 5.06.

LP Compressor

Four stages with blades and vanes mounted orthogonally, with dovetail offset from centre of pressure to reduce bending.

HP Compressor

Fourteen-stage, with inlet guide vanes and first five stator rows with variable incidence. Blades in stages 1-5 titanium, 6-14 steel; vanes all steel. One-piece steel casing with insulated aft stages. Core air flow typically 154 kg (340 lb)/s. OPR: C2A1, C2A8, 30.4; C2A2, 27.8; C2A3, C2B6, 31.1; C2A5, C2A5F, 31.5; C2B1, 29.3; C2B1F, C2B4, C2B4F, 29.9; C2B2, 27.1; C2B2F, 27.4; C2B5F, C2B6, 31.1; C2B6F, C2B6FA, C2B7, 31.4; C2D1F, 31.8.

Combustion Chamber

Annular, rolled ring construction, aft-mounted with film cooling. A modified combustor, initially adopted for 767-300ER freighters of UPS, reduces HC emissions to 12 per cent of ICAO standards, CO to 22 per cent and NO_x to 49 per cent.

HP Turbine

Two-stage. Stage one blades directionally solidified. Casing with active and passive clearance control. No mid-frame. Short blade life, especially in Middle East environments, has been countered by progressive improvements in material. In 1997, the P12 blade had virtually eliminated failures with Kuwait and UAE. Meanwhile, in a major long-term programme, a new boltless turbine is being made available, with blades of René 88. Eliminating the bolt features from the disc rims of both stages offers better durability, reduced maintenance cost, higher temperature capability and up to 2 per cent higher thrust. The new turbine will be available from May 2000, initially on the CF6-80C2B7F. Later, it will be applicable to all models of CF6-80C2 and CF6-80E1.

LP Turbine

Five stages, with cambered struts in rear frame to reduce exit swirl, effectively producing another half

stage. Rear hub heated by exhaust gas to reduce thermal stress.

Accessories

Drive enhanced on CF6-80C2B6FA to accommodate two 150 kVA alternators.

Control System

FADEC or hydromechanical fan speed control with electronic supervision; one throttle position corresponds to each engine rating in all flight conditions.

Dimensions

Length (all)	4,267 mm (168.0 in)
Diameter (all)	2,692 mm (106.0 in)

Weight, Dry

CF6-80C2A1, C2A2, C2A3, C2A5, C2A8	4,144 kg (9,135 lb)
CF6-80C2A5F, C2D1F	4,370 kg (9,634 lb)
CF6-80C2B1, C2B2, C2B4, C2B6	4,263 kg (9,399 lb)
CF6-80C2B1F, C2B2F, C2B4F, C2B5F, C2B6F, C2B7	4,309 kg (9,499 lb)
CF6-80C2B6FA	4,355 kg (9,602 lb)

Performance Ratings

T-O, real nozzle: see model listing	
Cruise (10,670 m; 35,000 ft; at M0.85; uninstalled, ideal nozzle):	typically 50.4 kN (11,330 lb)

Specific Fuel Consumption

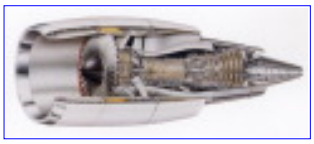
(T-O, as in model list)

CF6-80C2A1, C2B6	9.46 mg/Ns (0.334 lb/h/lb st)
CF6-80C2A2, C2B4F	8.98 mg/Ns (0.317 lb/h/lb st)
CF6-80C2A3	9.32 mg/Ns (0.329 lb/h/lb st)
CF6-80C2A5, A5F	9.63 mg/Ns (0.340 lb/h/lb st)
CF6-80C2A8	9.74 mg/Ns (0.344 lb/h/lb st)
CF6-80C2B1, B5F, B6F, B6FA	9.15 mg/Ns (0.323 lb/h/lb st)
CF6-80C2B1F	8.95 mg/Ns (0.316 lb/h/lb st)
CF6-80C2B2	9.01 mg/Ns (0.318 lb/h/lb st)
CF6-80C2B2F	8.70 mg/Ns (0.307 lb/h/lb st)
CF6-80C2B4	9.235 mg/Ns (0.326 lb/h/lb st)
CF6-80C22D1F	9.12 mg/Ns (0.322 lb/h/lb st)

Contract Price

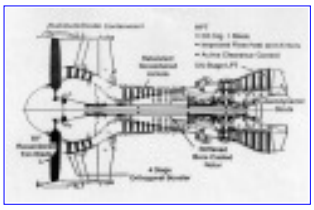
April 1997, engines for 10 767-300ER and 21 767-400ER, US$500 million; December 1998, engines for 10 767-300ER, US$160 million; December 1999, GECAS order for engines for 15 767 and five 747, ``more than US$500 million".

UPDATED



CF6-80C2 propulsion system

CF6-80C2



Features of CF6-80C2

CF6-80C2B8F (1999)

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AERO-ENGINES - TURBOPROP, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL

HONEYWELL TPE331

US military designation: T76
Based upon experience with APUs, this was the first Garrett AiResearch (later AlliedSignal, now Honeywell) engine for aircraft propulsion. Deliveries of all versions exceed 12,800, with flight time in October 2001 of approximately 97 million hours.

The following are major versions:

TPE331 series I, II

FAA certificated in February 1965. Rated at 451 ekW; 429 kW plus 0.33 kN (605 ehp; 575 shp plus 75 lb st). Redesignated **TPE331-25/61** and **-25/71** and produced until 1970. Inlet below gearbox. Powers MU-2 (A to E models), Super Turbo 18, FU-24, Pilatus Porter, Carstedt Jet Liner, Hawk Commander and 680, and Turbo Beaver.

TPE331-1 series

Certificated December 1967 at 526 ekW; 496 kW plus 0.44 kN (705 ehp; 665 shp plus 100 lb st). Inlet below gearbox. Powers MU-2 (F and G), Turbo-Porter and AU-23A Peacemaker, CJ600, Turboliner, Interceptor 400, Turbo Commander and (customer option) Thrush Commander, Merlin IIB and Fletcher 1284, Turbo Thrush and Turbo Ag-Cat, Pilatus Porter and Jet Liner 600.

TPE331-2 series

Certificated in December 1967 at 563 ekW; 533 kW plus 0.45 kN (755 ehp; 715 shp plus 102 lb st). Inlet below gearbox. Powers Skyvan, CASA 212 pre-series, Turbo Goose and Turbo Beaver.

TPE331-3 series

Certificated in March 1969 at 674 ekW; 626 kW plus 0.71 kN (904 ehp; 840 shp plus 159 lb st). Uprated gas generator with increased airflow and pressure ratio, but same turbine temperature as in original TPE331. Inlet above gearbox. Powers Merlin III, IV and Metro, and Jetstream III.

TPE331-5/6 series

The -5 was certificated in May 1970; this matches the gas generator of the -3 with the 715 shp gearbox, and is flat rated at 578.7 kW (776 shp) to 2,135 m (7,000 ft). Inlet below gearbox. Powers MU-2, King Air B100 (-6), CASA 212 (-5), Merlin IIB, Shorts Skyvan, Dornier 228 (-5), and Commander 840/900. The -5 designation indicates an output speed of 1,591 rpm; the -6, rated at 533 kW (715 shp) has an output speed of 2,000 rpm.

TPE331-8

Matches compressor and gearbox of -3 with new turbine section. Thermodynamic power of 676 ekW; 645 kW (905 ehp; 865 shp) plus 0.47 kN (105 lb st), but flat-rated at 533 kW (715 shp) to 36ºC. Certification was received in November 1976. Powers Conquest II.

TPE331-9

Thermodynamic rating 645 kW (865 shp).

TPE331-10

Rated at 746 kW (1,000 shp). Certificated January 1978. Inlet below gearbox. Powers Marquise and Solitaire, Commander 980/1000, Do 228, Dash-10 Conquest and CASA 212-200 and -300.

TPE331-10T

Flat rated at 522 kW (700 shp). Powers Predator B (NASA UAV).

TPE331-10U

As Dash-10 but inlet above gearbox. Powers Merlin IIIB and IIIC and Jetstream 31.

TPE331-11U

Certificated 1979. Inlet above gearbox. Higher gearbox limit; wet rating 820 kW (1,100 shp). Powers Metro III and Metro 23.

TPE331-12U

Same size as -10U but offers 834 kW (1,120 shp). Certificated December 1984. Inlet above gearbox. Powers Jetstream Super 31, CASA 212-400, Metro III and Metro 23. The **TPE331-12B**, inlet below gearbox, powers the Shorts Tucano. Rolls-Royce made 30 per cent by value of engines for the Tucano, and supports RAF engines.

TPE331-14/15

Scaled-up models, with thermodynamic power in the 1,227 kW (1,645 shp) class. Inlet below gearbox. The -14 was certificated in April 1984 and is flat rated at 746 kW (1,000 shp) for the Cheyenne 400 and Grob High Flyer (Egrett II, Strato I). TPE331-15AW powers one version of re-engined S-2 Tracker.

TPE331-14GR/HR

Inlet below gearbox. Handed (clockwise/anti-clockwise) for Jetstream 41, An-38, Marsh 52 Air Tractor and Albatross conversion. T-O rating 1,312 kW (1,759 shp); APR rating 1,462 kW (1,960 shp). For the An-38, Honeywell is supplying the complete propulsion package, which received Russian certification on 2 September 1996. In September 1998, following a sampling programme, TBO was extended on Jetstream 41 engines from 7,000 to 9,000 flight hours, with HSI extended from 3,500 to 4,500.

TPE331-25

For ag-aircraft, T-O rating typically 533 kW (715 shp); option on Dromader, Air Tractor, Turbo Ag-Cat and Turbo Thrush.

T76

Military engine, with gas generator similar to TPE331-1 but with front end inverted, to give inlet above gearbox. All models power OV-10 Bronco.

Except for the TPE331-14/15, all versions are of similar frame size, and the following data apply generally to all:

Type

Single-shaft turboprop.

Compressor

Tandem two-stage centrifugal made from titanium. Mass flow, 2.61 kg (5.78 lb)/s for -25/61, -25/71; 2.81 kg (6.2 lb)/s for -1; 2.80 kg (6.17 lb)/s for -2 and T76; 3.52 kg (7.75 lb)/s for -5 and -6; 3.54 kg (7.8 lb)/s for -3; 3.49 kg (7.7 lb)/s for -10U; 5.26 kg (11.6 lb)/s for -15AW. Pressure ratio 8.0 for -25/61, -25/71; 8.34 for -1; 8.54 for -2 and T76; 10.37 for -5, -3 and -10U; 10.8 for -15AW.

Combustion Chamber

Annular, with capacitor-discharge igniter plug on turbine plenum.

Turbine

Three-stage axial. In early models, blades cast integrally with disc. In -10, -11 first-stage disc with inserted blades. Speed, -5, -6, -10, -101, -201, 41,730 rpm; -15, 34,904 rpm. In -14/-15 inserted blades in all three stages. In -12 first- and second-stage with inserted blades. Inlet gas temperature, 987ºC for -25/61, -25/71, 993ºC for T76, 1,013ºC, for -1, -2, -3, -5, -6, -8 and -9, 1,050ºC for -10, -11 and 1,220ºC for -12.

Output

Two-stage reduction gear, one spur and the second planetary, ratio usually 20.865 or 26.3. Propeller shaft speed (-5) 1,591 rpm, (-6, -8, -101, -128, -201) 2,000 rpm, (-10U, -11, -10) 1,591 counter-clockwise from rear, 2,000 clockwise, (-15) 1,390 rpm, (-14GR/HR) 1,552 rpm.

Accessories

AND 20005 Type XV-B tachometer generator, AND 20002 Type XII-D starter/generator, AND 20010 Type XX-A propeller governor and AND 20001 Type XI-B hydraulic pump.

Control System

(some -5/-6) Bendix, (-8, -10N, -12B) Honeywell, (all others) Woodward system. Maximum fuel pressure 41.4 bars (600 lb/sq in).

Fuel Specification

(TPE331): ASTM designation D1655-64T types Jet A, Jet B and Jet A-1; MIL-F-5616-1, Grade JP-1.

Oil System

Medium pressure dry sump system. Normal oil supply pressure (through -12) 6.90 bars (100 lb/sq in), (-14/-15) 4.14 bars (60 lb/sq in).

Oil Specification

MIL-L-23699-B or MIL-L-7808D.

Starting

Pad for starter/generator (see variant list).

Dimensions (approx)

Length overall:	
TPE331	1,092-1,333 mm (43-52.5 in)
T76	1,118 mm (44 in)
Width: TPE331	533 mm (21 in)
T76	483 mm (19 in)
Height: TPE331	660 mm (26 in)
T76	686 mm (27 in)

Weight, Dry

TPE331-25/61, 71	152 kg (335 lb)
TPE331-1, -2	152.5 kg (336 lb)

T76	155 kg (341 lb)
TPE331-3	160 kg (353 lb)
TPE331-5, -6	163 kg (360 lb)
TPE331-8	168 kg (370 lb)
TPE331-10	175 kg (385 lb)
TPE331-11, -12	184 kg (405 lb)
TPE331-14/-14GR	281 kg (620 lb)
TPE331-15AW	299 kg (659 lb)

Performance Ratings

T-O: see under model listings	
Military (30 min): T76-G-410/411	533 kW; 563 ekW (715 shp; 755 ehp)
Normal: T76-G-410/411	485 kW; 514.5 ekW (650 shp; 690 ehp)
Max cruise (ISA, 3,050 m; 10,000 ft and 250 kts; 463 km/h; 288 mph):	
TPE331-25/61, 71	332 kW (445 shp)
TPE331-1	404 kW (542 shp)
TPE331-2, T76	430 kW (577 shp)
TPE331-3, -5	530 kW (710 shp)

Specific Fuel Consumption

At T-O rating:	
TPE331-25/61, 71	111.5 µg/J (0.660 lb/h/shp)
TPE331-1	102.2 µg/J (0.605 lb/h/shp)
TPE331-2	99.4 µg/J (0.588 lb/h/shp)
TPE331-3	99.7 µg/J (0.590 lb/h/shp)
TPE331-5	101.7 µg/J (0.602 lb/h/shp)
TPE331-8	96.7 µg/J (0.572 lb/h/shp)
TPE331-10	94.6 µg/J (0.560 lb/h/shp)
TPE331-11	94.3 µg/J (0.558 lb/h/shp)
TPE331-12	92.5 µg/J (0.547 lb/h/shp)
TPE331-14/-15	84.8 µg/J (0.502 lb/h/shp)
T76-G-410/411	101.4 µg/J (0.600 lb/h/shp)

Oil Consumption

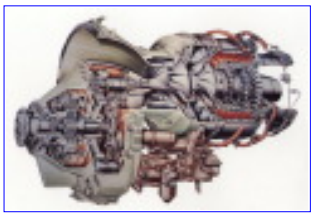
Max	0.009 kg (0.02 lb)/h

Contract Price

Typical catalogue figures, TPE331-10, US$300,000; TPE331-11/-12, US$350,000; TPE331-14/-14GR, US$550,000.

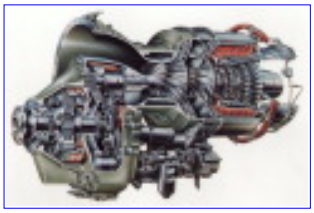
UPDATED

Cutaway drawing of TPE331-3



Cutaway drawing of TPE331-10U/-11U

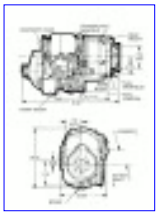
TPE331-10



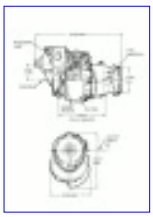
Cutaway drawing of TPE331-12U

TPE331-14

Cutaway drawing of TPE331-14



TPE331-IOU
(2000)



TPE331-14
(2000)

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AERO-ENGINES - TURBOSHAFT, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL

HONEYWELL LTC4, T55

US military designation: T55

This engine is based on the T53, but with considerably higher mass flow. Design began in 1954, and testing of complete engines started in 1957. Total operating time by late 1997 on 4,416 engines was over 7.5 million hours. By 2002 the total exceeded 9 million.

The following versions have been produced:

T55-L-7 (LTC4B-8)

Turboshaft engine, with output shaft speed equal to power turbine speed. Integral oil tank and cooling system. For Boeing CH-47A Chinook.

T55-L-7C

Uprated version of L-7, for CH-47B.

LTC4B-8D

Modified T55-L-7C.

T5508D

Commercial version of LTC4B-8D.

YT55-L-9 (LTC4G-3)

Turboprop using basic power producing section of T55-L-7. Powered North American YAT-28E and Piper Enforcer.

T55-L-11 (LTC4B-11)

Uprated and redesigned version of L-7, with a second stage added to the compressor turbine and variable-incidence inlet guide vanes ahead of the compressor. First two compressor stages transonic. New atomising fuel nozzles. Powers CH-47C Chinook, first deliveries in August 1968. Licensed to Piaggio (Italy). Current versions are L-11A/D/E.

LTC4B-12

Proposed growth version with 4,340 shp maximum power rating. Higher turbine entry temperature and increased turbine cooling. Replaced by L-712.

LTC4R-1

Turboprop version of L-11 with Lycoming `split-power' reduction gear.

T55-L-712

Improved L-11D. Wide-chord compressor blades without inlet guide vanes, and one-piece rotor. Licensed to Piaggio (Italy) and KHI (Japan), for CH-47D, Chinook HC.2/3. In 1998, the US Army was upgrading 300 CH-47D helicopters, a major change being to modify the L-712 engines to GA-714A standard.

AL5512

Commercial L-712, with engine-out contingency rating of 3,250 kW (4,355 shp). Boeing 234, 360.

T55-L-714

Growth version with cooled gas-generator turbine blades, FADEC and improved torquemeter. By May 1999 kits to upgrade engines to this standard were still in Low-Rate Initial Production (LRIP), with 115 ordered since 1997. OEI rating (contingency) 3,780 kW (5,069 shp); maximum continuous 3,108 kW (4,168 shp). For MH-47E Chinook.

T55-GA-714A

GA-714 plus latest RAM-D improvements, such as marinisation and longer-life rotating parts including cooled and pin-retained HP turbine blades, a segmented HP turbine nozzle, wide-bore longer-life

compressor disks and better compressor sealing. Other changes include spraying a thermal barrier coat on the combustor liner, eliminating the turbine exit guide vanes and fitting an increased-efficiency jetpipe. Some of these modifications stemmed from Joint Turbine Aviation Gas Generator (JTAGG) research.

In May 1999 the US Army awarded a US$15,119,251 modification to a letter contract for the conversion of 70 T55-L-712 engines to GA-714A standard, using retrofit kits. Most of this work is being performed by Honeywell at Greer, South Carolina. In April 2001 the Army awarded a further contract to build up to 259 new GA-714A engines. Future contracts are expected for 700 to 800 new engines. They will be produced at Honeywell's Phoenix site.

Type

Free-turbine turboshaft or turboprop.

Intake

Annular-type casing of aluminium alloy with four struts supporting integral oil tank and front main bearings. Provision for intake screens.

Compressor

Seven axial stages followed by a single centrifugal stage. Two-piece steel stator casing with one row of inlet guide vanes (fixed on L-7, L-7C and L-9, variable on later versions) and seven rows of steel stator vanes, bolted to steel alloy diffuser casing to which combustion chamber casing is attached. Rotor comprises seven stainless steel discs and one titanium impeller mounted on shaft supported in forward ball thrust bearing and rear roller bearing. Pressure ratio (L-7) 7, (L-7C) 9, (L-11, R-1) 8.2, (GA-714) 9.3. Mass flow (GA-714) 13.19 kg (29.08 lb)/s.

Combustion Chamber

Annular reverse-flow type. Steel outer shell and inner liner with 28 fuel burners with downstream injection.

Compressor Turbine

Single stage on L-7, L-7C and L-9, two-stage on all later versions. In early versions (L-11, R-1), first- and second-stage rotor blades lost-wax cast; in GA-714 both stages air-cooled. Discs flange-bolted to HP shaft.

Power Turbine

Two stages, widely separated by rear-bearing struts. Solid steel rotor blades held in discs splined to LP shaft.

Jetpipe

Fixed-area nozzle, with inner cone, supported by four radial rods.

Output

LP shaft passes down the centre of the engine to a front coupling shaft with internal splines. Turboshaft engines have no integral gearbox. Turboprops have a two-stage helical reduction gear centred in the annular inlet.

Accessories

Bevel gear at the front of the compressor drives two radial (tower) shafts, one above and one below. Fuel pump and airframe accessories driven by lower shaft.

Starting

Electric starter or starter-generator or air or hydraulic starter. Bendix-Scintilla TGLN high-energy ignition unit. Four igniter plugs.

Control System

(Early versions) Hamilton Standard JFC 31 fuel control system. Gear-type fuel pump, with gas producer and power shaft governors, flow control with altitude compensation and shut-off valve. (GA-714) Chandler Evans (Coltec Industries) FADEC-type EMC-32T-2, consisting of hydromechanical unit with gear-type pump and DECU.

Fuel Specification

MIL-J-5624L grade JP-4, JP-5, MIL-T-83133 grade JP-8 or CITE.

Oil System

Recirculating. Integral tank and cooler.

Oil Specification

MIL-L-7808, MIL-L-23699.

Dimensions

Length:	
T55-L-7B, -7C, LTC4B-8D (T5508D)	1,118 m (44.0 in)
T-55-L-712, GA-714	1,181 mm (46.5 in)
Diameter:	
All versions	617 mm (24.3 in)

Weight, Dry

T55-L-7C	267 kg (590 lb)
YT55-L-9	363 kg (799 lb)
LTC4B-8D	274 kg (605 lb)
T55-L-712	340 kg (750 lb)
T55-L-11	322kg (710 lb)
T55-L-714	374 kg (824 lb)
T55-GA-714A	396.4 kg (874 lb)

Performance Ratings

T-O, S/L:	
T55-L-7C	2,125 kW (2,850 shp)
YT55-L-9	1,887 ekW (2,529 ehp)

T55-L-11	2,796 kW (3,750 shp)
LTC4B-8D	1,678 kW (2,250 shp) flat rated
T55-L-712	3,267 kW (4,378 shp)
AL5512	3,039 kW (4,075 shp)
T55-GA-714	3,629 kW (4,868 shp)

Specific Fuel Consumption

T-O, as above:	
T55-L-7C	101.4 Ág/J (0.60 lb/h/shp)
YT55-L-9	102.7 Ág/J (0.680 lb/h/ehp)
T55-L-11, L-712, AL5512	896 Ág/J (0.53 lb/h/shp)
T55-GA-714	85.0 µg/J (0.503 lb/h/shp)
T55-GA-714A (spec. maximum)	86.7 Ág/J (0.513 lb/h/shp)

Contract Price

February 2000, 18 T55-GA-714A engines for Taiwan US$23,346,000

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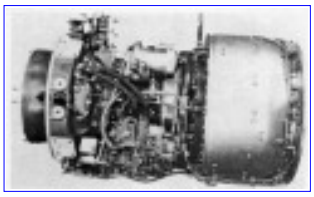
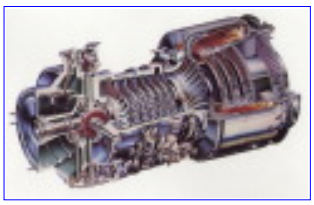


T55-L-11



Cutaway drawing of T55-GA-714 or AL5512

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL

HONEYWELL AS900

Recognising the age of the core of the ALF family of engines, the design team at Phoenix began studies for an upgraded engine in the 1980s. It was decided to canvas opinion among operators and service centres, and the result was astonishing. Almost without exception, the answer that came back was ``Forget fancy new technology, we want an engine that is bulletproof!" The call was for total reliability, easy maintainability and the lowest cost per unit thrust. The resulting AS900 family is all new, and is today regarded more as an upwards extension of the thrust range covered by the TFE731. Among the major design changes are elimination of the fan-drive gearbox and replacement of the folded reversed-flow combustor by a straight-through pattern. Compared with earlier engines the parts count has been dramatically reduced.

In mid-1997 the AS900 design team began a comprehensive 15-month technology validation programme. The compressor, combustor and HP turbine were put through a test programme that culminated with their integration into an engine core which ran in December 1997 and completed more than 100 hours of testing. These tests met or exceeded expectations, and paved the way for full-scale development. Bombardier selected the AS907 for its Continental business jet in mid-1998, and BAE Systems chose the AS 977 for the RJX regional airliner in February 1999. Launch of the AS900 family was announced in September 1998.

The first run of a complete engine took place at Phoenix on 30 July 1999. The fan-blade-off test was carried out ahead of schedule in December 1999. It was successful, and accordingly the AS900 made its first flight, in a nacelle on the right side of the forward fuselage of the Honeywell Boeing 720 testbed,

from Phoenix Sky Harbor airport on 29 January 2000. By July 2000, seven engines at Honeywell, TA and AIDC had run 2,000 hours, demonstrating thrust up to 35.69 kN (8,025 lb st), ahead of predicted temperature margins and with sfc `spot on'. Altogether the certification programme includes ten engines, in addition to others supplied to customers for flight test. In early 2002 certification was due in the first half of that year. At EIS total running time will have exceeded 10,000 hours.

All engines in the family are expected to be supplied as an integrated power plant system (IPPS), with all engine-mounted accessories and reverser. Customers will have the option of OCM (on-condition maintenance), with an on-wing life of 25,000 (possibly 30,000) cycles, or fixed inspection intervals of 3,500 hours for the hot section and 7,000 hours for the compressor zone. Hot-section inspection will be possible with the engine on-wing, as will replacement of the fan, LP turbine, main shaft bearings and seals and the accessory gearbox. LRU replacement will be possible in 15 to 20 minutes using a maximum of 13 standard tools.

Honeywell has assembled a team of partners, some of which are named in the description which follows. All commercial engines in this family will be designed to achieve noise and emissions levels well below Stage 3 requirements. Honeywell Engines has defined the growth steps to achieve a 25 per cent increase in thrust to meet anticipated future requirements. New hot-section alloys would enable ratings to reach at least 39.14 kN (8,800 lb st).

AS907

Selected on 19 October 1998 to power Bombardier Continental. Thermodynamic T-O rating 33.78 kN (7,595 lb st), but flat-rated at 28.91 kN (6,500 lb st) to ISA+15ºC. Smaller propulsive nozzle for increased thrust at high altitude. Configured for lateral pylon. Design on-wing life 15,000 cycles.

AS977-1A

Selected on 16 February 1999 to power Avro RJX. Flat-rated at 31.58 kN (7,100 lb st) to ISA+15ºC. Slightly larger flowpath through turbine stators and final nozzle for increased T-O thrust and longer design life. Will retrofit on existing BAE 146 and RJ overhead pylon.

Type

Two-shaft turbofan.

Fan

Single stage, with 22 inserted wide-chord blades of forged titanium with optimum aerodynamic form (no dampers) provided by AIDC. Pointed spinner not requiring anti-icing which centrifuges foreign objects out through bypass duct. Advanced external containment. Bypass ratio 4.2.

Compressor

Four axial stages with first two stators variable, constructed as integrally bladed blisks. Co-rotating single centrifugal stage. Engine has only two lubrication sumps, in cooler zones. Shaft bearings and rotating seals (supplied by Orenda Engines) can be changed on installed engine. Overall pressure ratio approximately 29.

Combustion Chamber

Compact annular with air entry at outer diameter. Simple manufacture from outer and inner flat panels with `a myriad of laser-drilled holes' to give precise temperature distribution. Effusion cooling. The

cross-section drawing shows one fuel burner (of 16) and one igniter (of two). Designed for uniform temperature profile and minimum emissions.

HP Turbine

Two stages, with total of only 72 blades and 50 vanes. Advanced cooling technology enables less-expensive materials to be used while achieving extended service life.

LP Turbine

Three stages with direct drive to fan. Rotating parts by TechSpace Aero. Static parts provided by ITP.

Jetpipe

Structural bypass duct. Bypass/core mixer upstream of single propulsion nozzle.

Reverser

Hurel-Dubois Papillon target type (see feature `Civil engines').

Accessories

Integrated package on underside of fan case. For RJ, supplied by Shimadzu. Air starter, eliminating need for 28 V supply.

Control System

Dual-channel FADEC with extensive diagnostic and troubleshooting capabilities.

Nacelle

Supplied by GKN Westland.

Dimensions

Length	Approximately 2,290 mm (90 in)
Fan diameter	868.7 mm (34.2 in)

Weight, Dry

AL907	641.8 kg (1,415 lb)
AL977	620.5 kg (1,368 lb)

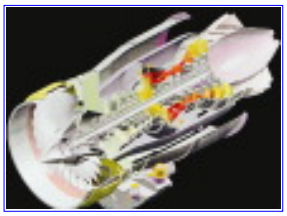
Performance Ratings

T-O, S/L:	
AS907	28.91 kN (6,500 lb st) to ISA+15 ºC
AS977	31.14 kN (7,000 lb st) to ISA+15 ºC

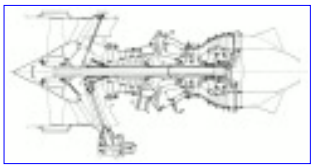
Specific Fuel Consumption

AS907 cruise, 13,716 m (45,000 ft), M0.8	19.34 mg/Ns (0.684 lb/h/lb)
AS977 cruise, 10,668 m (35,000 ft), M0.7	17.98 mg/Ns (0.636 lb/h/lb)

UPDATED

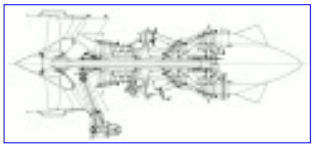


Computer-generated image of cutaway AS900 (1999)



Longitudinal section through AS977 (1999)

The first AS900, instrumented for test (2000)



AS900 (2001)

AS900 (2001)

AS900 (2002)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL

HONEYWELL TFE731

Announced in April 1969, the TFE731 is a two-spool geared turbofan designed for business jet aircraft. Use of a geared fan confers flexibility in operation and yields optimum performance at up to 15,545 m (51,000 ft). As noted below, Honeywell offers an ongoing programme of upgrades to reduce maintenance and improve engine value.

TFE731-2

First production model. Deliveries for the Falcon 10 took place in August 1972. Also powers CASA C-101EB, IA 63 and AT-3. Dash-2A powers Chinese L8/K8, and -2B powers Pampa 2000 and Learjet C-21A, 31, 35 and 36; in each case with digital control. An upgrade to long-life Dash-2C standard was introduced in late 1998, 131 engines being thus modified by October 2001.

TFE731-3

Increased turbine temperature. Certificated September 1974 and first delivered in late 1974 for re-engined JetStar. Also selected for the production JetStar II, Dassault Falcon 50, Hawker 125-700, 125-400 and -600 retrofit, IAI Westwind 1124, Sabreliner 65 and 65A, CASA C-101BB and IA 63. In June 1997 TBO was extended from 1,400 to 2,100 hours. By October 2001 a total of 862 TFE731-3

engines had been upgraded to Dash-3C or -3D configuration.

TFE731-3A

Improved fan stage. Certificated January 1981. Powers the Learjet 55 and IAI Astra 1125.

TFE731-3B

HP turbine fitted with DS blades, improved third-stage of LP turbine. Certificated March 1981. Powers the Cessna Citation III and VI.

TFE731-3C

Improved HP and LP turbine cooling, to permit increased climb and cruise temperatures, increased cycle life and maintenance intervals. Certificated January 1993. New production and field conversions power the Citation III and VI and Astra 1125.

TFE731-3D

Improved HP and LP turbine cooling as -3C, offered as a field conversion for operators of the TFE731-3.

TFE731-4

Powers Citation VII. Certificated late 1991. Powers Czech L-139 with digital control.

TFE731-5

Higher bypass ratio fan, driven by new LP turbine. Digital control. Certificated in November 1983 for Hawker 125-800 and CASA C-101.

TFE731-5A

Mixer nozzle reducing sfc and raising standard thrust to that of the APR rating. Certificated in December 1984 for Dassault Falcon 900, and offered as retrofit for Falcon 20. Volvo Aero has a 5.6 per cent share of the 731-5 production programme.

TFE731-5B

Uprated version, certificated in 1990 for Falcon 900B and Falcon 20 retrofit. **5BR** for Hawker 800XP. In June 2001 Honeywell received approval for a replacement N1 digital control system. By October 2001 a total of 273 TFE731-5A engines had been upgraded to -5B standard.

TFE731-20

First of new series certificated 1996. Changes include smaller -5 fan, new HP compressor, HP turbine and gearbox; FADEC. Selected for Learjet 45.

TFE731-40

Uprated version of TFE731-20. Certificated in July 1995 for IAI Astra SPX and Falcon 50EX. Available as retrofit for Falcon 50B.

TFE731-60

Matches power section of TFE731-40 with a larger fan giving higher BPR, driven via a new gearbox. Certificated May 1995. Powers Dassault Falcon 900EX.

Over 8,200 engines in service by 2001 had flown a total of 49 million hours.

Type

Turbofan with two shafts and geared front fan.

Fan

Single-stage axial titanium fan, with inserted blades. The fan shaft is connected directly to the planetary gearbox ring gear. Mass flow, sea level static, TFE731-2, 51.25 kg (113 lb)/s; -3, 53.7 kg (118.3 lb)/s; -5, 64.86 kg (143 lb)/s; -20, 66.22 kg (146 lb)/s; -40, 65.77 kg (145 lb)/s; -60, 84.82 kg (187 lb)/s. Bypass ratio, -2, 2.66; -3, 2.80; -5, 3.48; -20, 3.10; -40, 2.90; -60, 3.90.

Compressor

Four-stage LP, followed by centrifugal HP on separate shaft running at higher speeds. Overall pressure ratio (S/L, static): -2, 14.0; -3, 14.6; -20, 14.3; -40, 16.3; -60, 17.8.

Combustion Chamber

Annular reverse flow type, with 12 nozzles injecting tangentially. Meets EPA/FAA emission requirements.

HP Turbine

Single-stage HP. Average TGT, S/L, maximum T-O, -2, 860ºC; -3, 907ºC; -5, 952ºC; -20, 963ºC; -40/-60, 1,022ºC.

LP Turbine

Three stages.

Accessories

Pads provided for hydraulic pump, starter/generator or starter motor and alternators. Pads on rear side of gearbox drive fuel control and oil pump.

Control System

Single-channel digital electronic with hydromechanical back-up.

Dimensions

Length overall: -2, -3, -3D	1,522 mm (59.92 in)
-3A, -3B, -3C	1,431 mm (56.34 in)
-4	1,464 mm (57.63 in)

-5	1,652 mm (65.03 in)
-5A, -5B	2,314 mm (91.10 in)
-20, -40	1,547 mm (60.91 in)
-60	2,083 mm (81.99 in)
Intake diameter:	
-2, -3, -3A, -3B, -4, -20, -40	716 mm (28.2 in)
-5, -5A, -5B	754 mm (29.7 in)
-60	781 mm (30.73 in)
Width:	
-2, -3, -3A, -3B, -4	869 mm (34.20 in)
-20, -40	847 mm (33.35 in)
-5, -5A, -5B	858 mm (33.79 in)
-60	932 mm (36.68 in)
Height overall:	
-2, -3, -3A, -3B, -4, -20, -40	1,000 mm (39.36 in)
-5, -5A, -5B	1,029 mm (40.52 in)
-60	1,076 mm (42.37 in)

Weight, Dry

-2	337 kg (743 lb)
-2A	350 kg (772 lb)
-2B	351 kg (773 lb)
-3	342 kg (754 lb)
-3A	352 kg (775 lb)
-3B	349 kg (769 lb)
-3C	352 kg (777 lb)
-3D	348 kg (767 lb)
-4	373 kg (822 lb)
-5	387 kg (852 lb)
-5A	402 kg (884 lb)
-5B	408 kg (899 lb)
-20, -40	406 kg (895 lb)
-60	448 kg (988 lb)

Performance Ratings

Max T-O (S/L):	
-2	15.57 kN (3,500 lb st) to 22.2ºC
-2A	16.01 kN (3,600 lb st) to 23ºC

-2B	15.57 kN (3,500 lb st) to 25.6ºC
-3, -3A, -3D	16.46 kN (3,700 lb st) to 24.4ºC
-3B, -3C	16.24 kN (3,650 lb st) to 21.1ºC
-4	18.15 kN (4,080 lb st) to 24.4ºC
-5	19.15 kN (4,304 lb st) to 23ºC
-5A	20.02 kN (4,500 lb st) to 23ºC
-5B	21.13 kN (4,750 lb st) to 25ºC
-20	15.57 kN (3,500 lb st) to 33.9ºC
-40	18.9 kN (4,250 lb st) to 25ºC
-60	22.24 kN (5,000 lb st) to 32ºC
Max T-O (APR, auto performance reserve):	
-2B	16.24 kN (3,650 lb st) to 25.6 ºC
-3, -3D	17.13 kN (3,850 lb st) to 24.4 ºC
-3A	17.26 kN (3,880 lb st) to 24.4 ºC
-3B, -3C	17.13 kN (3,850 lb st) to 21.1 ºC
-4	18.15 kN (4,080 lb st) to 29.4 ºC
-5	20.02 kN (4,500 lb st) to 23 ºC
-5A	20.02 kN (4,500 lb st) to 27.9 ºC
-5B	21.13 kN (4,750 lb st) to 29.4 ºC
-20	16.24 kN (3,650 lb st) to 33.9 ºC
-40	18.90 kN (4,250 lb st) to 30 ºC
Cruise (12,200 m; 40,000 ft at M0.8):	
-2	3.36 kN (755 lb st)
-2A, -2B	3.54 kN (795 lb st)
-3, -3D	3.63 kN (817 lb st)
-3A	3.73 kN (838 lb st)
-3B, -3C	3.75 kN (844 lb st)
-4	4.13 kN (929 lb st)
-5	4.25 kN (955 lb st)
-5A	4.39 kN (986 lb st)
-5B	4.67 kN (1,050 lb st)
-20	3.87 kN (870 lb st)
-40	4.49 kN (1,010 lb st)
-60	4.98 kN (1,120 lb st)

Specific Fuel Consumption

Cruise (as above):	

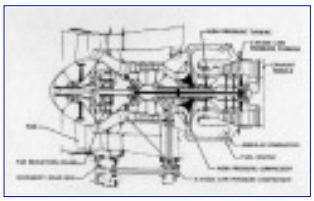
-2	22.38 mg/Ns (0.790 lb/h/lb)
-2B	22.45 mg/Ns (0.792 lb/h/lb)
-3, -3D	22.49 mg/Ns (0.794 lb/h/lb)
-3A	22.21 mg/Ns (0.784 lb/h/lb)
-3B, -3C	22.09 mg/Ns (0.780 lb/h/lb)
-4	21.84 mg/Ns (0.771 lb/h/lb)
-5	22.01 mg/Ns (0.777 lb/h/lb)
-5A	21.16 mg/Ns (0.747 lb/h/lb)
-5B	20.82 mg/Ns (0.735 lb/h/lb)
-20	20.54 mg/Ns (0.725 lb/h/lb)
-40	20.79 mg/Ns (0.734 lb/h/lb)
-60	18.78 mg/Ns (0.663 lb/h/lb)

Contract Price

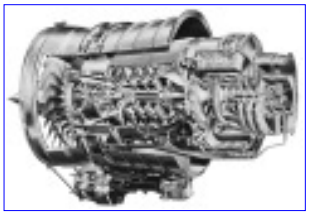
Honeywell quote about US$5.4 million to upgrade Falcon 50B to Dash-40 engines, inclusive of new air-data computer and yaw damper; owner could expect trade-in credit of about US$1 million for original TFE731-3 engines.

UPDATED

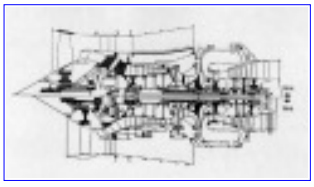
TFE731-2



Longitudinal section through TFE731-2



Cutaway drawing of TFE731-2



Longitudinal section through TFE731-3 (bottom half) and TFE731-5 (top half)

Cutaway drawing of TFE731-5

TFE731-20

TFE731-60

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AERO-ENGINES - TURBOPROP/ - TURBOSHAFT, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL

HONEYWELL LTS101 AND LTP101

US military designation: YT702-LD-700
This family of turboshaft and turboprop engines share a common core, with minor variations, designed for the lowest possible life-cycle costs. Each engine comprises a front gearbox module incorporating accessory drives (turboprop versions have an extra stage of gearing), a compressor module and a combustor/turbine module.

Turboprop versions:

LTP101-600A-1A

T-O rating 462 ekW (620 ehp). Powers Riley Cessna 421, Piaggio P.166-DL3 (pusher installation), Air Tractor conversion, Cresco 08-600 and Fletcher FU-24 conversion. Has flown in Turbine Islander, OMAC, Piper Brave and Do 128-6.

LTP101-700A-1A

T-O rating 522 ekW (700 ehp). Powers P.166-DL3SEM (pusher installation), Ag-Cat conversion and Turbo-Thrush conversion.

Turboshaft versions:

LTS101-600A-2

T-O rating 459 kW (615 shp). Provides 6,000 rpm front output for AS350D AStar.

LTS101-600A-3

T-O rating 459 kW (615 shp). Powers AS350D Mk 3.

LTS101-600A-3A

T-O rating 485 kW (650 shp).

LTS101-650B-1

T-O rating 410 kW (550 shp) to 41ºC (106ºF). Radial inlet and 6,000 rpm output for BK117A and B.

LTS101-650C-3, C-3A

T-O rating 470 kW (630 shp). Provides 9,545 rpm output for Bell 222.

LTS101-750B-1

T-O rating 410 kW (550 shp) to 41~C (106~F). Provides 6,000 rpm output for BK117 B-1 and B-2 and BK117-1.

LTS101-750B-2

T-O rating 510 kW (684 shp). Radial inlet. Provides 6,000 rpm output for HH-65A Dolphin.

LTS101-750C-1

T-O rating 510 kW (684 shp). Provides 9,545 rpm output for Bell 222B and 222UT.

Type

Free-turbine turboshaft or turboprop.

Intake

Most versions have an all-round circumferential inwards-radial inlet protected by a screen. An alternative is an overhead scoop or scroll inlet.

Compressor

Single axial stage followed by a single centrifugal stage. From 1994 axial rotor is single forging. Typical mass flow 2.31 kg (5.1 lb)/s. Pressure ratio (all variants) 8.5.

Combustion Chamber

Annular, folded, reverse-flow, effusion-cooled, with ring of fuel nozzles around the rear face (a modified fuel manifold was introduced in 1994).

Compressor Turbine

Single stage splined to shaft adjacent to centrifugal compressor. Blades held in fir-tree roots. From 1994 single-crystal blades available.

Power Turbine

Single stage, overhung behind rear thrust bearing. A 1994 upgrade is insertable power-turbine rotor blades.

Jetpipe

Direct connection to rear of engine to suit installation.

Output

All versions have an initial primary spur gearbox. On most turboshaft versions the gearbox can provide an output from the bottom of the gearcase to front or rear. Turboprops have a second stage of gears, giving an output at the top of the gearcase at 1,925 rpm.

Dimensions

Length:	
LTP101-600, A-1A	914 mm (36.0 in)
LTP101-700	949 mm (37.37 in)
LTS101-600A-2	785 mm (30.9 in)
LTS101-600A-3, A-3A	800 mm (31.5 in)
LTS101-650B-1	790 mm (31.1 in)
LTS101-650C, 750B-1, 750C-1	795 mm (31.3 in)
LTS101-750B-2	823 mm (32.4 in)
Diameter:	
LTP101-600A-1A, -700	592 mm (23.3 in)
LTS101-600A-2, A-3, A-3A	599 mm (23.6 in)
LTS101-650C, 750C	574 mm (22.6 in)
LTS101-650B, 750B-1	645 mm (25.4 in)
LTS101-750B-2	627 mm (24.7 in)

Weight, Dry

LTP101-600A-1A	147 kg (325 lb)
LTP101-700	152 kg (335 lb)
LTS101-600A-2, A-3, A-3A	115 kg (253 lb)
LTS101-650B-1	121 kg (266 lb)
LTS101-650C	109.5 kg (241 lb)

LTS101-750B-2	121.5 kg (268 lb)
LTS101-750B-1	123 kg (271 lb)
LTS101-750C	111 kg (244 lb)

Performance Ratings

See model listing

Specific Fuel Consumption

T-O, S/L:	
LTP101-600, A-1A	92 Ág/J (0.544 lb/h/ehp)
LTP101-700	93 Ág/J (0.550 lb/h/ehp)
LTS10S-600A-2, 750B-2	96.6 Ág/J (0.571 lb/h/shp)
LTS101-600A-3, A-3A	98.0 Ág/J (0.580 lb/h/shp)
LTS101-650B-1, 750B-1, C-1	97.5 Ág/J (0.577 lb/h/shp)
LTS101-650-C	96.7 Ág/J (0.572 lb/h/shp)

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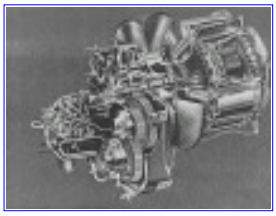
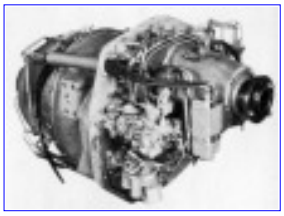
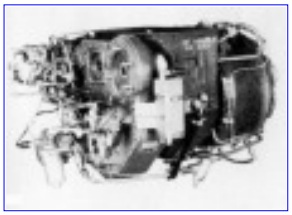


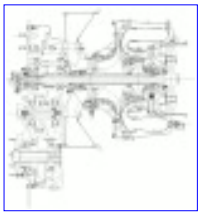
Cutaway LTS101-750B-1 (2000)



LTS101-750B-1 turboshaft



Left to right: LTP101-700A-1A turboprop and LTS101-750C-1 turboshaft



Longitudinal section through LTS101-750B-1 (2000)

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AERO-ENGINES - TURBOSHAFT, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL

HONEYWELL LTC1, T53

US military designation: T53
The T53 was developed under a joint US Air Force/Army contract. More than 19,000 of these engines have logged over 47 million hours since 1956. Licences for manufacture of the T53 are held by Rolls-Royce Deutschland in Germany, Piaggio in Italy, Kawasaki in Japan, and AIDC in Taiwan.

T53-L-2

Navy version of L-11.

T53-L-7 (LTC1F-2)

Turboprop. Produced for Grumman OV-1A and B Mohawk. The corresponding commercial version was the **T5307A**.

T53-L-11 (LTC1K-5)

Turboshaft. Produced for Bell UH-1B, D and E, and Kaman HH-43F. The commercial version, powering Bell 204B helicopters, is the **T5311A**.

T53-L-13

Uprated version of L-11, which it superseded in production in August 1966. Redesigned hot end and initial stages of compressor section to provide increased power for hot day and high-altitude performance. Four turbine stages, compared with two in earlier models, and variable-incidence inlet guide vanes combined with redesigned first two compressor stages, permit greater air flow and lower turbine temperatures. This version has an atomising combustor to facilitate operation on a wider range of fuels. Powers Bell UH-1C, UH-1H, UH-1M and CUH-1H Iroquois and AH-1F, G and J HueyCobra. The commercial version of the T53-L-13 was the T5313A, later superseded by the T5313B, for Bell 205A and A-1.

T53-L-15

Turboprop version of L-13, powering the Grumman OV-1C and D Mohawk.

LTC1K-4

Generally similar to T53-1-13. Powered prototype Canadair CL-84 tilt-wing aircraft and Bell XV-15.

T53-L-701

Turboprop incorporating `split-power' reduction gear.

T53-L-701A

Improved turboprop for OV-1 and T-CH-1A.

T53-L-703

Improved-durability L-13, with PTIT (power-turbine inlet temperature) instead of EGT (exhaust-gas temperature) limiter system. Thermodynamic rating 1,343 kW (1,800 shp). Powers Bell UH-1H and AH-1E/F/P/S. Selected for Airwork Huey 703 (UH-1H enhancement package).

LTC1K-4K

Direct drive L-13 suitable for operation from 105~ nose up to 90~ nose down.

T5317A-1

Similar to -703 except has EGT instead of PTIT. Improvements over L-13 include improved cooling of first gas producer turbine nozzle plus air-cooled blades in first turbine rotor. In production for Kaman K-Max.

T5317B

Similar to 5317A-1 except has PTIT instead of EGT. Powers Fuji-Bell 205 and Bell 205A-1.

The following details apply to the T53-L-13, T53-L-703 and T5317.

Type

Free turbine turboshaft.

Intake

Annular casing of magnesium alloy, with six struts supporting reduction gearbox and front main bearings. Anti-icing by hot air tapped from engine.

Compressor

Five axial stages followed by a single centrifugal stage. Two-piece magnesium-alloy casing with one row of variable-incidence inlet guide vanes and five rows of steel stator blades, bolted to one-piece steel diffuser casing with tangential outlet to combustion chamber. Rotor comprises one stainless steel and one titanium-alloy drum, with stainless-steel blades and one titanium centrifugal impeller, mounted on shaft supported in forward ball thrust and rear roller bearings. Compression ratio 7.4. Mass flow 4.85 kg (10.5 lb)/s at 25,240 rpm.

Combustion Chamber

Annular reverse-flow type, with one-piece sheet steel outer shell and annular liner. Twenty-two atomising fuel injectors.

Compressor Turbine

Two stages. Air-cooled nozzle vanes, cored-out cast (lost-wax) rotor blades in first stage, solid in second, held in high-nickel discs flange-bolted to HP shaft.

Power Turbine

Two stages. Solid blades with tip shrouds mounted in discs splined to output shaft.

Jetpipe

Fixed-area nozzle. Steel outer casing and inner cone, supported by four radial struts.

Output

LP shaft passes through centre of engine to two-stage helical spur gear driving front output shaft with internal splines.

Accessories

Bevel gear at the front of the HP shaft drives a `banana' N_1 gearbox under the compressor case. Spur gear at the front of the shaft drives an N_2 gearbox.

Starting

Electric starter or starter-generator (not furnished). Bendix-Scintilla TGLN high-energy ignition unit. Four igniter plugs.

Control System

Chandler Evans TA-7 fuel control with one dual fuel pump. 41.1 bars (600 lb/sq in). Interstage air bleed control.

Fuel Specification

ASTM-A-1, MIL-J-5624, MIL-F-26005A, JP-1, JP-4, JP-5, CITE.

Oil System

Recirculating system, with gear pump, 4.83 bars (70 lb/sq in).

Oil Specification

MIL-L-7808, MIL-L-23699.

Dimensions

Length:	
All turboshaft versions	1,209 mm (47.6 in)
L-15, L-701	1,483 mm (58.4 in)
L-701A	1,509 mm (59.4 in)
Diameter (all versions):	
Inlet	584 mm (23.0 in)
Overall	622 mm (24.49 in)

Weight, Dry

L-13	249 kg (549 lb)
T5311A	225 kg (496 lb)
L-13B, T5313B	245 kg (540 lb)
T5317A, B	248 kg (547 lb)
L-703	247 kg (545 lb)
L-15	274 kg (605 lb)
L-701	312 kg (688 lb)

Performance Ratings

T-O, S/L:	
T5311	820 kW (1,100 shp)
T5313B, L-13, L-13B	1,044 kW (1,400 shp)
T5317A, B	1,119 kW (1,500 shp)
LTC1K-4K	1,156 kW (1,550 shp)
L-703	1,108 kW (1,485 shp)
L-15	865 kW (1,160 shp) + 0.48 kN (108 lb st)
L-701	1,082 ekW (1,451 ehp)

Specific Fuel Consumption

T-O, as above	

T5311A	115 Ág/J (0.68 lb/h/shp)
T5313B, L-13B	98.0 Ág/J (0.58 lb/h/shp)
T5317A, B	99.7 Ág/J (0.59 lb/h/shp)
T53-L-703, L-701	101.4 Ág/J (0.60 lb/h/shp)
LTC1K-4K	98.7 Ág/J (0.584 lb/h/shp)

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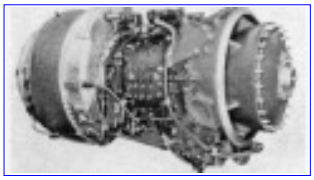
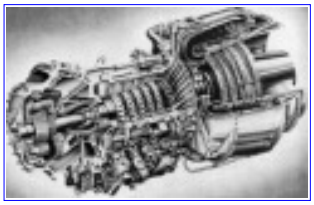
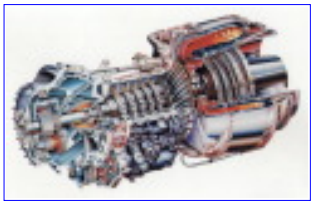


T53-L-11



Cutaway drawing of T5317



Cutaway drawing of T53

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL

HONEYWELL LF 507

This family of commercial turbofans is based on the ALF 502R, using the same core. The **LF 507-1H**, the first engine in the family, was certificated in October 1991. The **LF 507-1F** was certificated in March 1992 and entered airline service with the Avro RJ85 in April 1993. By October 2001 over 824 engines had been produced, with 6.8 million hours of service experience.

The following are among the differences, compared with the ALF 502:

Fan

Minor improvements, rotating with a core booster (supercharger) with two stages. Drive gearbox lubrication improved.

Compressor

Casing fabricated in steel.

Combustion Chamber

Machined liner (flame tube).

HP Turbine

Reduced TGT. Since 1985 the HP turbine bearing has been redesigned.

LP Turbine

Final (fourth) nozzle made by casting.

Control System

LF 507-1F has Chandler-Evans EMC-32R FADEC. LF 507-1H has Hamilton Standard JFC-31 hydromechanical fuel control.

Dimensions

As ALF 502L

Weight, Dry

LF 507-1F	628 kg (1,385 lb)
LF 507-1H	624 kg (1,375 lb)

Performance Ratings

T-O, S/L (both)	31.14 kN (7,000 lb st)

Specific Fuel Consumption

T-O, as above (both)	11.245 mg/Ns (0.397 lb/h/lb st)

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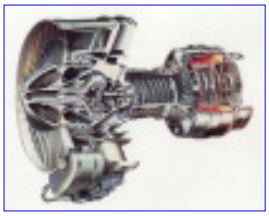

Cutaway drawing of LF 507

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL

ITEC TFE1042-70

US military designations: dry F124-GA-100, afterburning F125-GA-100
This engine was developed by the ITEC (International Turbine Engine Co) as the power plant of the AIDC (Aircraft Industry Development Corporation) Ching-Kuo IDF (Indigenous Defensive Fighter). ITEC is a joint venture between AlliedSignal Engines and Taiwan's AIDC. Fully modular, it has matured as a family of engines marketed by Honeywell.

The following are existing and planned versions:

TFE1042-70

Basic version, in production by ITEC for Ching-Kuo IDF (Indigenous Defensive Fighter). T-O rating, Intermediate (max dry) 26.8 kN (6,025 lb st), max afterburner 41.15 kN (9,250 lb st). By mid-1999 it had completed over 36,000 hours of testing and 130,000 hours in field service. Initial flight release was achieved in February 1989, with full qualification in September 1991; production deliveries began a month later. By mid-1999, a total of 335 engines had been delivered. Inspection intervals, 2,000 hours hot section, 4,000 hours remainder.

F125-GA-100

Similar to TFE1042-70. For US market.

F124-GA-100

A non-afterburning version of the F125, the F124 is being offered for advanced trainer and light combat aircraft. T-O rating 28.02 kN (6,300 lb st). In July 1997, ITEC signed a contract with Aero Vodochody for 79 engines to power L-159 combat trainers of the Czech air force. This aircraft was previously exclusively powered by the DV-2 (see Klimov-PS in International section). The same engine was previously selected for the derived L-159 in both trainer and tactical (single-seat) versions. The L-159 first prototype was rolled out on 12 June 1997, and by January 2000 had completed 607 test flights. The engine received full qualification in October 1998, and by January 2000 ITEC had delivered 24 production engines.

F124-GA-200

A non-afterburning version of the F125, very similar to the GA-100. This engine is being developed for the Aermacchi M-346 twin-engined trainer (derived from the AEM/Yak-130). The principal difference is a new accessory-drive gearbox. Other changes include an engine-mounted dual-channel MIL-1553-capable FADEC, an application-specific exhaust system and different engine mountings. Specification is unchanged from the GA-100. Development was started in late 2000, and the first M-346 is scheduled to fly in spring 2003.

F124-GA-400

This engine is basically the F124-GA-100, with the same ratings. It differs in having external systems and customer interfaces tailored to suit the British Aerospace Hawk and Boeing T-45 Goshawk. One engine was flight tested in a T-45A in September/October 1996. The engine is being marketed as a retrofit for either aircraft.

In June 1999 AlliedSignal (now Honeywell) announced that a different variant of the F124 had been selected by Boeing to power the UCAV-ATD (Unmanned Combat Air Vehicle, Advanced Technology Demonstrator). The Boeing Phantom Works was selected by DARPA (Defense Advanced Research Projects Agency) and the USAF to continue into this programme's second phase, to last 42 months. Honeywell will provide propulsion engines and engineering support. Details and designation of this F124 were not disclosed in the announcement, though it stated that the F124 "can deliver (30.02 kN) 6,750 lb thrust". Growth versions for manned applications will be rated at 30.24 kN (6,800 lb st). It has been suggested that it will have a low-observable vectoring nozzle.

Type

Two-shaft turbofan, with or without afterburner.

Fan

Three stages with rotating spinner. Maximum air flow (1042-70) 43.29 kg (95.4 lb)/s, (F124) 42.7 kg (94.1 lb)/s. Bypass ratio 0.4.

Compressor

Four axial stages followed by one centrifugal. Overall pressure ratio 19.

Combustion Chamber

Annular.

HP Turbine

Single-stage air-cooled.

LP Turbine

Single stage.

Afterburner

When fitted, reheat in bypass and core flows. Three mechanical actuators drive 10-flap variable nozzle.

Control System

Dual redundant FADEC.

Dimensions

Inlet diameter	591 mm (23.25 in)
Width: TFE1042-70	782 mm (30.8 in)
F124	762 mm (30.0 in)
Height: TFE1042-70	848 mm (33.4 in)
F124	914 mm (36.0 in)
Length: TFE1042-70	3,561 mm (140.2 in)
F124	1,697 mm (66.8 in)

Weight, Dry

TFE1042-70	617 kg (1,360 lb)
F124	499 kg (1,100 lb)

Performance ratings

See list of variants.

Specific Fuel Consumption

TFE1042-70 (intermediate)	22.66 mg/Ns (0.80 lb/h/lb st)
F124	22.94 mg/Ns (0.81 lb/h/lb st)

Contract Price

79 F124-GA-100 for Czech Republic, with spares and support, US$175+ million.

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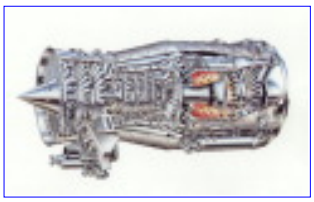

Cutaway drawing of F124-GA-100

Cutaway drawing of TFE1042-70 (F125 similar)

TFE1042-70

TFE1042 display model

F124-GA-100
(2000)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL

HONEYWELL ALF 502

The ALF 502 geared turbofan was launched in 1969, primarily for commercial and executive aircraft. The core is almost identical with that of the T55 turboshaft, and construction is modular.

Current versions are as follows:

ALF 502L

First commercial version. ALF 502L-2 FAA certificated in February 1980 to power Canadair Challenger 600. ALF 502L-2A, L-2C and L-3 certificated 1982-3.

ALF 502R

Reduced rating. ALF 502R-3 FAA certificated January 1981 to power BAE 146. Improved R-3A, R-4 and R-5 certificated 1982-3. R-6 certificated 1984.

By January 2002 ALF 502 engines had flown nearly 21 million hours. The ALF 502 operates on-condition.

Type

High-bypass ratio, two-shaft geared turbofan.

Fan

Cast frame includes four engine mounts 90º apart, and may carry reverser. Fan rotor blades are base and part-span shrouded. Mounted directly behind rotor (6,700-6,970 lb st engines) is a single or (7,500 lb st engines) two stages of compression. Anti-icing of LP compressor inlet by bleed air. Accessory gearbox on fan frame takes HP shaft power. Reduction gear couples LP turbine to fan. Bypass ratio: 502R-3, 5.71; 502R-5, 5.6; 502L, 5.0.

Compressor

Core supercharged by one or two LP stages, as described above. HP compressor has seven axial stages and single radial stage, driven by two-stage HP turbine. Supported by rear roller bearing and forward ball thrust bearing. Acceleration bleed control between stages 6 and 7 operated by main fuel control. Upper and lower casings contain stator half-rings, removable for maintenance, and borescope ports. Overall pressure ratio: R-3, 11.6; R-5, 12.0; L-2, 13.6.

Combustion Chamber

One-piece annular combustor wraps around turbine. Atomising nozzles inserted through outer chamber at rear. Two customer bleed ports around exterior. Combustion liner and housing bolted to compressor diffuser. Disconnecting permits removal for combustor/turbine module, providing access to HP turbine. Borescope ports permit inspection of combustor and turbine without removal.

HP Turbine

Two stages with air-cooled blades.

LP Turbine

Two stages with tip shrouds.

Jetpipe

Fan and core exhaust include flanges for bolted ducts and nozzles.

Accessories

Accessory gearbox carries Hamilton Standard JFC-31 main fuel control, oil pump and filter, tachometer (if required) and provisions for customer accessories.

Dimensions

Length overall:	
ALF 502L	1,487 mm (58.56 in)
ALF 502R	1,443 mm (56.8 in)
Diameter (all)	1,059 mm (41.7 in)

Weight, Dry

ALF 502L	595 kg (1,311 lb)
ALF 502R-3, -3A, -5	606 kg (1,336 lb)
ALF 502R-6	624 kg (1,375 lb)

Performance Ratings

T-O, S/L:	
ALF 502L, L-2, L-2A, L-2C, L-3	33.36 kN (7,500 lb st)
ALF 502L-3 (APR)	34.69 kN (7,800 lb st) APR, or 33.36 kN (7,500 lb st)
ALF 502R-3	29.8 kN (6,700 lb st)
ALF 502R-3A, R-5	31.0 kN (6,970 lb st)
ALF 502R-6	33.36 kN (7,500 lb st)

Specific Fuel Consumption

T-O, as above

ALF 502L, L-2	11.99 mg/Ns (0.424 lb/h/lb st)
ALF 502L-2A, L-2C, L-3, R-6	11.74 mg/Ns (0.414 lb/h/lb st)
ALF 502R-3	11.64 mg/Ns (0.411 lb/h/lb st)
ALF 502R-3A, R-5	11.50 mg/Ns (0.406 lb/h/lb st)

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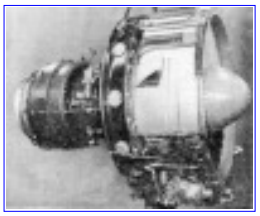
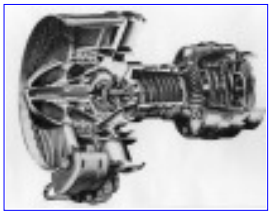
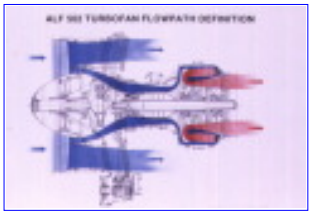


ALF 502R prototype



Cutaway drawing of ALF 502L

ALF 502R-3



ALF 502R longitudinal section

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

GE AIRCRAFT ENGINES

GENERAL ELECTRIC CF6-80E1

This subfamily of engines is based on the CF6-80C2 but with major modifications for higher thrust. It incorporates features and technology from other programmes, including the GE/NASA Energy Efficient Engine (E^3). The entire core structure, frames and rotor have been strengthened, and all versions have been developed as a complete propulsion system, including the nacelle and exhaust system. All versions power the A330, and a second application was intended to be an MD-11 Stretch.

Partners on the CF6-80E1 programme are SNECMA of France (20 per cent), MTU of Germany (9 per cent), FiatAvio and Alfa Romeo Avio of Italy (9 per cent) and Volvo Aero of Sweden, GE's share being 61 per cent.

CF6-80E1A1

T-O rating 281.5 (63,290 lb st). First run in December 1990, achieving a corrected thrust of over 320.3 kN (72,000 lb st). Flight tested on an A300B in early 1992 and certificated on 25 May 1993. Entered service on the A330 on 17 January 1994. JAA-approved for 180-minute ETOPS on 6 February 1995.

CF6-80E1A2

T-O rating 287.0 kN (64,530 lb st) to 30ºC.

CF6-80E1A3

T-O rating 320.27 kN (72,000 lb st). Highest-thrust derivative of the CF6 family. New HP turbine made of R88, with hardware common to C2B8F engine. HP compressor with advanced 3D aerodynamics in Stages 7-14. Launched in October 1999 (see Contract Price, below). Scheduled for certification in 2001, when new customers included EVA Air and Qatar.

CF6-80E1A4

T-O rating 297.5 kN (66,870 lb st) to 30ºC. Certificated by JAA, FAA and Transport Canada on 18 July 1997 at 311.33 kN (70,000 lb st) and delivered as launch engine of A330-200 at 230 tonnes (507,000 lb) gross weight to Canada 3000 in April 1998 with 180-minute ETOPS.

As of 31 July 1997, nine CF6-80E1-powered aircraft were in service, and 14 additional aircraft were on order. Flight time was 116,236 hours, the high-time engine having flown 13,033 hours and the high-cycle leader having accumulated 4,592 cycles without a shop visit. The dispatch reliability was 99.97 per cent.

The CF6-80E1 differs from the CF6-80C2 in the following features:

Fan

Redesigned with diameter increased to 2,438 mm (96.0 in), with number of solid titanium blades with mid-span shrouds reduced from 38 to 34. Bypass ratio typically 5.3. Mass flow typically 873.6 kg (1,926 lb)/s.

LP Compressor

Four-stage core booster redesigned with new aerodynamics to increase core air flow by 9.0 per cent and pressure ratio by 12 per cent.

HP Compressor

Higher temperature alloy in final stage. OPR (E1A2) 32.6.

HP Turbine

New high-temperature alloys and improved cooling. See note on CF6-80C2 HP TURBINE.

LP Turbine

New high-temperature alloys, improved cooling and modified aerodynamics.

Control System

FADEC made on-wing programmable, and system capacity increased.

Dimensions

Length:	
Engine	4,405 mm (173.5 in)
Propulsion system	7,356 mm (289.6 in)
Diameter of fan case	2,794 mm (110.0 in)

Weight, Dry

Engine (all ratings)	5,075 kg (11,189 lb)
Propulsion system	6,733 kg (14,844 lb)

Performance Ratings

See model listing

Specific Fuel Consumption

T-O, as above:	
E1A2	9.26 mg/Ns (0.327 lb/h/lb st)
E1A4	9.40 mg/Ns (0.332 lb/h/lb st)

Contract Price

October 1999, engines for 13 A330, ``more than US$250 million''; March 2001, E1A3 engines for five A330, US$120 million, and E1A3 engines for eight A330, US$180 million.

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CF6-80E1
(1998)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

CFE COMPANY

CFE738

This turbofan is being produced to power large business jet aircraft. It has been designed to the latest airline standard technology, with modular construction for `on wing' maintenance. Engine cores are shipped from CFE-East (GEAE Lynn) complete with the engine control system to CFE-West (Honeywell Engines, Systems & Services) which is responsible for the fan, LP turbine and accessory gearbox, and for engine assembly and test.

CFE738-1-1

This initial version was FAA certificated on 17 December 1993, and JAA validated in August 1994. First deliveries were then made to power the Dassault Falcon 2000. Flight time passed the 100,000-hour mark in April 1998, and in September 1999 time between major inspections was raised to 2,000 hours. President Daub drew attention to ``one of the industry's best standard warranties''. Saudi Arabia's NetJets fractional-ownership confirmed its buy of 12 Falcon 2000s, but conversely the Teal Group's Richard Aboulafia says the engine ``hasn't done the Falcon 2000 any favours''.

CFE738-2

This growth version has been published by Yakovlev as selected for the Yak-77 business jet.

The following data relate to the CFE738-1-1. Growth versions are planned with thrust ratings

exceeding 31.1 kN (7,000 lb).

Type

Two-shaft subsonic turbofan.

Fan

Single stage with 28 inserted titanium blades with part-span dampers and rotating pointed spinner. Front end of LP shaft held in large-capacity ball bearing. Mass flow 95.3 kg (210 lb)/s. Pressure ratio 1.7. Bypass ratio 5.3.

HP Compressor

Five axial stages followed by one centrifugal. First three stator stages variable. Overall pressure ratio, S/L 23, top of climb 35.

Combustion Chamber

Centrifugal diffuser leads into annular chamber with 15 fuel injectors.

HP Turbine

Two stages with cooled blades.

LP Turbine

Three stages. Inter-turbine temperature (cruise) 861ºC.

Jetpipe

Fixed mixer assembly with 20 chutes for combining the hot and cold flows from core and bypass duct. Provision for reverser.

Control System

Dual FADEC.

Dimensions

Length	2,514 mm (99.0 in)
Width	1,092 mm (43.0 in)
Height	1,219 mm (48.0 in)

Weight, Dry

	601 kg (1,325 lb)

Performance Ratings

(uninstalled)

S/L, T-O	26.52 kN (5,918 lb st) to 30ºC (86ºF)
Cruise, 12,200 m (40,000 ft), M0.8	6.51 kN (1,464 lb)

Specific Fuel Consumption

S/L, T-O	10.54 mg/Ns (0.372 lb/h/lb st)
Cruise (as above)	18.13 mg/Ns (0.64 lb/h/lb)

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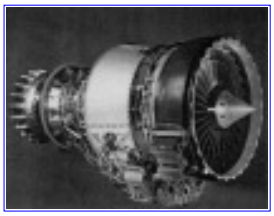
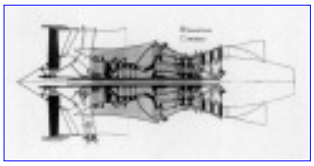


CFE738-1-1



Longitudinal section through CFE738, showing work-split

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AERO-ENGINES - TURBOPROP, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOLOY CORPORATION

SOLOY DUAL PAC

The **Dual Pac 785-1000** power plant was developed jointly by Soloy and Pratt & Whitney Canada (PWC), to provide aircraft with the in-flight dependability of two engines whilst retaining a single centreline propeller. It is a single power plant in which two free-turbine PWC PT6D-114A engines drive a single propeller through a Soloy combining gearbox. The Dual Pac's redundant gearbox provides speed reduction to the propeller shaft, while a multiple clutch arrangement permits independent operation of either engine. Thus, the Dual Pac provides multi-engine reliability in a single package.

The power plant was certified in 1997 as a multi-engine propulsion system. When installed in an aircraft, that aircraft can, given an adequate certification basis, be certified as a twin. The Dual Pac is designed for tractor or pusher installation.

The Soloy/PWC business agreement provides that PWC has exclusive responsibility for providing the Dual Pac power plant to aircraft manufacturers (OEMs). Soloy Corporation supplies the power plant for retrofit of ex-factory aircraft and is developing an installation of the power plant for the Cessna 208B Grand Caravan, the converted aircraft being known as the Soloy Pathfinder 21. US Supplemental Type.

Dimensions

Length overall	1,917.4 mm (75.49 in)
Max width	1,256 mm (49.45 in)
Max height	751 mm (29.57 in)

Weight, Dry

	483.1 kg (1,065 lb)

Performance Ratings

T-O, S/L	991 kW (1,329 shp) to 39.8ºC at 1,700 propeller rpm
Max climb or cruise	991 kW (1,329 shp) to 15.7ºC
Max continuous OEI	493 kW (661 shp) to 40ºC

Specific Fuel Consumption

T-O or max cruise	104.9 µg/J (0.620 lb/h/shp)

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Dual Pac 785-1000 with propeller

Installing the Dual Pac in the prototype Soloy Pathfinder 21

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AERO-ENGINES - TURBOPROP, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

SOLOY CORPORATION

SOLOY TURBINE PAC

The Soloy Turbine Pac is an FAA Supplemental Type Certificate approved turboprop engine assembly, rated at 312 kW (418 shp) with a propeller rpm range of 1,450 to 1,810. Its Rolls-Royce 250-C20 (usually C20S) turboshaft engine is combined with Soloy's propeller gearbox and other components to produce a turboprop configured for single-engined aircraft. Its high thrust line and rear inlet suit it particularly to bush aircraft and float seaplanes. The engine assembly includes propeller governing and overspeed systems, and a self-contained lubrication system. Customised models are available in pusher configuration and can also utilise the 485 kW (650 shp) Rolls-Royce 250-C30 engine in either tractor or pusher configurations.

Current appplications for the **Turbine Pac 780-1000-1** are the Cessna 206 and 207 retrofit modifications. Propeller drive ratio is 3.323:1. By 2000 Soloy had sold `well over 400' Model 250 conversions.

Dimensions

Length overall	1,220.2 mm (48.04 in)
Max width	754.38 mm (29.7 in)
Max height	754.89 mm (29.72 in)

Weight

Dry	143.9 kg (317.3 lb)

Performance Rating

T-O, S/L	311.7 kW (418 shp) at 1,810 propeller rpm

Specific Fuel Consumption

T-O and cruise	110 µg/J (0.650 lb/h/shp)

Contract Price

Inclusive of Allison 250-C20 engine, complete installation price for a Cessna 206/207 is about US$400,000.

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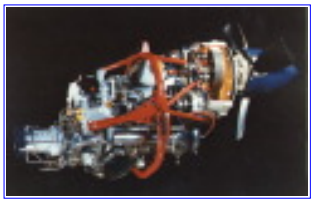
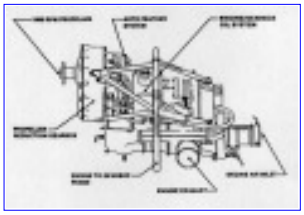


Turbine Pac 780-1000



Features of Turbine Pac 780-1000

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

GE AIRCRAFT ENGINES

GENERAL ELECTRIC CF6-80A

In November 1977 GE launched a programme to simplify the CF6-50. The objectives were nothing less than improved performance, reduced sfc, smaller overall dimensions and reduced weight. The chief design changes were to eliminate the turbine mid-frame and reduce the length of both the diffuser downstream of the HP compressor and the combustion chamber. Volvo Flygmotor joined the programme with a 9 per cent share.

CF6-80A

The first engine of the family ran in October 1979, and was certificated in October 1981. Option for 767.

CF6-80A1

Minor configuration changes, same rating.

CF6-80A2

Mechanically identical to A but increased ratings. Option for 767.

CF6-80A3

Mechanically identical to A1 but increased ratings. Option for A310.

At 30 April 1997 a total of 165 A310 and 767 aircraft were flying with Dash-80A engines, which had then accumulated 11.3 million hours. The high-time engine was on 45,495 hours, and the high-cycle engine had recorded 29,617 cycles. Engine-caused SVR was 0.161, equivalent to one per 6,211 hours. Dispatch reliability (like other figures, a 12 month rolling average) was 99.94 per cent.

Previous descriptions apply with the following differences:

Fan

Similar to CF6-50 series but with revised aerodynamics for higher efficiency, and modified structure for increased birdstrike resistance. Mass flow (A/A1) 651 kg (1,433 lb)/s, (A2/A3) 663 kg (1,460 lb)/s. BPR (A/A1) 4.7, (A2/A3) 4.6.

HP Compressor

Incorporates bore cooling for active control. Horizontally split full-length steel casing with insulated aft stages and shorter diffuser section. OPR 28.4.

Combustion Chamber

Improved rolled-ring type, 152 mm (6.0 in) shorter, mounted on rear flange.

HP Turbine

Mid-frame and one main bearing eliminated, casing with active clearance control. Stage-1 rotor blades being improved by cast-in tips and internal aluminide coatings.

LP Turbine

Revised design with active clearance control.

Reverser

Fan reverser unchanged on A1/A3; A/A2 reverser by Boeing.

Accessories

Main drive gearbox moved from core to fan case.

Starting

Pneumatic starter on front of fan-case gearbox.

Dimensions

Length overall	3,998 mm (157.4 in)

Weight, Dry

Basic engine, without reverser:	
A1, A2	3,854 kg (8,496 lb)
A1, A3	3,819 kg (8,420 lb)

Performance Ratings

Max T-O, S/L, uninstalled:	
CF6-80A, A1	213.5 kN (48,000 lb st)
CF6-80A2, A3	222.4 kN (50,000 lb st)
Max cruise, uninstalled, 10,670 m (35,000 ft), M0.85, flat rated to ISA +10ºC:	
CF6-80A, A1	45.9 kN (10,320 lb st)
CF6-80A2, A3	46.6 kN (10,477 lb st)

Specific Fuel Consumption

T-O, as above	9.74 mg/Ns (0.344 lb/h/lb st)

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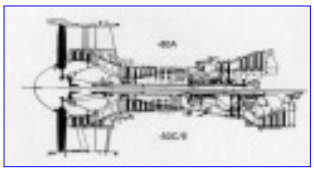

Longitudinal sections through CF6-50C/E and CF6-80A

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

CFE - CFE COMPANY

111 South 34th Street, PO Box 62332, Phoenix, Arizona 85082-2332
Tel: (+1 602) 231 45 70
Fax: (+1 602) 231 50 30

President: Kenneth Nusky

Director of Operations: Karl Johnson
Tel: (+1 602) 231 32 24

This company was formed jointly by Garrett Engine Division (now Honeywell Engines, Systems & Services) and General Electric Aircraft Engines in June 1987. It is managing all phases of the development, manufacture, marketing and support of the CFE738 turbofan.

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 22 March 2002

Jane's Aero-Engines 02

HONEYWELL

HONEYWELL ATF3

US military designation: F104-GA-100

With the joint objectives of maximum propulsive efficiency and minimum noise, the former AiResearch Manufacturing Co, at its Torrance (Los Angeles) plant (a Honeywell predecessor), adopted a unique geometric layout with this engine. This company later became part of AlliedSignal. Aimed at business jets and commuter transports, it was initially a company venture. Its development was later funded by the US Air Force, for the large Compass Cope unmanned vehicle.

A three-shaft engine, the ATF3's unique feature was cross-compounding of the LP and IP systems. Air is rammed in at the direct forward-facing inlet and passes through the fan. Most is then expelled through the fan duct. The proportion entering the core then passes successively through the axial IP compressor, a long and narrow annular duct, to make a 180º turn to enter the centrifugal HP compressor (travelling from the rear to the front), the combustion chamber (making two further 180º changes in direction), the HP turbine, the IP turbine (which drives the fan), the LP turbine (which drives the IP spool), and leaves via eight cascade boxes where it makes a final 180º turn to mix with the air in the fan duct.

ATF3-6-4C

This version powers the Dassault HU-25A Guardian of the US Coast Guard. As the 6-2C it was FAA certificated in May 1981.

ATF3-6A-4C

This very similar engine was selected in May 1976 to power the Dassault Falcon 200 and Gardian. FAA certification was completed in May 1982.

Type

Three-shaft cross-compound turbofan.

Intake

Direct pitot fixed type, without guide vanes.

LP Compressor

Single-stage fan of titanium alloy, with part-span shrouds, driven by IP turbine. Mass flow 73.5 kg (162 lb)/s. Bypass ratio 2.8.

IP Compressor

Five-stage axial, each with its own bladed disc of titanium, driven by the LP turbine. Delivers via eight slim tubes into an annular duct surrounding the remainder of the core.

HP Compressor

Single-stage centrifugal, of titanium alloy, with aft-facing entry. Driven by HP turbine. OPR (ATF3-6) 22.8, (ATF3-6A) 21.3.

Combustion Chamber

Annular, with double reverse flow of gas.

HP Turbine

Single stage, with air-cooled nozzles and rotor blades, with short shaft driving HP compressor.

IP Turbine

Three stages with shrouded rotor blades, coupled by long central shaft to LP compressor (fan).

LP Turbine

Two stages with shrouded rotor blades coupled by conical shaft to IP compressor.

Jetpipe

Gas escapes from LP turbine via eight ducts containing two sets of 90º cascade vanes which reverse the gas flow to mix with the fan air flow. The combined jet is discharged to atmosphere through a single annular nozzle surrounding the core.

Mounting

Two-plane pick-up system.

Accessories

Three drive pads on aft-facing rear gearbox driven by HP shaft provide for a starter/generator, hydraulic

pump and one spare. Accessories are cooled by fan air exhausted through a nozzle at the tip of the core fairing.

Starting

Electric, with pneumatic option.

Control System

Electromechanical, incorporating solid-state computer, with emergency manual back-up system.

Oil System

Self-contained hot-tank type, tank integral with gearbox.

Dimensions

Length:	
ATF3-6	2,611 mm (102.8 in)
ATF3-6A	2,590 mm (101.97 in)
Diameter	856 mm (33.7 in)

Weight, Dry

ATF3-6	528.5 kg (1,165.2 lb)
ATF3-6A	538.1 kg (1,186.4 lb)

Performance ratings

T-O, S/L, ISA, uninstalled, both	24.2 kN (5,440 lb st)
Cruise (12,200 m; 40,000 ft at M0.8)	5.10 kN (1,100 lb)

Specific Fuel Consumption

T-O, as above:	
ATF3-6	14.33 mg/Ns (0.506 lb/h/lb st)
ATF3-6A	14.25 mg/Ns (0.503 lb/h/lb st)
Cruise, as above (both)	22.83 mg/Ns (0.806 lb/h/lb st)

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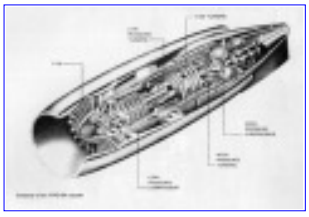
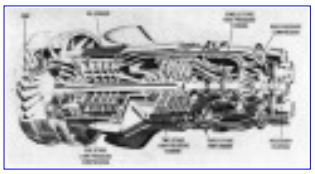


Cutaway of an ATF3-6A nacelle



Cutaway of the ATF3-6 (IP compressor called `low pressure', IP turbine called `fan turbine')

ATF3-6A

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AERO-ENGINES - TURBOPROP, UNITED STATES OF AMERICA

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE CORPORATION

ROLLS-ROYCE ALLISON T56

This single-shaft, constant-speed turboprop has sustained one of the longest unbroken production runs of any aero-engine in history. It was derived from the first Allison turboprop, the T38, first ran in 1949, and first flew in the nose of a B-17 in 1954. By that time the military T56 had been adopted for the Lockheed C-130 Hercules, and in support of this programme the USAF funded 3,000 hours of development flying with YT56 engines in Convair YC-131C (Convair-Liner) testbeds. Subsequently Allison developed the 501 commercial version (described separately), and the following military variants:

T56-A-7

T-O rating 2,800 kW (3,755 shp), 3,020 ekW (4,050 ehp). Powers C-130B, E, F.

T56-A-9

T-O rating 2,580 kW (3,460 shp), 2,796 ekW (3,750 ehp). Powers C-130A.

T56-A-10W

T-O rating 3,117 kW (4,180 shp), 3,456 ekW (4,500 ehp) with water injection. Dry T-O 2,800 kW (3,755 shp). Derived from Model 501 with reduction gear below the power section, not above, and with

seven-point suspension in nacelle. Powers Lockheed P-3A Orion.

T56-A-14

T-O rating 3,424 kW (4,591 shp), 3,661 ekW (4,910 ehp). Configuration as T56-A-10W. Powers P-3B and P-3C.

T56-A-15

T-O Rating 3,424 kW (4,591 shp), 3,661 ekW (4,910 ehp). Introduced air-cooled turbine blades. Powers C-130H family.

T56-A-16

T-O rating 3,424 kW (4,591 shp), 3,661 ekW (4,910 ehp). Powers US Navy C-130 versions.

T56-A-422

Powered E-2C, upgraded to Dash-425.

T56-A-423

T-O rating 3,424 kW (4,591 shp), 3,661 ekW (4,910 ehp). Powers US Navy C-130 versions.

T56-A-425

T-O rating 3,424 kW (4,591 shp), 3,661 ekW (4,910 ehp). Powers Grumman E-2C Hawkeye (pre-1987) and derived C-2A Greyhound.

T56-A-427

T-O rating 3,910 kW (5,250 shp). Increased rpm and digital control. Powers Northrop Grumman E-2C (post-1987).

501-D22A

Though designated as a Model 501, this is because it has civil certification. It is actually a commercial T56-A-15, with the high gearbox to suit the Lockheed L-100, commercial derivative of the C-130.

Total deliveries of all T56 versions, including the 501-D22A but not other 501 versions, had by the start of 2000 reached 16,900 with a very small number remaining on the line. Total flight time was then 221 million hours. Because it is a purely military engine the T56 retains the name Allison in its title, and support contracts are addressed to `Allison Engine Co Inc, dba (doing business as) Rolls-Royce Allison'. Most recent support contracts have concerned the A-427, a recent example being US$6,131,360 for 11,950 turbine vanes (stator blades).

Type

Single-shaft turboprop.

Intake

Air is rammed in at an inlet in the cowling below the spinner, leading up in a curved duct to the inlet on the engine. The latter is a one-piece magnesium-alloy casing with eight radial struts carrying the front bearing and rear end of the extension shaft driving the propeller. Bleed air anti-icing.

Compressor

Fourteen-stage axial. Series of 14 discs with rotor blades dovetailed in peripheries and locked by adjacent discs. Rotor assembly tie-bolted to shaft which runs on one ball- and one roller-type bearing. Fifteen rows of stator blades, welded in rings. Disc, rotor and stator blades and four-piece cast casing of stainless steel. Compressor inlet area 1,004 cm^2 (155.65 in^2). Pressure ratio (early) 9.5, (A-15 to A-423) 9.6, (A-427) 12. Mass flow (early) 14.5 kg (32.0 lb)/s, (A-15 to A-423) 14.7 kg (32.4 lb)/s, (A-427) 15.2 kg (33.5 lb)/s.

Combustion Chamber

Six stainless steel cannular-type perforated combustion liners within one-piece stainless steel outer casing. Fuel nozzles in forward end of each combustor liner. Primary ignition by two igniters in diametrically opposite combustors.

Turbine

Four stages. Each rotor comprises a stainless-steel Timken 16-25-6 disc carrying blades secured by fir-tree roots. In early versions the first three stages have blades cast in GMR-235 and the fourth forged S-816; TGT 971~C. From the A-15 the first stage has air-cooled blades; TGT 1,077~C. Speed under all flight conditions (except A-427) 13,820 rpm, (A-427) 14,239 rpm.

Jetpipe

Circular stainless steel, fixed area.

Mounting

Three-point suspension on power section, with a separate mount on each side of the propeller gearbox. Dash-10W and -14 have seven-point mounting similar to Model 501.

Output

Front of compressor drives a 711 mm (28 in) extension shaft housed in a load-bearing casing connecting the engine inlet to the reduction gear. The latter is housed in a magnesium-alloy gearbox braced to the engine by two inclined pin-jointed struts. The gearbox weighs (early versions) 204 kg (450 lb), (A-15 onwards) 249 kg (550 lb) and contains a primary spur gear followed by a planetary gear giving an overall reduction of (early) 12.5, (A-15 onwards) 13.54.

Accessories

One group mounted on remote gearbox under the compressor and driven off the HP shaft; a second group mounted on the rear face of the propeller gearbox.

Starting

Air turbine starter (usually Bendix-Utica) mounted on propeller gearbox. Dual Bendix-Scintilla high-energy ignition.

Control System

Hydromechanical, Bendix high-pressure fuel control. Automatic control of start and acceleration, and co-ordinated fuel flow, propeller pitch and TGT, and of water/alcohol injection in Dash-10W.

Fuel Specifications

MIL-J-5624, JP-4, JP-5, JP-8; (pre-A-15) 100/130 grade gasoline.

Oil System

Early versions, high-pressure dry sump system with Pesco dual-element pump supplying at 13.8 bars (200 lb/sq in); A-15 onwards, low-pressure Pesco pump delivering at 3.8 bars (55lb/sq in).

Oil Specifications

MIL-L-7808.

Dimensions

Length:	
(pre-A-15)	3,685.5 mm (145.1 in)
A-15 to A-425	3,716 mm (146.3 in)
A-427	3,711 mm (146.1 in)
Diameter (power section)	686 mm (27.0 in)
Height:	
A-7, A-9	1,016 mm (40.0 in)
A-10W, A-14	1,245 mm (49.0 in)
A-15, -16, -423, -425	1,133 mm (44.6 in)
A-427	1,227 mm (48.3 in)

Weight, Dry

A-7, A-9	746 kg (1,645 lb)
A-10W	848 kg (1,870 lb)
A-14	857 kg (1,890 lb)
A-15	838 kg (1,848 lb)
A-425	861 kg (1,899 lb)
A-427	880 kg (1,940 lb)

Performance Ratings

T-O, S/L: See model listing	
Normal (MIL):	
A-7, A-10W	2,782.5 ekW (3,730 ehp)
A-9	2,517 ekW (3,375 ehp)
A-14, -15, -16, -425	3,028 kW (4,061 shp), 3,255 ekW (4,365 ehp)

Specific Fuel Consumption

At T-O rating:	
A-7, A-10W	89.23 Ág/J (0.528 lb/h/ehp)
A-9, A-425	91.26 Ág/J (0.540 lb/h/ehp)
A-14, A-15	84.67 Ág/J (0.501 lb/h/ehp)
A-427	79.43 Ág/J (0.470 lb/h/ehp)

Contract Price

US Navy contracts for the T56-A-427, September 1997 US$29.55 million for eight; December 1999, US$113.4 million for 54; February 2000, US$113.0 million for 52.

UPDATED

T56-A-7

T56-A-15

T56-A-427

T56-A-15 power section

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE CORPORATION

ROLLS-ROYCE AE 3007

The AE 3007 turbofan is being developed to power regional airliners and medium/large business jets. The engine utilises the high-pressure spool from the T406/2100 with a new low-pressure spool.

The first AE 3007 went on test in July 1991. Additional development during 1992 included full altitude calibration, acoustic tests, initial endurance testing, and initiation of flight testing on a Cessna Citation VII prototype flying test bed, which first flew on 21 August 1992. By late 1999 the common core had operated for well over 1 million hours. AE 3007 demonstrated thrust has exceeded 42.26 kN (9,500 lb st), and sfc has consistently bettered specification. Acoustic testing has shown results dramatically below Stage III requirements.

Rolls-Royce has 14 facilities providing global capacity for engine repair and overhaul. The first to be certified to handle the AE 3007 are RR Canada at Montreal, Motores RR at São Paulo and RR East Kilbride, Scotland.

Since EIS in December 1996 the AE3007A has established an excellent record. At mid-1999 the only significant problem has been three IFSDs in the first week of 1999, which resulted in operators of 3007A and 3007C engines being ordered to perform cold-weather tests.

The fully rated T-O thrust of the AE 3007 is 38.26 kN (8,600 lb st). In its manned applications it is significantly derated:

AE 3007A1

T-O rating 33.72 kN (7,580 lb st). Selected in 1990 for Embraer EMB-145 (now ERJ-145). At that time,

the engines were to be hung under the wings, but the configuration finally adopted mounts them on the rear fuselage, with optional nacelle-mounted reversers. The prototype first flew on 11 August 1995, and by September 1996 this aircraft and three pre-series ERJ-145s had flown over 1,300 hours, completing certification.

Deliveries began in November 1996, and the AE 3007 entered service with Continental in April 1997. In 2000, Continental Express signed a US$700 million contract for fleet-hour support of 400 engines over 10 years. By February 2000, orders and options for the ERJ-145 had reached 673, of which 186 had been delivered. Total flight time on the AE 3007A1 and AE 3007A3 (see next) had at that time exceeded 1.1 million hours.

AE 3007A1P

T-O ratings up to 35.59 kN (8,000 lb st). Powers ERJ-145LR.

AE 3007A1/3

T-O rating 32.03 kN (7,201 lb st). Powers ERJ-135LR and ERJ-140. Also selected for ECJ-135 Legacy business jet.

AE 3007A3

T-O flat-rated at 31.14 kN (7,000 lb st). Selected in 1997 to power Embraer ERJ-135. The first ERJ-135 flew in July 1998, and EIS was in July 1999. By February 2000 orders and options for the ERJ-135 had reached 322.

In July 2000 the order book for AE 3007A engines passed the 2,100 mark. Their 2-millionth flight hour was expected later that year.

AE 3007C

T-O flat rated at 28.89 kN (6,495 lb st). Selected in 1990 for Cessna Citation X, the first of which flew on 21 December 1993. Nacelle-mounted target-type reverser standard. Achieved joint JAA/FAA certification on 28 February 1995. By September 1996, four test aircraft had flown over 6,500 hours and over 30 engines had been shipped to Cessna, all with a five-year/2,500-hour warranty. In February 2000, a Citation X set a bizjet round-the-world record, cutting the time from 47 hours to 41.

In July 2000, Rolls-Royce announced a US$103 million Power by the Hour® agreement for long-term maintenance for Executive Jet Aviation.

AE 3007H

T-O rating 36.88 kN (8,290 lb st). Modified to power Global Hawk UAV on 24-hour missions at altitudes up to 21,336 m (70,000 ft). Rolled out 1997, flight testing since spring 1998.

At the Paris Air Show in June 1999 Carmen Lloyd, RR President - Corporate and Regional Airlines, said ``The AE 3007 is a prime contributor to the tremendous expansion of the regional and corporate aircraft industry. Since 1995 we have shipped 588 engines, and the rate of delivery is accelerating to meet demand."

Type

Two-shaft subsonic turbofan.

Fan

Single-stage, direct drive featuring wide-chord, clapperless blades. Mass flow 118 kg (260 lb)/s. Bypass ratio of 5.0. Blades replaceable on aircraft. Diameter 978 mm (38.5 in). PMR15 composite fan duct made by Hispano-Suiza.

HP Compressor

Fourteen-stage axial flow, with variable inlet guide vanes and first five stator rows, all steel. Overall pressure ratio 23.

Combustion Chamber

Annular design with 16 airblast fuel nozzles. Dual capacitor-discharge ignition.

HP Turbine

Two-stage axial with air-cooled single-crystal blading.

LP Turbine

Three-stage axial uncooled design.

Jetpipe

Minimal, centred in single-piece full length composite bypass duct with provisions for thrust reverser.

Control System

Full-authority digital (FADEC).

Dimensions

Inlet diameter	978 mm (38.5 in)
Diameter overall	1,105 mm (43.5 in)
Length	2,705 mm (106.5 in)
Length with spinner	2,923 mm (115.1 in)

Weight, Dry

With bypass duct:	717.1 kg (1,581 lb)

Performance Ratings

S/L, T-O	See model listing

Specific Fuel Consumption

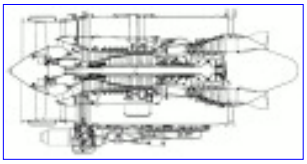
S/L, T-O	9.35 mg/Ns (0.33 lb/h/lb st)

Contract Price

May 1998, engines for 25 + 50 option ERJ-135, US$275+ million; February 1999, for six ERJ-145 plus six options, up to US$65 million; February 2000, engines for 75 ERJ-145 and 25 ERJ-135, US$490 million; July 2000, engines for 50 ECJ-135, US$160 million.

UPDATED

AE 3007



Longitudinal section through AE 3007 (1999)

AE 3007A1 (2001)

AE 3007

AE 3007C

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AERO-ENGINES - TURBOPROP, UNITED STATES OF AMERICA

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE CORPORATION

ROLLS-ROYCE AE 2100

The AE 2100 turboprop has been developed for high-speed regional aircraft as well as military transports and maritime patrol aircraft. It combines the T406 power section with a new propeller reduction gearbox based on the T56/501 design. Though thermodynamic power is 4,474 kW (6,000 shp), in each of its first three applications it is derated, in one case to little more than half the power available. This confers exceptional hot/high performance.

AE 2100A

Flat-rated at 3,096 kW (4,152 shp) driving Dowty R381 six-blade propeller at 1,100 rpm. The first AE 2100 went on test in June 1988. A prototype engine successfully completed flight testing on a P-3A aircraft in late 1990. The prototype consisted of a T406 power section, the new reduction gearbox and a flange-mounted Dowty Aerospace six-bladed propeller. Ten flight test and 33 production engines had been delivered by January 1995. FAA certification for the Saab 2000 was gained on 23 April 1993. The Saab 2000 received JAA and FAA certification on 31 March 1994, and EIS was in September 1994. By December 1998, AE had delivered 143 engines to Saab. By October 1999, flight time in scheduled service was 750,000 hours, with about the same number of cycles. Early problems with the FADEC sensor signals have been resolved, and dispatch reliability in late 1999 was 99.85 per cent, with a goal of 99.94 within five years. In late 1998, Saab discontinued production of the Saab 2000, with sales standing at 55 aircraft.

AE 2100C

Flat-rated at 2,386 kW (3,200 shp) or 2,684 kW (3,600 shp), driving Dowty slow-running R384 propeller. Certificated December 1993. Powers the IPTN 250-100, the first two of which flew in 1996-97. Two further prototypes due to fly in 1998 have been held up by lack of funds and other problems in Indonesia. Despite technical aid from Boeing, in mid-2001 (when certification was due) little progress was being made.

AE 2100D2

Same engine and propeller as AE 2100D3, but for Alenia/Lockheed Martin C-27J. Flight testing from 24 September 1999, with 14 engines delivered by late 1999. Aircraft certification was achieved in June 2001.

AE 2100D3

Flat-rated at 3,424 kW (4,591 shp). Engine for Lockheed Martin C-130J matched with Dowty R391 six-bladed propeller. Test installation on C-130J demonstrator rebuilt by Marshall Aerospace, on flight test from 24 March to May 1994. In production for UK RAF and other air forces. This aircraft received certification in September 1998. In August 1999 Indianapolis shipped the 300th D3 engine; this represented the 1,105th delivery from the Common Care family, the other members of which are the AE 1107 and AE 3007. By mid-2001 orders for all versions of the C-130J totalled 157, almost half of which had been delivered, and numerous further sales are in prospect. All these are for new-build aircraft.

AE 2100J

Flat-rated at 3,424 kW (4,591 shp). Selected in 1996 with Dowty R414 six-blade propeller as part of major upgrade for nine ShinMaywa US-1A of Japan Maritime Self-Defence Force to convert them to US-1A Kai. Two engines delivered 1999 to assist conversion. Further funding was denied until in August 2001 ¥1.6 billion (US$13 million) was authorised for conversion of the first aircraft. This funds three more engines, which with one of the originals (the other serving as a spare) should fly the first Kai aircraft in August 2003, for JMSDF service from June 2006. The plan now is to upgrade six further aircraft and, if funding permits, build seven more from scratch.

The AE 2100 is also a candidate engine for P-3 Orion fleet modernisation, and for several high-speed 70-passenger projects. A global overhaul capability is being built up, the first AE 2100 centre being East Kilbride, Scotland.

Type

Free turbine, axial flow turboprop.

Compressor

Fourteen-stage, axial flow, with variable inlet guide vanes and first five stator rows. Mass flow 16.33 kg (36.0 lb)/s. Pressure ratio 16.6.

Combustion Chamber

Annular design with 16 airblast fuel nozzles and two high-energy igniters.

HP Turbine

Two-stage axial design with air-cooled vanes and single-crystal first-stage blades and solid second-stage blades.

Power Turbine

Two uncooled stages on straddle-mounted shaft. First LP vane provides mounting for thermocouples.

Output

New design of gearbox saving 68 kg (150 lb) weight, with life of 30,000 hours. Alternator on rear face.

Accessories

Aluminium gearbox mounted under air inlet, driven off front of compressor. Provides drives on rear face for starter, fuel measuring unit and oil pump. Front drive for permanent-magnet alternator providing power for FADEC.

Control System

Full-authority digital (FADEC), co-ordinating engine and propeller functions with single-lever control.

Dimensions

Height	1,151 mm (45.3 in)
Core diameter	622 mm (24.5 in)

Weight, Dry

	702.2 kg (1,548 lb)

Performance Ratings

Thermodynamic power	4,474 kW (6,000 shp)
S/L, T-O	See model listing

Specific Fuel Consumption

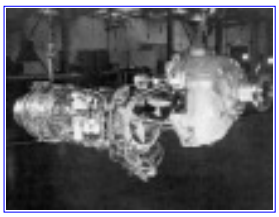
S/L, T-O	69.31 µg/J (0.41 lb/h/shp)

Contract Price

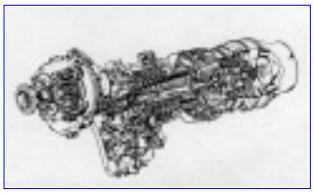
December 1999, engines for 12 C-27J `up to US$30 million'.

UPDATED

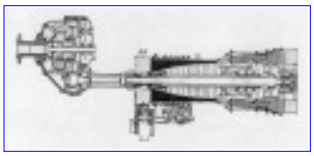
AE 2100 development engine



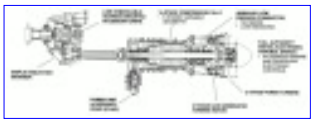
AE 2100 production engine



Cutaway drawing of AE 2100

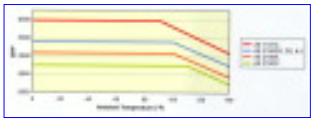


Longitudinal section through AE 2100



Longitudinal section through AE 2100D3 (2001)

AE 2100 production engine



T-O power of the four related AE turboprops (2001)

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 11 January 2002

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ROLLS-ROYCE - ROLLS-ROYCE CORPORATION

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Rolls-Royce plc, a pioneer of gas-turbine technology for aerospace, power generation and marine propulsion, acquired Allison Engine Company in March 1995 and has now fully integrated Allison into the worldwide Rolls-Royce organisation.

The Indianapolis, Indiana, company was originally founded as a machine shop by James A Allison in 1915. Two years later early fame was gained through the building of Liberty aero-engines during the First World War. Subsequently, the 12-cylinder, liquid-cooled V-1710, became one of the most successful engines of the Second World War, powering the P-38, P-40 and early versions of the P-51 Mustang.

Following the Second World War, the company not only continued to refine its turboprop technology, but also assumed design and production responsibilities from GE for the Whittle-designed J33 and J35 gas-turbine engines. These engines would power early US Air Force aircraft such as Lockheed's P-80 Shooting Star and the Republic F-84 Thunderjet. Its most popular turboprop, the T56, became synonymous with the C-130 Hercules heavy military airlifters.

The late 1950s saw design and development of the first of the company's many helicopter engines, beginning with the T63 and T703, followed by the Model 250 toward the end of the next decade. These engines established the benchmark for this organisation to become the world leader in gas-turbine engines powering light helicopters. More than 28,000 of the latter engine have been produced. Today that standard continues as the Rolls-Royce Allison Model 250 powers the latest offerings from Bell Helicopter Textron and Boeing Helicopters. A partnership with AlliedSignal Aerospace resulted in the design and development of the LHTEC T800/CTS800 engines for civil and military applications.

In December 1985, the company's T406 was chosen to provide propulsion for the US Marine Corps' revolutionary V-22 Osprey tilt-rotor aircraft, which takes off and lands like a helicopter then flies like an aeroplane. Derived from the proven T56 and Model 501 engines, the T406 turboshaft's common core would later provide the baseline for newer, more efficient and more powerful turboprop (AE 2100) and turbofan (AE 3007) engines.

The Rolls-Royce AE 3007 turbofan has the unique distinction of powering the two best-selling regional jets, the Embraer RJ-145 and RJ-135; the world's fastest business jet, Citation X from Cessna; and Northrop Grumman Ryan Aeronautical's state-of-the-art high-altitude unmanned aerial surveillance vehicle, Global Hawk. The turboprop engine heritage moves forward with the Rolls-Royce AE 2100 engine for the newest `J' version of the C-130 transport and the C-27J medium airlifter being jointly developed by Lockheed Martin and Alenia.

With an eye toward the next generation of military stealth fighters, development continues on a shaft-driven LiftFan® to propel Lockheed Martin's Joint Strike Fighter (JSF) for the US Marines and US Navy. Additionally, the company is participating in an alternative JSF propulsion programme with GE on the F120-FX.

In addition to the engineering and manufacturing operations, Rolls-Royce activities at Indianapolis include an Aero-Engine Repair and Overhaul operation, along with a facility in Evansville, Indiana, primarily dedicated to metal forming. Model 501K, 570, and 601 industrial engines are also produced in Indianapolis.

For contractual reasons, engines for the US military are designated Rolls-Royce Allison. Civil engines do not retain the Allison name.

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 11 January 2002

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GE AIRCRAFT ENGINES

GENERAL ELECTRIC GE90

Boeing's 1989 decision to launch the 777 prompted GE Aircraft Engines to produce an all-new propulsion system for the largest commercial jets. This `clean sheet of paper' engine, the GE90, was intended to set completely new standards of high thrust, low fuel burn and environmental acceptance. After assembling a multinational risk-sharing team, major players in which are SNECMA (25.25 per cent), FiatAvio (7.0 per cent) and IHI (8.66 per cent), leaving GE to bear 59.09 per cent, the new engine was announced on 16 January 1990.

GE stated that the GE90 was to meet all thrust requirements in the band from 333.6 kN (75,000 lb st) to 444.9 kN (100,000 lb st), ``in order to power all new and derivative wide-body aircraft that may enter the market in the mid-1990s''. It has been developed so that the same basic engine can be offered at the following thrust levels:

GE90-76B

T-O rating 339.9 kN (76,400 lb st) to 32.8ºC. For 777-200.

GE90-85B

T-O rating 376.8 kN (84,700 lb st) to 30ºC. For 777-200 and -200IGW. By 2 September 1996, six aircraft were in service and engine flight hours totalled 19,258 (high-time engine 2,483 hours, high-cycle 915). Engine-caused SVR was 0.052, equivalent to one visit in 19,231 hours, and dispatch reliability 99.97 per cent.

GE90-90B

T-O rating 400.4 kN (90,000 lb st). For 777-200IGW. This engine can be upgraded by the GE90PIP (Product-Improvement Program), which was developed in parallel with the GE90-94B described below. The PIP upgrade kit includes a new HPC (high-pressure compressor) with aerodynamically improved 3-D blading and with the final (10th) stage removed. This new HPC has since become a key building-block in the GE90-115B, described later. Other PIP components are fan outlet guide vane sealing, active clearance control optimization in the HP turbine, and clearance reduction in the LP turbine. By mid-May 2001 four customers had purchased PIP kits for a total of 96 GE90-90B engines, at a cost of more than US$160 million. The kit is claimed to reduce cruise fuel burn by 1.6 per cent and to increase EGT margin by not less than 20ºC. When first marketed, in 1999, GE also claimed a cumulative noise advantage on the 777, relative to competitor engines, of 6-7 EPNdB. A second upgrade package is now available to increase the engine's thrust rating to the same level as the -94B. According to Chaker Chahrour, GE90 Project general manager, ``The PIP may pay for itself in two years or less, and the customers are thrilled".

GE90-92B

T-O rating 409.3 kN (92,000 lb st). Improved HPT and LPT (high-pressure and low-pressure turbines), as detailed in the accompanying section drawing. For 777-300.

GE90-94B

T-O rating 418.2 kN (94,000 lb st). This engine is essentially a -90B incorporating the GE90PIP. In August 1999, the first 94B engine began a 100-hour test programme on the GEAE 747 at Mojave, putting the engine on track for certification testing in October. This flight programme was completed in 19 flights totalling 95 hours in September 1999, by which time a second engine had completed 500 cycles of endurance testing at SNECMA Villaroche. The original flight engine was sent in December 1999 for ice testing at GE Peebles. FAR Pt 33 certification was due in April 2000, and achieved on 9 June. The first 777 powered by 94B engines (an Air France aircraft) flew on 12 June and completed the flight-test programme on 31 July. FAR Pt 25 certification of this aircraft/engine combination was achieved in the fourth quarter of 2000, and EIS with Air France (derated to 400.4 kN, 90,000 lb st) took place in November 2000.

GE90-100B

T-O rating 444.9 kN (100,000 lb st). Originally intended for later 777 versions, this engine was shelved by GE in March 1998, on the grounds that no clear business case could be made out for it.

Despite shelving the GE90-100B, in March 1999, General Electric proposed that an even further-uprated GE90 (see next) should exclusively power the projected 777-200X/300X (Pratt & Whitney made the same suggestion for its PW40115, while Rolls-Royce, which at that time had much more powerful engines on test, did not).

GE90-115B

To meet propulsion requirements of future 777 aircraft, this engine has a target T-O rating of 511.6 kN (115,000 lb st). It was designed to incorporate the improved PIP compressor of the GE90-94B as well as a larger fan with blades of toughened composite material, blade-out load-reduction features, improved fan mid-shaft material (GE1014) to transmit the increased torque without the need to increase diameter, an HP turbine with 3-D aerofoils, and an LP turbine with low-solidity (reduced-chord) aerofoils. These technologies were validated by early 2000, in order to support the development of the GE90-115B engine for FAR/JAR33 certification in 2002 and EIS in September 2003.

On 7 July 1999, GE announced that Boeing had selected this engine as the *exclusive* power plant for the proposed 777-200X and 777-300X and all other future 777 growth versions. GE's McNerney naturally called

this ``one of the most significant wins in our long history'' and said that ``over the next two decades it could be worth US$15 billion to GE Aircraft Engines''. Boeing's Mulally said the GE offering had been chosen ``because it best met the overall evaluation criteria, including technical, schedule, customer service and business requirements''. In fact, a key element in the deal was massive investment by GE in the 777 growth programme. Later in July, Deborah Hopkins, then Boeing Chief Financial Officer, apparently surprised at widespread criticism of the exclusivity deal, said the GE engine had been chosen ``because it is quieter and more environmentally friendly'', though as the engine did not exist this statement was a mere estimate.

A Dash-94B engine incorporating the improved HP compressor (see GE90-94B) began flight testing in GE's 747 at Mojave in August 1999. At Wright-Patterson AFB tests were in progress on various fan blades strengthened by different arrangements of titanium and graphite-epoxy pins aligned along the Z-axis. These blades were manufactured by a hand-layup process and, though of the size and radial profile of the existing (123 in fan) blade, are expected to show how the blades can be strengthened to meet the much greater demands of the GE90-115B engine. The latter has larger blades, with forward-swept tips, rotating 10 per cent faster to give a redline tip speed of about 443 m (1,453 ft)/s. The larger fan increases bypass ratio to about 8.9. In July 1999 GE said it believed it could certificate the new engine in 24 months.

Engine history

The first complete GE90 engine went on test on 3 April 1993, achieving a then-record thrust of 468 kN (105,400 lb st). Flight testing began with an engine installed on a 747 in December 1993.

FAA certification was planned at a thrust level of 376.8 kN (84,700 lb st) in May 1994, to enable 777 EIS to be achieved in the third quarter of 1995 at the 76B rating. An HP turbine blade failure was among problems which delayed initial FAA certification until February 1995. Subsequently, problems were encountered with the requirements for birdstrike (8 lb bird), leading to a grounding order in June 1995. Additional problems included compressor surge and rotor tip rubbing. Icing trials were also protracted. The installation on the 777 finally achieved FAA/JAA certification in November 1995. Russian certification followed in February 1998.

The first GE90 engine to enter revenue service, derated to 337 kN (76,000 lb st), was delivered to launch customer BA (British Airways) in November 1995. By April 1996, BA and China Southern Airlines had logged 9,000 hours, with no IFSD but one engine-attributable delay. The GE90-92B was FAA certificated in July 1996. In August 1996, the GE-powered 777 completed a demanding 1,000-cycle flight programme for 180-min ETOPS. The reverser, difficulty with which caused BA to ground its 777 fleet in February 1996, is part of the airframe, not a part of the engine.

In February 1997, the GE90-92B became the first engine to enter service on the 777-200IGW (with launch customer BA, derated to 400.4 kN, 90,000 lb st). In March 1997, the airline suspended transatlantic ETOPS operations when chip detectors revealed debris traced to bearings in the accessory gearbox. GE redesigned the gearbox and BA reduced inspection interval. BA and China Southern also removed five engines to rectify compressor damage caused by ingesting cracked bleed ducts. Apart from this, GE claims ``outstanding performance in the first 20 months of revenue service . . . logging more than 30,000 revenue hours with a 99.97 per cent dispatch reliability''.

By February 1998, the total flight time had climbed to 170,000 hours. According to GE, the GE90 has ``maintained an industry-leading dispatch reliability rate of 99.98 per cent, while delivering the lowest specific fuel consumption, the lowest noise levels, the lowest total emissions and the best performance retention of the engines powering the 777''. In March 1998, an engine of a 777-200IGW of British Airways suffered a catastrophic failure of the LP turbine on takeoff, which was aborted. The failure was contained, but exhaust debris required closure of the runway. This event was a factor in the airline's decision to buy future 777s with a competitor engine. The French certification authority then cut back Air France ETOPS approval from 180 to 120 minutes following an IFSD and diversion of a scheduled service.

In 1998, the in-service fleet of GE90-engined aircraft increased from 26 to 58. Beyond the PIP could be an engine with a new and simpler (not double-annular) combustor. Hukam Mongia, GE manager of combustion technology, said the objective was to reduce oxides of nitrogen by 30 per cent and that the new combustor could be tested on a GE90 in mid-2000.

By March 2000, the GE90 had been selected by 12 customers to power 112 firm and 21 option 777 aircraft.

Total flight time had reached the 1,000,000 hour mark, at which point the dispatch reliability was 99.93 per cent and the IFSD rate 0.012. Each month GE90 engines log some 40,000 hours in over 10,000 flights, of which about 1,900 are ETOPS. GE particularly emphasises that since EIS in November 1995 no composite fan blade has been scrapped, though many have been removed to have the titanium leading edge repaired.

By late July 2000 a total of 90 GE-engined aircraft was in service. The total of engine flight hours had reached 1,325,844, in 305,254 cycles. In that time there had been five engine-caused IFSDs, three engined-caused ATOs and 14 UERs (out of 128 engine removals in all). The high-time engine was on 13,277 hours.

The PIP (see above, under GE90-94B) was at that time roughly on schedule, with FAR-33 certification due in 3rd-quarter 2000 and FAR-25 certification duel in the 4th quarter. The 3-D HPC was showing a `one point' gain in efficiency, equivalent to a 10ºC EGT margin and a 0.6 per cent improvement in sfc. On-wing experience with engines in service included an average-engine margin to the Max-Cruise Redline of 1,015ºC of 102ºC for the 85B engine and 60ºC for the 90B. The PIP engines, delivered from November 2000, were expected to improve these figures to 120ºC and 78ºC, and to add the 94B engine with a margin of 73ºC.

By July 2000, most of the new features of the 115B engine had been demonstrated, and some had completed rig testing. The fan blade to become available in prototype form in August 2000, and impact testing was expected to be completed in January 2001. In first-quarter 2000, fan blades in radial, `Hi-flow' and forward-sweep shapes were manufactured, and these were tested for performance on a Boeing rig in July, with acoustics testing following in August 2000. The novel load-reduction feature was subjected to blade-out and rub-simulation testing before June 2000, with testing on simulated 115B hardware due in early 2001. By July 2000, testing of the nine-stage HPC, with Stage-4 VSV removed, was essentially complete, with VSV schedule optimised. The first new fan mid-shaft, made of Daido steel (called GE 1014 alloy), was finish-machined at IHI in Japan in March 2000. At that time the planned number of test cycles was 14,400, following 17,876 on the original 76B/85B.

The first GE90-115B engine went on test on 16 November 2001. According to Project Manager Chahrour, ``we put the engine to test on Friday, set a new thrust record on Sunday and broke it again on Monday''. The record thrust was announced as 535.25 kN (120,316 lb st). After this burst of power, the engine ran steadily at 522.48 kN (117,446 lb st). Flight testing in the company 747 is due to begin in early 2002, and FAA/JAR certification is scheduled for later in the same year.

In early 2001 total flight time on GE90 engines passed the 2,000,000-hour mark. By May 2001, because of the exclusivity deal, the GE90 had become the best-selling engine on the 777. Altogether, by that time GE90 engines had been selected by 17 customers to power a total of 201 firm and 52 option 777 aircraft. In June 2001 a total of 101 aircraft were in service, and dispatch reliability had climbed to an acceptable 99.93 per cent. By December 2001 total flight time had reached 2.3 million hours, but in future this total should rise more sharply because of the exclusivity deal.

The following refers generally to GE90 versions prior to the -115B, but lists the enlarged engine's main differences:

Type

Two-shaft turbofan of high bypass ratio.

Fan

Single stage. The largest in service (diameter 3,124 mm, 123 in); 22 shroudless wide-chord blades made of intermediate-modulus graphite/epoxy composite, with replaceable titanium leading-edge guards and a polyurethane coating on the concave side. Each blade weights ``about 15.4 kg (34 lb)''. This fan has the highest bypass ratio of any current engine, varying with rating from 8.3 to 8.6. The GE90 engine cycle is claimed to result in a 10 per cent reduction in specific fuel consumption, compared with earlier engines, and noise levels 23 EPNdB lower than FAR.36 Stage 3 on a cumulative basis. The outlet guide vanes are structural, and the fan case incorporates a pylon mount with an integrally stiffened system which bypasses loads around the core. This is expected to deliver improved performance retention. GE performed a second FBO (Fan-Blade Off) test to evaluate the new aluminium case used on all production GE90 engines. Mass flow: GE90-76B, 1,361 kg (3,000

lb)/s; GE90-85B, 1,415 kg (3,120 lb)/s; GE90-90B, 1,449 kg (3,195 lb)/s; GE90-92B, 1,461 kg (3,221 lb)/s; GE90-94B, 1,483 kg (3,269 lb)/s. The GE90-115B has a fan with the same number of blades (22) made in advanced material (IM7/8551) with completely different curvaceous shape. Though fan diameter is increased to 3,256.3 mm (128.2 in), the diameter of the case is increased by only 38 mm (1.5 in). Blade average chord is increased from 485 mm (19.1 in) to 574 mm (22.6 in), and blade weight from 15.15 kg (33.4 lb) to 21.18 kg (46.7 lb). Mass flow is 1,641 kg (3,617 lb)/s, a record by a wide margin.

LP Compressor

Three stages rotating with the fan, and thus at low rpm. Moderate pressure ratio and low noise features.

HP Compressor

Ten stages (-90 and -94, nine stages), with variable inlet guide vanes and stators 1 through 5. This compressor is a scale of that developed for the GE/NASA E^3 (Energy Efficient Engine), which aimed at establishing a technology base for engines of the 1990s. It is claimed to result in a shorter, lighter and stiffer engine, with a high overall cycle pressure ratio, maintained by the engine's straddle mounting already referred to. Pressure ratio 23, giving an engine OPR of 39.3 to 45.5.

Combustion Chamber

Annular, in effect two concentric chambers, called by GE a dual-annular combustor (DAC). The inner and outer chambers are each perforated by many thousands of holes drilled by laser at an acute angle. Each has its own ring of 30 airspray fuel nozzles.The double annulus has been specifically designed for minimum NO_x emissions. During starting and low-power operation, the engine is fuelled by the outer nozzles only. At take-off and during cruise, both rings of nozzles are in operation. As power is reduced, the inner combustor annulus is progressively shut down, to optimise operability and reduce emissions.

HP Turbine

Two stages of small diameter with cast monocrystal blades held in powdered-metal (René R88DT) boltless discs with smooth side plates for minimum windage losses. The casing has active clearance control.

LP Turbine

Six stages of large and increasing diameter, held in centreless rings. The casing has active clearance control, which on both turbines is governed by Parker valves. The use of six stages (previously seen only on the TF39) makes possible a low stage loading for improved efficiency and reduced noise.

Fan Duct

As the illustrations show, there is only a short duct, incorporating the reverser, downstream of the fan case and outlet guide vanes. There is no mixing of the fan and core jets.

Reverser

Not part of the engine. The nacelle incorporates a translating cowl and cascade-type reverser. No core reverser.

Jetpipe

Fixed-area, terminating in peripheral nozzle around large central plug.

Control System

FADEC with enhanced diagnostics and on-wing software upgrade. Controls fuel flow, variable IGV scheduling and active clearance control.

Dimensions

Length, flange to flange	4,902 mm (193 in)
Length overall	5,182 mm (204 in)
Diameter of case	3,404 mm (134 in)

Weight, Dry

GE90-85B, -90B, -92B basic engine	7,559 kg (16,664 lb)

Performance Ratings

T-O: See model listing	
Max climb:	
76B and 85B, 10,670 m; 35,000 ft at M0.8	83.68 kN (18,100 lb)
90B and 92B, same conditions	86.00 kN (18,600 lb)
Max cruise:	
76B and 85B, same conditions	77.85 kN (17,500 lb)
90B and 92B, same conditions	85.07 kN (18,400 lb)

Specific Fuel Consumption

Not stated, but said to be ``the lowest of any engine in its thrust class".

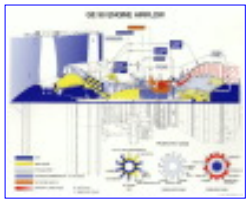
Contract Price

In May 2001 said engines (of different models) for 201 firmly ordered 777 aircraft were valued at more than US$6.7 billion, or US$33.3 million per shipset. The first 98 GE90-115B engines were priced at ``more than US$1.5 billion".

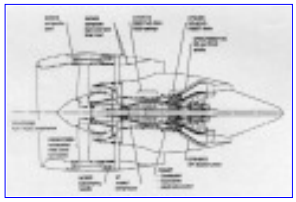
UPDATED

GE90-85B

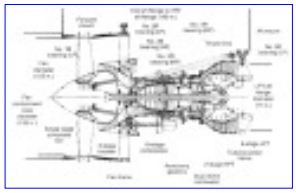
Cutaway drawing of GE90 nacelle



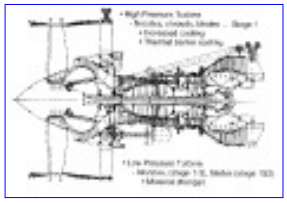
GE90 secondary air flows



GE90 features proven in previous GE engines

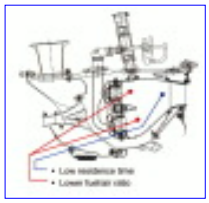


Features common to the original GE90-76B, -85B and -90B

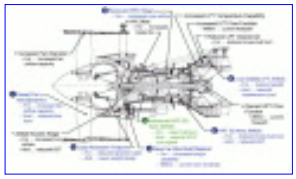


GE90-92B differences from the GE90-85B

GE90-92B (this is a real engine)
(2001)



GE90B dual-annular combustor
(2001)



GE90-115B features
(2001)

GE90-94B/-115B propulsion systems
(2001)

GE90-115B mockup
(2001)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 11 January 2002

Jane's Aero-Engines 11

GE AIRCRAFT ENGINES

GENERAL ELECTRIC F414

This afterburning turbofan is a direct development from the F404. Major differences are indicated in the accompanying drawing.

F414-GE-400

Baseline engine developed to power the US Navy's F/A-18E/F Super Hornet. This is an enlarged and even more capable derivative of earlier F/A-18 versions, originally designed at St Louis by McDonnell Douglas and now a product of Boeing. Likewise, the F414 engine is an upgraded derivative of the F404, incorporating technology from other GE engines including the F412, the engine designed for the Grumman A-12 stealth attack aircraft (which was terminated).

By June 1997, a total of 29 engines had run 10,220 hours, including over 1,900 in F/A-18E/F flight tests and development was 95 per cent complete. Described by the Navy as `a model development program', it has met or exceeded all objectives. The first engines for Navy/Marines aircraft were scheduled to achieve EIS in 2001. Navy orders are now unlikely to reach the original planned figure of 2,300, the total being likely to be 1,250 to 1,900, with completion of production in 2017. Additional engines are expected to be produced for export. In April 1997, following over 13,000 hours of testing of 32 development engines, the Navy awarded Limited Production Qualification. The LRIP I (Low-Rate Initial Production) batch of 27 engines is being delivered from July 1998 under a US$154,750,000 contract. In April 1998, the Navy awarded a US$189,646,240 modification and a further US$66,338,235 modification to provide additional funding for 46 LRIP II engines, followed by a further US$16,000,000 under LRIP III. In February 1999 GE Lynn was awarded a Navy contract for US$318,657,828 to definitize (complete the definition of) the LRIP III contract for 66 engines and related services. The first two production engines were delivered to Boeing St Louis in August 1998, at which time the F414 core had completed 25,000 hours of test time. Of this, 10,000 hours was on complete engines, and by March 2000 this

total had more than doubled. In that month, the engine had shared in the award to the F/A-18E/F of the Collier Trophy. The F414-GE-400 received Full Production Qualification in December 1998, and by late April 2000, 68 engines had been delivered, logging 4,673 hours in 3,172 flights. The F/A-18E/F will deploy to the *Abraham Lincoln* Battle Group in 2002.

Note: In 1997 the Navy suggested that, should JSF (see under International section) development move quickly, this next-generation aircraft could replace some or all of the F/A-18E/F programme. By 1999 this appeared unlikely.

F414BJ

Though merely a study programme, Dassault has selected an unaugmented version of the F414 for its supersonic business jet. The engine for this application would be rated at about 57.8 kN (13,000 lb st), but GE does not at present have plans to build and test such an engine.

Type

Two-shaft low-BPR turbofan with afterburner.

Fan

Preceded by 17 forward-sloping radial inlet struts and one row of variable stators. Three stages with larger blades handling mass flow up to 76.8 kg (169 lb)/s. OPR 30.0. First stage with removable blades, stages 2 and 3 blisks. Variable Stage 1 stators. Titanium case.

Compressor

Seven stages, in an advanced core adapted from the GE23A technology demonstrator, which was similar to that of the F412. First three stages blisks, variable inlet guide vanes and Stage 1 and 2 stators.

Combustion Chamber

Annular with coated multihole flame tube and 18 fuel nozzles.

HP Turbine

Again based on the F412; single-stage with air-cooled single-crystal nozzles and blades accepting TET increased by 111-167ºC.

LP Turbine

Single stage with air-cooled single-crystal nozzles and blades. Redesigned with advanced aerodynamics to drive larger fan.

Fan Duct

Upper and lower halves in black PMR-15 composite material.

Afterburner

Advanced design giving enhanced augmentation. Radial flameholder, with improved cooling and maintainability, derived from the YF120.

Nozzle

Options for circular or two-dimensional AVEN (Advanced Vectoring Engine Nozzle).

Control System

Advanced FADEC, in box under bypass duct.

Performance Ratings

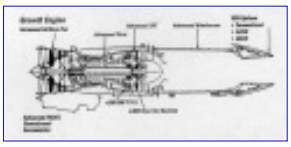
Max afterburner	97.86 kN (22,000 lb st)
Intermediate	over 55.6 kN (12,500 lb st)

Contract Price

The figures given above for LRIP contracts do not give an indication of the significantly reduced price of engines in full production.

UPDATED

F414-GE-400



Longitudinal section through the F414-GE-400 showing differences from F404-GE-402

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AERO-ENGINES - TURBOPROP/ - TURBOSHAFT, UNITED STATES OF AMERICA

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE

ROLLS-ROYCE MODEL 250

US military designations: T63 and T703

This free-turbine engine was developed under a US Army contract of June 1958 and first ran in April 1959. It is notable on several counts. It was the first of its size to go into mass production; previously it was widely thought that gas-turbine engines would not be economically viable below about 500 hp. It used an extremely small six-stage axial spool upstream of a centrifugal impeller. Not least, the centrifugal diffuser discharged via a slim tube along each side of the engine to a single relatively large combustion chamber arranged `back to front' with the fuel burner at the extreme rear of the engine.

Despite this, large orders for military helicopters ensured a successful programme. Gradually civil orders followed, an important proportion being for turboprops. By 1995 extensive research on advanced designs and materials, including low-cost ceramics for hot parts, had led to a new-generation concept, called World Engine. However, though this was feasible for industrial engines, the severe certification requirements for aero engines have so far kept this on hold. Today many Model 250 versions are still in full production, for a wide range of aeroplanes and helicopters. The following versions have been produced:

T63-A-5

Initial production turboshaft. T-O rating initially 186.5 kW (250 shp). Completed PFRT in March 1962, flight test in Bell UH-13R, FAA certification December 1962 and delivered to US Army as engine of Hughes OH-6A prototypes. It also powered the competing Bell OH-4A and Hiller OH-5A.

T63-A-5A

Uprated to 236.4 kW (317 shp). FAA certificated September 1965. Selected for production OH-6A, and for Navy remotely piloted Gyrodyne QH-50E, flat-rated at 188.3 kW (252.5 shp).

T63-A-700

Improved version of Dash-5A, almost identical to commercial 250-C18, delivering full thermodynamic power of 236.4 kW (317 shp). Powers Bell OH-58A.

T63-A-701

Military counterpart of 250-C20 and B17, with T-O rating of 298 kW (400 shp).

T63-A-720

Military counterpart of 250-C20B and B17B, with hot-end improvements. T-O rating 313 kW (420 shp). Powers OH-58C and TH-57.

250-C18

Initial commercial version, based on T63-A-5A: T-O rating 236.4 kW (317 shp). Powers early versions of Bell JetRanger and TH-57A SeaRanger, Hughes (McDD) 500, Fairchild FH-1100, Agusta 206A and Kawasaki 369HS. Deliveries began December 1965. Licensed to MTU and Kawasaki.

250-B15

A direct conversion of the C18, the B15 was the original 236.4 kW (317 shp) turboprop version of the Model 250. The engine is essentially a C18 mounted in the inverted position, with compressor below the added propeller reduction gearbox and the twin jetpipes discharging obliquely downwards. The B15 was certificated in March 1969. Production engine, designated **B15G**, was fitted to the prototype SIAI-Marchetti SM.1019.

250-B17

Announced in 1972, the B17 is an uprated version of the B15 turboprop, corresponding to the C20 turboshaft. Rated at 298 kW (400 shp), it is fitted to the American Jet Industries Turbostar 402 (Cessna 402 conversion), the ASTA (previously GAF) Nomad and SIAI-Marchetti SM.1019E. In April 1974, Allison announced the **B17B**, operating at 17ºC higher turbine gas temperature and with hot-end improvements similar to those of the C20B. Rated at 313 kW (420 shp) for 5 min, the B17B entered production in September 1974 for Turbostar 402 conversions, Turbostar 414, ASTA Nomad N22 and 24, SIAI-Marchetti SM.1019E and various agricultural aircraft. **B17C** introduced improved gearbox, T-O rating remaining as B17B. Produced for Nomad N22 and 24, SF.260TP and 600TP, Turbostar 402/414, Allison Bonanza, LoPresti Piper SwiftFury, AASI Jetcruzer prototype, Glasair III, Advanced Airship ANR, BN-2T Turbine Islander, Composite Eagle, AP.68TP Viator and Maule M-7 and MT-7. **B-17D** produced for SF.260TP, Aucán, Redigo, Fuji T-5, HTT-34 and Mentor 420. **B17F**, with new compressor as in C20R for increased T-O power of 335 kW (450 shp) to 26.7ºC for 5 min, introduced in 1985 for BN-2T, Fuji KM-2D, Vulcanair SF.600, HX-1, RediGO, O&N P210 Silver Eagle and Tradewinds Turbines Bonanza. Selected for Fuji T-7, Grob G140 and Groen Jet Hawk 4T.

250-C20

Incorporating numerous improvements to increase air flow, component efficiency and turbine temperature, the C20 is rated at 298 kW (400 shp). Dry weight is increased by only 8.6 kg (19 lb) compared with the C18. Certificated in February 1974, the C20 is fitted to the Bell 206B JetRanger II, Eurocopter BO 105A, Agusta-Bell 206B and 206B-1, Agusta A 109, Dornier Do 34 Kiebitz RPV (with MTU power transmission), Dornier Seamos demonstrator and Soloy conversion of the UH-12E.

250-C20B

Introduced 1974 and rated at 313 kW (420 shp) to 26.7ºC for 5 min. For Bell and Agusta-Bell 206B JetRanger III and 206L LongRanger, MD 500D, BO 105CBS and E, Kitty Hawk, FH-1100 and UH-12E, RFB Fantrainer 400 and Bell 47G conversions. `Dramatic improvement' in first-stage turbine nozzle life resulting from 1995 introduction of Mar-M 247 material.

250-C20F

Rated at 313 kW (420 shp). For AS 355 Ecureuil 2/TwinStar.

250-C20J: Rated at 313 kW (420 shp). For Bell and Agusta-Bell 206B JetRanger III.

250-C20R: Derivative of C20B with new axial/centrifugal compressor. Rated at 335 kW (450 shp) to 26.7ºC for 5 min. **C20R/1** with redundant overspeed system for twin-engine applications certificated September 1986 for A 109A Mk II and selected for PZL SW-4, Kamov Ka-226 and Mil Mi-34A. **C20R/2** for single-engine helicopters certificated early 1987 for MD 500ER, JetRanger III, LongRanger and MD 520N.

250-C20R+

Compressor has new shroud, scroll and diffuser, and power-turbine wheels and nozzles, for improved hot/high performance. Entered service summer 1999, and these features are now standard on all C20R engines.

250-C20S

Turboprop for Soloy Turbine Pac conversions of Cessna 185, 206 and 207. Also powers APEC Nexus. Rated at 313 kW (420 shp) to 26.7ºC for 5 min.

250-C20W

Generally similar to C20B, with same ratings. Powers Enstrom 480, Schweizer 330SP and 333, and Northrop Grumman RQ-8A Fire Scout VTUAV.

250-C28

Representing Series III of the Model 250 evolutionary process, the C28 is a near-total redesign. The axial multistage compressor has been eliminated. Instead, a single-stage front entry centrifugal impeller is used, operating at a high pressure ratio and handling a considerably increased air flow. Despite the increase in power, the sound pressure level of the bare engine has been reduced (at 500 ft, from 68 to 63 dB). Emissions are approximately halved by the completely new premix-swirl combustion chamber, which also improves fuel efficiency and water ingestion capability. The compressor bleed is eliminated, and exhaust leaves through a single low-velocity stack which also has a minimal infra-red signature. The main gearbox has new gears with increased helix and decreased pressure angles. Flight cleared prototype engines were first available in March 1975. Certificated at T-O rating of 373 kW (500 shp) for 30 minutes in December 1977.

250-C28B: With particle separator; 2.5 minute rating of 410 kW (550 shp). Powers Bell LongRanger I.
250-C28C: Improved model with plain inlet; 2.5 minute rating of 410 kW (550 shp). Powers BO 105 LS.
250-C30: Advanced single-stage compressor and dual ignition. Initial rating 485 kW (650 shp), with a 2.5 minute rating of 522 kW (700 shp). Certification completed March 1978. Produced for S-76 and MD 530G.

250-C30G

Produced for Heli-Air Bell 222; C30G2 for Bell 230.

250-C30L

With digital control, produced for Bell 406CS.

250-C30M

Produced for AS 350G.

250-C30P

Produced for LongRanger III and IV.

250-C30R

With digital control, produced for AHIP OH-58D. **C30R/1** increased air flow for uprated hot-day power for OH-58D; certificated early 1994. **C30R/2** discontinued. **C30R/3** derived from C47 with new high-airflow compressor, low-smoke combustion liner, improved turbine nozzles and FADEC. Certificated June 1997 for Kiowa Warrior. **C30R/3M** has FADEC and bleed-valve changes, certificated in 2001 for MDHI AH/MH-6.

T703-AD-700

Military turboshaft engine corresponding to 250-C30R, for Bell OH-58D.

250-C30S

Produced for S-76A.

250-C40

Derived from C30R, with uprated gearbox and FADEC; T-O 584 kW, 783 shp; max continuous 521 kW, 699 shp; for Bell 430. In 1996, C40 and C47 models were being improved by minimising leakages, introducing an impingement-cooled first-stage nozzle, low-smoke combustor liner, enhanced filtration of new turbine oils, and using FADEC for auto-start cycle with temperature limiting and speed/temperature/torque limiting.

250-C47

C40 variant for Bell 407 and Boeing MD600N. T-O 590 kW, 791 shp; max continuous 525 kW (704 shp).

Deliveries of Model 250 engines began in 1958, and by mid-2001 the total exceeded 28,500. These had then logged about 150 million flight hours, of which 90 million had been recorded by the -C20 family. More than 17,000 Model 250 engines are in current operation, powering approximately half the world's turbine-engined light helicopters. These engines are supported by a global network of 27 independently owned Authorised

Maintenance Centers, with an additional 30 AMC branch locations. Each provides total support including unit exchange and Power by the Hour ® programmes. Rolls-Royce has temporarily shelved plans for an all-new successor engine.

Type

Free-turbine turboshaft or turboprop.

Compressor

Early versions, six axial stages and one centrifugal; axial rotors are blisks in 17-4PH steel, B17F and C-20R, four axial and one centrifugal. C28, C30, C40, C47, centrifugal only. Speed (B17/C20) 50,970 rpm, (C30/40/47) 51,000 rpm. Mass flow: all early versions 1.36 kg (3.0 lb)/s, B17C/D 1.62 kg (3.58 lb)/s, B17F 1.73 kg (3.82 lb)/s, C20B/F/J 1.56 kg (3.45 lb)/s, C20R 1.73 kg (3.82 lb)/s, C20W 1.56 kg (3.45 lb)/s, C22 1.814 kg (4.0 lb)/s, C28 2.00 kg (4.42 lb)/s, C30/M/P/S 2.54 kg (5.6 lb)/s, C40B/C47B/M 2.77 kg (6.1 lb)/s. Pressure ratio: all early versions 6.2, B17C/D 7.2, B17F 7.9, C20B/F/J/W 7.1, C20R 7.9, C22/C28 8.4, C30/P/S 8.6, C40B/C47B/M 9.2.

Compressor Turbine

Two-stage, integrally cast blisks. Max rpm: B17B, C20B, 50,970; C28, 51,005; C30, 50,000; C40B, C47B, 51,000.

Power Turbine

Two-stage, integrally cast blisks. TGT (317 shp) 750ºC; B17B, C20B, 810ºC; C28, 801ºC; C30, 771ºC. Max rpm: B17B, C20B, 33,290; C28, C30, 33,420; C40B, 30,908; C47B, 32,183; C47M, 30,650.

Jetpipe

After passing through the turbines, the gas enters a hood in the centre of the engine from where it is discharged through twin stacks pointing diagonally upwards in most turboshaft versions and downwards in turboprops. The C28, C30, C40 and C47 have a single low-velocity stack of rectangular cross-section.

Output

A magnesium casting which forms primary structure of engine and contains all power and accessory gear trains; torque sensor, oil pumps and engine main bearings. Compressor and combustor/turbine assemblies bolted to front and rear faces respectively. One spur gear train engages pinion driven by power turbine shaft and transmits output power to horizontal shaft on centreline of engine below (in turboprops and optionally on turboshaft models, above) compressor turbine output shaft, accessible on both front and rear faces of gearcase. Rated shp available at either front or rear spline, or any combination totalling rated power. Second spur gear train engages on gas generator turbine shaft and provides drive for engine accessory pads. Turboshaft version has usual output speed of 6,016 rpm (C40B, 9,598; C47B, 6,317). Turboprop has additional reduction gear to propeller shaft at top front of engine; B17, 2,013 rpm.

Fuel Specifications

Most engines, MIL-5624, JP-4, JP-5, ASTM-1655, Type A, A-1, B.

Oil Specifications

MIL-L-7808, MIL-L-23699.

Dimensions

Length:	

T63-A-5A	1,030 mm (40.5 in)
B17C/D/F (including fuel burner)	1,143 mm (45.0 in)
C18	1,036 mm (40.8 in)
C20B/F/J/W	985 mm (38.8 in)
C20R	1,036 mm (40.8 in)
C28	1,021 mm (40.2 in)
C30/M/P/S, C47B/M	1,041 mm (40.988 in)
C40B	1,040 mm (40.964 in)
Width:	
T63, B17C/D, C18, C20B/F/J/W	483 mm (19.0 in)
B17F	493 mm (19.4 in)
C20R	528 mm (20.8 in)
C28, C30/M/P/S, C40B, C47B/M	555 mm (21.86 in)
Height:	
T63, B17C/D, C18	572 mm (22.5 in)
B17F, C20W, C22	574 mm (22.6 in)
C20B/F/J/R	589 mm (23.2 in)
C28, C30/M/P/S, C40B, C47B/M	638 mm (25.13 in)

Weight, Dry

T63-A-5A	63.0 kg (139 lb)
B17B/C	88.4 kg (195 lb)
B17D	89.8 kg (198 lb)
B17F	93.0 kg (205 lb)
C20B/J	71.5 kg (158 lb)
C20F	73.0 kg (161 lb)
C20R	78.5 kg (173 lb)
C20W	73.9 kg (163 lb)
C28	105.7 kg (233 lb)
C30	113.8 kg (251 lb)
C30M	113.4 kg (250 lb)
C30P	111.1 kg (245 lb)
C30S	113.9 kg (251 lb)
C40B	127.0 kg (280 lb)
C47B	124.3 kg (274 lb)
C47M	122.5 kg (270 lb)

Performance Ratings

(S/L, ISA)

T63-A-5A:	
T-O flat rated at	188.3 kW (252 shp)
Cruise 75%	151 kW (203 shp)
T-O	236.4 kW (317 shp)
B17C/D:	
T-O, max cont	313 kW (420 shp)
Normal cruise	275 kW (369 shp)
75%	206 kW (277 shp)
B17F:	
T-O	335 kW (450 shp)
Normal cruise	283 kW (380 shp)
75%	234 kW (314 shp)
C18:	
T-O	23.64 kW (317 hp)
C20B/F/J/W:	
T-O, 30 min, max cont	313 kW (420 shp)
Normal cruise	276 kW (370 shp)
75%	207 kW (278 shp)
C20R:	
T-O, 30 min, max cont	335 kW (450 shp)
Normal cruise	283 kW (380 shp)
75%	236 kW (317 shp)
C28/B/C:	
T-O	372.3 kW (500 shp)
Normal cruise	368.4 kW (494 shp)
75%	283.4 kW (380 shp)
C30/C30S:	
2.5 min OEI	522 kW (700 shp)
30 min OEI, T-O 5 min	484.7 kW (650 shp)
75%	311.7 kW (418 shp)
C30M/P:	
T-O 5 min	484.7 kW (650 shp)
Max cont, 75%	335.6 kW (450 shp)
C40B:	
30 s OEI	611.5 kW (820 shp)
2 min OEI	574.2 kW (770 shp)
Cont OEI, T-O 5 min	533.2 kW (715 shp)
Max cont	457 kW (613 shp)
75%	343 kW (460 shp)

C47B, C47M:	
T-O 5 min	484.7 kW (650 shp)
Max cont	447.4 kW (600 shp)
75%	335.6 kW (450 shp)

Specific Fuel Consumption

all at 75% cruise:	
T63-A-5A	128.7 µg/J (0.762 lb/h/shp)
B17C/D	120.8 µg/J (0.715 lb/h/shp)
B17F	113.7 µg/J (0.673 lb/h/shp)
C20B/F/J/W	119.8 µg/J (0.709 lb/h/shp)
C20R	112.5 µg/J (0.666 lb/h/shp)
C28	112.0 µg/J (0.664 lb/h/shp)
C30/S	112.35 µg/J (0.665 lb/h/shp)
C30M/P	109.3 µg/J (0.647 lb/h/shp)
C40B	105.6 µg/J (0.625 lb/h/shp)
C47B	108.47 µg/J (0.642 lb/h/shp)
C47M	108.6 µg/J (0.643 lb/h/shp)

Contract Price

September 1997, US$7,897,878 for 27 C30R/3; US$188,920,755 for 602 C30R/3 (multiyear award); February 2000, three C20W (a cheaper engine), about US$500,000.

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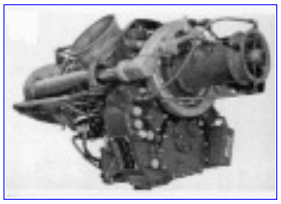
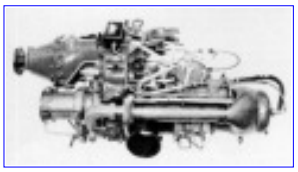
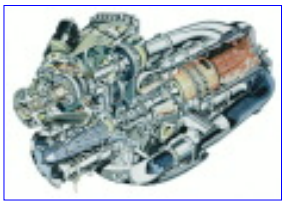


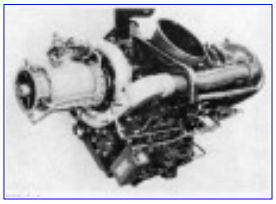
T63-A-5A



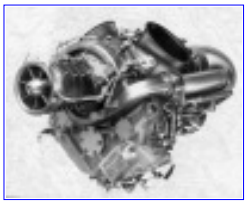
250-B17C



Cutaway drawing of 250-B17F (1999)

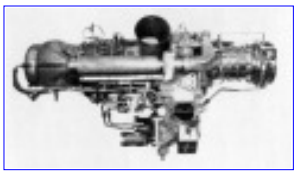


250-C18

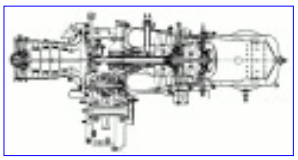


250-C20

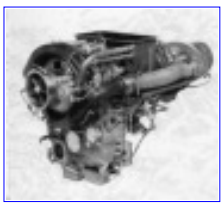
Cutaway drawing of 250-C20B/J (1999)



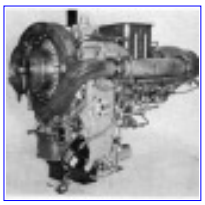
250-C20R



Longitudinal section through 250 C20R (1999)

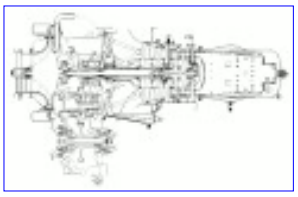


250-C28



250-C30

250-C40



Longitudinal section through 250-C40B (1999)

In September 1996, Ron Bower made a second flight round the world in a C40B-engined [Bell 430]

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AERO-ENGINES - TURBOSHAFT, UNITED STATES OF AMERICA

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE ALLISON

ROLLS-ROYCE AE 1107

US military designation: T406-AD-400

The T406 turboshaft, developed for the tilt-rotor Bell/Boeing V-22 Osprey, was derived from the T56/501. It is the basis for the AE 1107 turboshaft, AE 2100 turboprop and AE 3007 turbofan. The last two are described separately.

The T406-AD-400 is a free turbine, front drive 4,588 kW (6,150 shp) turboshaft incorporating high-efficiency components and reduced-maintenance features required for operation in the V-22 Osprey. It features six rows of variable compressor stators, dual FADEC fuel controls, a self-contained lubrication system capable of sustained engine operation in the vertical position, and modular construction. The T406 completed its flight rating tests in 1995 and 21 engines were delivered for the V-22 Full-Scale Development Flight Test Program, followed by a further 12 for the Engineering and Manufacturing Development Flight Test Program in 1997.

The T406-AE-400 entered low-rate production in June 1998, and rate of delivery increased in 2000 with the first production orders for the Marine Corps MV-22B and USAF CV-22B. In 2000 the programme was put on hold, but it was relaunched in November 2001 with additional funding to support design changes, which are confined almost entirely to the aircraft rather than the engine. Later the US Navy is expected to order the HV-22B CSAR (combat search and rescue) version. In June 2000, a further market opened up with the award by DARPA (US Defense Advanced Research Projects Agency) of a US$500,000 contract to Bell Helicopter Textron for Phase 1 investigation of the Quad Tiltrotor. This would be a multirole transport with payload/range similar to that of the C-130-30 (Stretched Hercules), but with hovering and VTOL capability. It would have tandem wings, each with a V-22 engine/rotor system. There are obvious civil applications.

The T406 is applicable to other helicopter and tilt-rotor aircraft as the **AE 1107**. This engine was FAA

certificated in 1998.

Type

Axial flow turboshaft.

Compressor

Fourteen-stage axial flow, with variable inlet guide vanes and first five stator rows. Pressure ratio 16.7. Mass flow 16.1 kg (35.5 lb)/s.

Combustion Chamber

Annular effusion-cooled, with 16 airblast-type fuel nozzles providing smoke-free operation. Dual capacitor discharge ignition.

Compressor Turbine

Gas generator turbine has two axial stages with air-cooled single-crystal blading; both stages overhung to the rear of the gas generator thrust bearing.

Power Turbine

Two axial stages on a straddle-mounted shaft which runs the entire length of the engine. Film-damped bearings eliminate the need for a centre bearing.

Output

Power turbine forward shaft drives a torquemeter assembly which is directly coupled to the V-22 rotor gearbox. The torque tube housing serves as the front engine mount.

Accessories

An engine accessory gearcase is mounted beneath the air inlet housing. It provides for engine starter, generator, oil pump and fuel pump metering unit drives.

Control System

Full-authority digital.

Fuel Specification

MIL-T-5624; grades JP-4, JP-5 and MIL-T-83188; JP-8.

Oil System

Self-contained, featuring positive scavenging sumps, 3 Ám filtration, and a bottom-mounted, all-attitude oil reservoir with service scuppers on each side of the engine.

Oil Specification

MIL-L-7808 or MIL-L-23699.

Dimensions

Length overall, without gearbox	1,958 mm (77.08 in)
Length from inlet flange	1,521 mm (59.88 in)
Width	671 mm (26.40 in)
Height	864 mm (34.00 in)

Weight, Dry

	440.4 kg (971 lb)

Performance Ratings

Max power (S/L)	4,586 kW (6,150 shp) to 43~C
Max continuous power (static 1,219 m; 4,000 ft)	3,253 kW (4,362 shp) to 25~C

Specific Fuel Consumption

Max cont power, as above	72.0 Ág/J (0.426 lb/h/shp)

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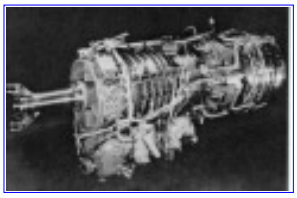
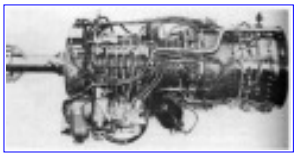


T406-AD-400



T406-AD-400 development engine

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AERO-ENGINES - TURBOPROP, UNITED STATES OF AMERICA

Date Posted: 11 January 2002

Jane's Aero-Engines 11

ROLLS-ROYCE CORPORATION

ROLLS-ROYCE 501

Originally known as the Allison 501, this turboprop is the commercial derivative of the T56. It was designed in 1953-55 to power the Lockheed L-188 Electra, matched initially with an Aeroproducts (later also a Hamilton Standard) reversible propeller. Allison claimed that 75 per cent of the parts of the 501 were identical to those of the T56. In particular, there is a high degree of commonality in the power sections and also in the propeller speed-reducing gearboxes, but an immediately obvious change is that in the 501 the gearbox is mounted below the axis of the engine, not above. This was so that in the low-slung engines of the Electra the air inlet could be above the propeller. Similar engines power the P-3 Orion family derived from the Electra. As these are military engines they retain the T56 designation, and are described in that entry.

The initial preproduction Model 501 was awarded a CAA (later FAA) Type Certificate on 20 May 1955. The following are major versions:

501-D13, D13A

T-O rating 2,580 kW (3,460 shp) + 3.33 kN (726 lb st) = 2,796 ekW (3,750 ehp). Powers Electra, which entered service in January 1959.

501-D13D

T-O rating as D13A. Powers CV-580.

501-D22A

T-O rating 3,257 kW (4,368 shp), 3,490 ekW (4,680 ehp). Commercial version of T56-A-15, with high propeller gearbox. Powers L-100 (Model 382) commercial Hercules.

501-D22C

T-O rating 3,257 kW (4,368 shp), 3,663 ekW (4,912 ehp). Powers Aeromaritime Super Guppy.

501-D22G

T-O rating 3,430 kW (4,600 shp), unstated ehp. Powers Super-580 and KFC CV-5800.

501-D39

Planned commercial version of T56-A-427 (which see).

501-M62

Military designation T701-AD-700; turboshaft engine for XCH-62 heavy-lift helicopter. Upgraded variable-stator power section. T-O rating 6,025 kW (8,079 shp).

501-M78

Experimental engine for NASA PTA (Propfan Test Assessment) programme. Flown on left wing of GII driving single-rotation propfan.

501-M80

Military designation **T406**, described separately. Rolls-Royce Corporation also markets versions of the Model 501 for marine propulsion and industrial purposes.

Type

Single-shaft, constant-speed turboprop.

Intake

Except in D22A, the cowl inlet is above the spinner, the air passing down through a curved duct to reach the engine. The engine power section is substantially the same as that of the T56. Operating speed in all flight regimes is 13,820 rpm.

Compressor

The rotor is that of corresponding T56 versions, but the casing is made as four forged quadrant sections bolted permanently together.

Mounting

Seven-point suspension, with special provisions to avoid dynamic forced vibration at high airspeeds.

Output

Propeller reduction gear mounted below the axis of the engine, the gearbox itself being essentially the T56 gearbox inverted. Drive ratio 13.54 (spur gear 3.13, planetary 4.33).

Fuel Specification

Cleared to operate on JP-1.

Dimensions

Length:	
D13	3,688 mm (145.2 in)
D22A, D22C	3,708 mm (146.0 in)
Width	686 mm (27.0 in)
Height:	
D13	914 mm (36.0 in)
D22A	991 mm (39.0 in)

Weight, Dry

D13	794 kg (1,750 lb)
D22A	832 kg (1,834 lb)

Performance Rating

T-O, S/L: See model listing	
Normal:	
D13	2,307 kW (3,094 shp) + 3.12 kN (702 lb st)
D22A, D22C	3,028 kW (4,061 shp), 3,255 ekW (4,365 ehp)

Specific Fuel Consumption

T-O, S/L:	
D13	91.26 µg/J (0.540 lb/h/ehp)
D22A, D22C	84.67 µg/J (0.501 lb/h/ehp)

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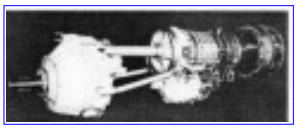
501-D13



501-M78

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AERO-ENGINES - TURBOJET, UNITED STATES OF AMERICA

Date Posted: 24 October 2001

Jane's Aero-Engines 11

TELEDYNE CONTINENTAL MOTORS

TURBINE ENGINES J69

This turbojet is a licensed version of the French Turbomeca Marboré II, further developed by the former Continental Motors and built in several versions. Most applications are for unmanned vehicles, but the most important version is the following:

J69-T-25A

This is the current version fitted to the USAF's primary jet pilot trainer, the Cessna T-37A. Several thousand engines of the basic T-25 (Model 352) family have been in USAF service in this aircraft since 1954, and they will not be replaced for many years, even though the turboprop-engined T-6A Texan II is entering full service from 2000.

Confirming the amount of T-37A flying yet to be done, in March 1999 TCM Turbine Engines was awarded a USAF contract for US$5,282,849 for 404 (best estimated quantity) of J69 compressor inducer rotor assemblies. This is machined from a light-alloy forging and guides air into the centrifugal compressor to which these curved vanes are attached. Later in 1999 the company received a US$43 million award to provide support until the J69 is finally retired from service. In 2000 it was estimated that the T-37A will continue to train pilots until 2012 with the USAF and possibly until 2030 with other nations.

Type

Single-shaft turbojet.

Compressor

Single-stage centrifugal. Mass flow 9.07 kg (20.0 lb)/s. Pressure ratio 4.1.

Combustion Chamber

Annular, rotary centrifugal fuel distributor.

Turbine

Single axial stage with replaceable blades.

Jetpipe

Fixed area, maximum gas temperature 663ºC.

Accessories

Fuel and oil pumps, electric starter/generator, tachogenerator and hydraulic pump, off front of shaft.

Control System

Hydromechanical.

Dimensions

Length	898.9 mm (35.39 in)
Diameter	566.9 mm (22.32 in)

Weight, Dry

	162.4 kg (358 lb)

Performance Ratings

(S/L, ISA)

T-O	4.559 kN (1,025 lb st) at 21,730 rpm
Normal	3.914 kN (880 lb st) at 20,700 rpm

Specific Fuel Consumption

T-O	32.29 mg/Ns (1.14 lb/h/lb st)
Normal	31.72 mg/Ns (1.12 lb/h/lb st)

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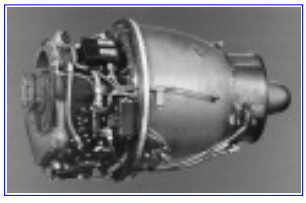

J69-T-25A

Cutaway drawing of J69-T-25A (2000)

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Bill Gunston OBE, FRAeS

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 23 October 2001

Jane's Aero-Engines 11

UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY PW4000

Pratt & Whitney's PW4000 engine family covers the thrust range from 222.4 kN (50,000 lb st) to 435.9 kN (98,000 lb st) to meet power requirements for current and future wide-body aircraft. The programme includes an exceptional team of participants: Techspace Aero (Belgium), FiatAvio (Italy), Norsk Jet Motors (Norway), Kawasaki and Mitsubishi Heavy Industries (Japan), Samsung (South Korea), Eldim (Netherlands), Singapore Technologies, and MTU Motoren- und Turbinen-Union (Germany).

The key feature of all PW4000 versions is that, compared with the JT9D family, there are roughly half as many parts. Pratt & Whitney considers that this should reduce maintenance costs by more than 25 per cent. Another important advance is that, at full power, the HP spool rotates 27 per cent faster, increasing the HP pressure ratio by 10 per cent. Other advances include monocrystal turbine rotor blades, aerodynamically enhanced aerofoils and FADEC control.

The engine family is capable of such a broad range of thrust levels through the utilisation of three fan diameters. The 94 in fan models utilise the same basic engine, with changes only to externals and airframe accessories, to power the Airbus A300 and A310 (designation beginning PW41), Boeing 747 and 767 (PW40), and the (Boeing MD-11 (PW44). The 100-in fan growth model powers the Airbus A330, and the 112-in fan model powers the Boeing 777. The advantages of commonality include reduced tooling requirements, common parts and lower spares/inventory requirements, as well as similar maintenance procedures.

94-in FAN

PW4000

The first engine began testing in April 1984 and quickly reached a thrust of 275 kN (61,800 lb st). Flight testing began in an A300B on 31 July 1985. Subsequent models have a four-digit designation beginning with 40 for Boeing, 41 for Airbus and 44 for Boeing aircraft designed by McDonnell Douglas, the last two digits denoting the T-O thrust in thousands of pounds.

PW4050

Original 2,377 mm (93.6 in) fan. T-O rating 222.4 kN (50,000 lb st) to 33.3ºC. FPR 1.63, BPR 5.0, OPR 26.3. Powers 767-200, 767-200ER, 767-300.

PW4052

T-O rating 232.1 kN (52,200 lb st) to 33.3ºC. FPR 1.65, BPR 5.0, OPR 27.5. Powers 767-200, 767-200ER, 767-300, -767-300ER.

PW4152

T-O rating 231.2 kN (52,000 lb st) to 42.2ºC. Powers A310-300. First version to enter service, with PanAm on 20 June 1987.

PW4056

T-O rating 252.3 kN (56,750 lb st) to 33.3ºC. FPR 1.71, BPR 4.9, OPR 30.0. Mass flow 773 kg (1,705 lb)/s. Powers 747-400 and -400F, 767-200, -200ER, -300, -300ER.

PW4156

T-O rating 249.1 kN (56,000 lb st) to 30ºC. Powers A300-600.

PW4158

T-O rating 258.0 kN (58,000 lb st) to 30ºC. FPR 1.73, BPR 4.8, OPR 30.6. Powers A300-600R.

PW4060

T-O rating 266.8 kN (60,000 lb st) to 33.3ºC. FPR 1.74, BPR 4.8, OPR 31.1. Powers 767-200, -200ER, -300F, -300ER.

PW4460

T-O rating 266.8 kN (60,000 lb st) to 30ºC. Powers MD-11.

PW4062

T-O rating 275.8 kN (62,000 lb st). FPR 1.76, BPR 4.8, OPR 32.0. Powers all 767 versions.

PW4462

PW4460 rerated at 275.8 kN (62,000 lb st) to 30ºC. Mass flow 816.5 kg (1,800 lb)/s. FPR 1.8, BPR 4.8, OPR 32.3. Powers MD-11.

The above engines remained in production at the end of 1999. At that time 2,033 engines has been shipped to 65 airlines, with an aggregate flight time of 39,114,372 hours, of which 11 million were 180-min ETOPS. Pratt claims ``99 per cent dispatch reliability, the best hot section and the best performance retention".

In 1998, and again in April 1999, the FAA issued ADs (Airworthiness Directives) applicable to all high-time engines with the 94-in fan. Following a number of single- and dual-engine surge events, first experienced in 1992 but by 1999 much less common, the AD requires specific on-wing and test-cell procedures to assess the stability of the HP compressor. Pratt is working on a long-term fix involving a redesign of the HPC stators and reprogramming of the FADEC. These changes were validated in flight tests in third quarter 1999. If approved, the modifications will be incorporated at scheduled overhauls.

Despite this effort, in July 2001 the FAA issued a further AD. This followed compressor surges on take-off on both engines of an A300 of Air Sudan. The directive required 94-in engines to be removed before exceeding specified cycle/time limits for modification to the HP compressor.

Type

Two-shaft turbofan.

Fan

Single stage. Titanium-alloy hub retains 38 titanium alloy blades with aft part-span shrouds. Diameter 2,377 mm (93.6 in). FPR and BPR, see model list. For temperatures, see small drawing of PW4052.

LP Compressor

Four stages with controlled-diffusion aerofoils, rotating with the fan.

HP Compressor

Eleven stages with first four vane rows variable. Clearance control achieved via rotor/case thermal matching. OPR, see model list.

Combustion Chamber

Annular, forged nickel-alloy rolled ring with double-pass cooling, fitted with 24 airblast anti-coking injectors. Segmented Floatwall burner liner introduced in 1993.

HP Turbine

Two stages with air-cooled blades cast as single-crystal (PWA1480) in first row and advanced single-crystal (PWA1484) in second row. Double-hub nickel-alloy rotor with active clearance control. Vane aerofoils thermal-barrier coated.

LP Turbine

Four stages with active clearance control.

Control System

Dual-channel FADEC.

Dimensions

Length	3,901 mm (153.6 in)
Fan case diameter	2,463 or 2,469 mm (96.98 or 97.20 in)

Weight, Dry

Typical	4,273 kg (9,420 lb)

Performance Ratings

See model listing

Specific Fuel Consumption

T-O, S/L:	
PW4050	9.858 mg/Ns (0.348 lb/h/lb st)
PW4052	9.943 mg/Ns (0.351 lb/h/lb st)
PW4056, PW4156	10.170 mg/Ns (0.359 lb/h/lb st)
PW4158, PW4060, PW4062	10,340 mg/Ns (0.365 lb/h/lb st)
PW4460	10.481 mg/Ns (0.370 lb/h/lb st)
PW4462	10.708 mg/Ns (0.378 lb/h/lb st)

Contract Price

Air China, PW4056 for three 747-400, about US$100 million; China Airlines, PW4056 for eight 747-400, about US$270 million; Delta, 24 installed PW4060, US$170 million.

100-in FAN

PW4168

Specifically designed from December 1991 for A330, with new fan of 2,535 mm (99.8 in) diameter. Loosely known as `100-inch' fan. Fifth stage added to LP compressor (core booster) and fifth stage added to LP turbine. Floatwall combustor, FADEC control and ``the industry's lightest weight and most advanced nacelle'' with load-sharing structure. Certificated August 1993 as PW4164/4168 at T-O ratings of 284.7 kN (64,000 lb st) and 305.2 kN (68,000 lb st), in each case to 30ºC. Awarded 90-min ETOPS in November 1994 (claimed as ``industry first''), followed by 120 min in February 1995 and 180 min in July 1995. In 2000 the PW4168A was offered with 4.5 per cent greater T-O thrust at ``hot and high'' airports. FPR 1.75, BPR 5.1, OPR 32.0.

PW4173

Same fan as PW4168 but advanced 3-D compressor aerodynamics and low-emissions combustor.

Interchangeable with PW4164/4168. T-O rating 324.7 kN (73,000 lb st). This engine was to have been certificated in mid-2000, with 180-min ETOPS available in early 2001. To provide ``10 per cent reduction in maintenance cost, 30 per cent reduction in emissions and 15 per cent longer time on wing''. FPR 1.84, BPR 4.8, OPR 35.4. In June 1999 Pratt claimed ``just over half'' of all A330 engine orders (disputed by competitors), adding ``nearly 50 of the 115 Pratt-powered A330s are already in service, and have accumulated twice the service experience of the competition''. In fact, the PW4173 secured only six firm sales (from two leasing companies) and ran into a severe problem with the hot bleed air (from the 12th stage, bleed from the 4168 being taken from the 9th), which was impinging upon the composite nacelle and causing rapid degradation. Curing the problem would have added 68 kg (150 lb), and the PW4173 was already on the A330 pylon limit. In February 2000 the PW4173 was shelved, and this entry will probably be removed at the next update.

By December 1999 Pratt & Whitney had shipped 162 of the so-called `100-in' engines. These had then flown 917,114 hours, and had established a ``near zero IFSD rate and 99.9 per cent dispatch reliability''. The 1-million hour mark was passed in May 2000. Development is in hand to incorporate core technology from the 112-in engine for thrust growth to 333.6 kN (75,000 lb st).

The following are differences compared with the basic 94-in engine:

Fan

Hub retains 34 titanium-alloy blades with aft part-span shrouds. Diameter 2,535 mm (99.8 in)

LP Compressor

Five stages.

LP Turbine

Five stages.

Dimensions

Length	4,143 mm (163.1 in)
Fan case diameter	2,718 mm (107.0 in)
Weight, Dry	5,307 kg (11,700 lb)

Performance Ratings

See model listing

Specific Fuel Consumption

Not stated.

Contract Price

In 1996, Korean, four PW4168, US$45 million, and engines for 10 optioned A330, US$237 million; January 1997, engines for 17 A330, US$380 million.

112-in FAN

PW4084

Major growth engine, with entirely new 2,845 mm (112 in) fan with hollow titanium blades without part-span clappers, six-stage LP compressor and seven-stage LP turbine. Mass flow 1,088.6 kg (2,400 lb)/s. First tested in August 1992, achieving 400.3 kN (90,000 lb st). A thrust of ``100 K'' (444.8 kN) was achieved in May 1993, and the engine first flew on a 747 testbed in November of that year. The 777 first flew in June 1994.

The PW4084 was certificated in April 1994 at 376.3 kN (84,600 lb st), cleared April 1995 to 386.0 kN (86,760 lb st), to 30ºC. FPR 1.70, BPR 6.4, OPR 34.2. Cleared to 180-minutes ETOPS in May 1995 (the first engine to be thus certificated prior to EIS). Entered service with United Airlines 7 June 1995 at initial rating of 342.5 kN (77,000 lb st). Powers 777-200. By July 1997, the PW4084 had flown over 280,000 hours in over 93,000 cycles with United, All Nippon and JAL, with dispatch reliability of 99.84 per cent. By July 2000 these figures had reached 1.6 million hours in over 500,000 cycles with dispatch reliability of 99.97 per cent.

PW4090

PW4084 with improved HP compressor aerodynamics; bowed stators create radial forces which reduce diffusion in the endwall regions, while contoured rotor-blade roots do the same at the roots, minimising flow separation. Mass flow 1,233.8 kg (2,720 lb)/s. The new HP spool has demonstrated a 2 per cent efficiency gain at S/L T-O. Other new features include redesigned fan exit guide vanes and improved HP and LP turbine cooling and materials, permitting increased TET. First tested July 1995. A series of 850 flight cycles was completed in December 1995 at TET exceeding 1,649ºC (3,000ºF), representing the maximum for both the PW4090 and the PW4098. Certificated on June 28 1996 at 408.3 kN (91,790 lb st) to 30ºC. FAA-certificated on 777-200 March 1997. FPR 1.74, BPR 6.3, OPR 38.3. First flight 3 August 1996, on 777. EIS March 1997. Powers 777-200IGW and 777-300.

By 1999 the only significant problem had been failure of the No 2 carbon seal, which prevents airflow into the large compartment between the N1 and N2 rotors at the front of the engine. It was decided to change this from a dry to a wet-face assembly running against an oil film. On 29 July 1998, on its last flight before this modification was incorporated, this seal failed during a take-off from London for San Francisco. The engine oil was lost, the 1.5 seal also failed, the N1 rotor jammed and a fire started, but the aircraft returned to London safely. The change to a wet-face seal appears to have afforded a complete solution.

PW4098

Based on PW4090 core with seven-stage LP compressor handling increased core air flow, and fan of 2,867.7 mm (112.9 in) diameter with improved aerodynamics to give FPR 1.8, BPR 5.8, OPR 42.8. Mass flow 1,292.8 kg (2,850 lb)/s. First complete engine run October 1996. The PW4098 was to have been certificated in March 1998 at 435.9 kN (98,000 lb st) to 30ºC for EIS in September 1998. Certification has been delayed by cracked compressor vanes (stators), in-flight compressor rubs and, according to P&W Vice-President Bob Leduc, by ``sudden fuel chops''. Certification date was moved to July 1998 and then to April 1999, and was then postponed again. To power 777-300.

PW40102

Development started December 1997, but delayed by problems with the PW4098. FPR 1.83, BPR 5.5, OPR 46.4. To be rated at 453.7 kN (102,000 lb st) to 25ºC. Intended for future 777 versions.

According to Pratt, the 112 in (fan diameter) engines ``have won more than 40 per cent of the 777 market, and 65 per cent of the 777-300 market''. Despite the severe problems with the PW4098, Pratt has, like GE, suggested that it should exclusively power the proposed 777-200X/300X, which will need engines in the over-100 k class. Boeing picked GE in July 1999. See PW40115 below.

By December 1999 total shipments of all versions of the 112-in engine had reached 226, all for the 777. These engines had then flown a total of 1,347,161 hours. Thus, at that time, total shipments of all PW4000 engines had reached 2,421.

The following are differences compared with the basic 94-in engine:

Fan

Totally redesigned containment ring retains 22 shroudless blades of graphite-epoxy composite, FPR and BPR, see model list.

LP Compressor

Six stages.

HP Compressor

Number of stages unaltered, but aerodynamically and structurally redesigned. OPR, see model list

LP Turbine

Completely redesigned, with seven stages

Dimensions

Length:	
PW4084, PW4090	4,868 mm (191.7 in)
PW4098, PW40102	4,945 mm (194.7 in)
Fan case diameter:	
PW4084, PW4090	3,001 mm (118.5 in)
PW4098, PW40102	3,035 mm (119.5 in)

Weight, Dry

PW4084	6,768 kg (14,920 lb)
PW4090	7,140 kg (15,740 lb)
PW4098	7,484 kg (16,500 lb)

Specific Fuel Consumption

T-O, S/L:	
PW4084	9.320 mg/Ns (0.329 lb/h/lb st)
Later versions	not stated

Contract Price

JAL, 10 installed PW4090 and two spare, US$160 million.

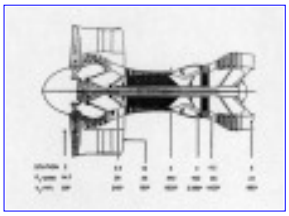
PW40115

In order to compete in the propulsion of the proposed 777-200X/300X Pratt & Whitney offered an engine with a T-O rating of 511.5 kN (115,000 lb st). This engine would have a fan with wide-chord hollow titanium blades and a diameter of 3,200 mm (126 in) and a completely new HP spool with eight highly loaded stages rotating at high rpm.

UPDATED

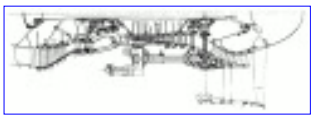
PW4000 (93.6 in fan)

Early production PW4152



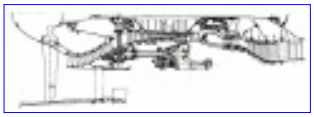
Pressures and temperatures in PW4052 (values much higher in more powerful versions)

Cutaway drawing of PW4052-4060 family (93.6 in fan, loosely called 94 in)



Longitudinal half-section through 94-in PW4000 (2000)

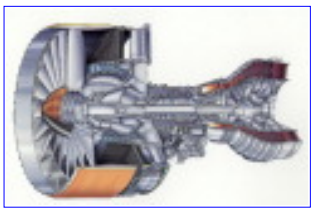
Cutaway drawing of PW4168 (99.8 in fan, loosely called 100 in) complete propulsion system



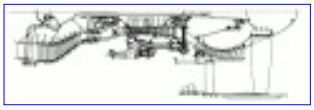
Longitudinal half-section through 100-in PW4000 (2000)

PW4090

PW4084 on endurance test



Cutaway drawing of PW4084-4098 (112 in fan)



Longitudinal half-section through 112-in PW4000 (2000)

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 18 September 2001

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For many years the US Air Force, like other US armed forces, has sponsored or managed many programmes intended to hasten the introduction of new technology. Most of the major advances in aerospace technology over the past 40 years have owed at least something to military funding, and this has certainly been the case with engines for fighters, heavy airlifters and helicopters.

After prolonged discussions with industry and research centres, the Air Force in July 2001 launched a major programme notable for looking exceptionally far ahead. It aims to pay for work which US engine manufacturers would probably consider a costly high risk. This programme will involve US industry, but under present plans most of the expenditure will be in-house at the Air Force Research Laboratory (AFRL) at the above address. The following appeared in *Jane's Defence Weekly* on 1 August 2001, under the byline of Staff Reporter Michael Sirak:

`The US Air Force (USAF) plans to launch a new aircraft engine development programme in 2005 with the aim of developing intelligent, versatile and durable propulsion systems for its next generation

of manned and unmanned platforms.

`USAF officials say the key driver of the two-phased effort, known as the Versatile Affordable Advanced Turbine Engines (VAATE) programme, is to reduce the development, production and maintenance costs of military turbofan and turbojet engines, perhaps tenfold by 2017, while achieving higher thrust-to-weight-ratio performance and lower thrust-specific fuel consumption.

`Tim Lewis, of the AFRL's Propulsion Directorate, said the laboratory will have to explore "radically different" innovative concepts to achieve this. Such concepts include 'pulse detonation', which relies on the physics of combustion within the engine and not the engine's rotating machinery to achieve thrust.

`The VAATE programme builds on the AFRL's existing Integrated High Performance Turbine Engine Technology Programme, which has brought government and industry together to double aircraft and missile engine performance since the mid 1980s. The programme is entering its final Phase III period, which will conclude in 2005. Phase III advances will be incorporated into programmes like the Joint Strike Fighter, according to the laboratory.

`AFRL officials are in the midst of the VAATE concept definition phase, which runs through 2003/04 to identify engine configurations in combination with notional future platforms, AFRL briefing charts show. Early demonstrations are planned during this phase with the goal of achieving fourfold cost savings.

`VAATE phase I will run from 2005 to 2010, with the goal of creating a "versatile core, robust intelligent engine" at sixfold cost savings. Phase II will last through 2017 and seek to develop an "innovative engine architecture, compact, cloakable [infra-red/radar cross section] design, maintenance-free core/damage adaptable engine" at tenfold cost savings.

`Lewis said the phased approach manages risk and will produce technology demonstrator engines every several years, which could be transitioned to commercial or military use.

`The programme is structured into three focus areas. Underpinning the programme will be the 'versatile core' area to leverage commercial engine technologies and identify common tooling and manufacturing processes to provide high-performance military engines based on a commercial core engine.

`The 'ultra-intelligent engine' area will apply advances in machine intelligence and learning technologies to create engines that can adapt to overcome battle damage/foreign-object damage or optimise their performance for a variety of missions or even for stages within a mission.

`The AFRL also envisions proactive capabilities to monitor the engine's health to maximise its availability, said Lewis.

`The 'durability and readiness' area will seek to ensure that the engine has an inherent robustness to accomplish more than just its defined mission.

`The AFRL will concentrate its in-house research on the 'high-impact' technologies needed to achieve substantial cost savings, said Lewis. Researchers will also examine if trade-offs can be made to unmanned air vehicle propulsion systems to achieve additional cost savings over engines designed for manned platforms, he added.'

UPDATED

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AERO-ENGINES - TURBOPROP/ - TURBOSHAFT, UNITED STATES OF AMERICA

Date Posted: 18 September 2001

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LIGHT HELICOPTER TURBINE ENGINE COMPANY

LHTEC T800

This fully metric, high-power density shaft-turbine engine, designated T800-LHT-800 (T800) stems from programmes that started before 1983. The T800's core was based on AlliedSignal's F109 turbofan engine and the power turbine and accessories were based on Allison's Advanced Technology Demonstrator Engine (ATDE) developed for the Army. The original requirement was to power the LHX helicopter, now called the Comanche RAH-66, for the US Army. In 1984, AlliedSignal (Garrett) and Allison joined as a 50/50 partnership called Light Helicopter Turbine Engine Company (LHTEC) to develop the T800. After fierce competition, the T800 was selected by the US Army in October 1988 as the main power plant for the Comanche. Today the partners have become Rolls-Royce Corporation and Honeywell.

T800-LHT-800

The T800 is designed for unprecedented reliability, maintainability and supportability, tilt-rotor compatibility and low specific fuel consumption. Its basic design philosophy includes module/LRU removal and replacement times of less than 15 minutes using six basic hand tools, and on-condition maintenance. In 1993, the T800 became the first engine concurrently to satisfy both FAA certification (CTS800-0) and Army Qualification requirements. This engine was the original power unit of the RAH-66 Comanche EMD helicopters.

A second application is the Global Huey 800 (re-engined UH-1H). By 1995, the T800 had flown approximately 1,650 hours in five US Border Patrol UH-1H helicopters along the Arizona and Texas

borders. In 1993, the Global Helicopter UH-1H/T800 demonstrator established a world point-to-point record, flying 1,714 n miles (3,175 km; 1,973 miles) in 13 hours and 6 minutes. This demonstrator logged over 550 hours performing flight demonstrations worldwide.

T800-LHT-801

In January 1993, the US Army awarded LHTEC a growth contract to increase power of the T800 engine by 17 per cent (see data). This growth engine, designated T800-LHT-801 (T801) had by May 1999 accumulated over 5,000 hours of development testing. By spring 2000 this total had doubled, and certification was due at mid-2000. Powers RAH-66 Comanche. In June 2000, the US Army was at last able to proceed with the Comanche programme, following a protracted review by the Pentagon Acquisition Board. It at once ordered five more helicopters for EMD and eight for field evaluation by USA units. The USA requirement is for 1,213, costing some US$34 billion.

By May 1999, over 27,200 T800-family engine hours had been recorded, including over 5,550 flight hours in seven types of helicopter.

Civil Turboshaft versions:

CTS800-0

: First commercial version, FAA-certificated in 1993, based on the military T800. First flown in 1988 in Agusta A129, and subsequently with minor changes in a Lynx, HH-65, UH-1H, ALH and Panther.

CTS800-1G

Max rating 1,312 kW (1,760 shp). First flight 1992. Fitted to BP800 and Huey 800 (UH-1 upgrades).

CTS800-2

T-O rating 1,024 kW (1,373 shp). Civil production engine with a high-speed output. Selected for Agusta A129 International (A129I) and A129 Scorpion. By late 1999, the A129I demonstrator had logged over 270 hours, in the course of which gun and rocket trials were completed in the Turkish ATAK competition.

CTS800-4

T-O rating 1,016 kW (1,362 shp). Civil production engine with integral speed-reduction gearbox. Other features include a commercial digital control (ARINC 429 databus) and elimination of specific military features. A variant designated **CTS800-4N** has been selected as the powerplant of the Agusta-Westland Super Lynx. The Super Lynx 200 first flew on 23 September 1991, and this was followed by Agusta-Westland's demonstrator ZT800, converted to the CTS800-4N in August 1999. The Mk 200 has now been superseded by the Super Lynx 300, with upgraded avionics. This is now in low-rate production, against an order for six from the Malaysian Navy. The South African Air Force has a requirement for four, and Oman and Thailand are both discussing the purchase of shipboard versions. The first Super Lynx 300 began flight testing on 12 June 2001. A further application of the CTS800-4N was the Indian ALH, but an initial order for 30 engines for this helicopter was vetoed by the US State Department, because of Indian nuclear testing, leaving the Turbomeca TM 333 2B as exclusive powerplant. The CTS800-4N is now fully certificated for use in all environments, with a design TBO of 6,000 hours.

CTS800-50

T-O rating 1,186 kW (1,591 shp). Civil production growth engine with high-speed output based on military T800-LHT-801, in development.

CTS800-54

T-O rating as CTS800-50 but derated to 1,009 kW (1,353 shp). Fitted with reduction gearbox and civil control for re-engined UH-1H. FAA certification is scheduled for mid-2000.

The following description refers basically to the CTS800 turboshaft models:

Type

Free-turbine turboshaft engine.

Intake

Annular ring surrounding optional front power take-off. Integral Particle Separator (IPS) with discharge duct able to extract up to 97 per cent of particles.

Compressor

Two centrifugal stages, each with one-piece impeller in titanium alloy. Mass flow (T800-LHT-801) 4.43 kg (9.8 lb)/s. (CTS800-4) 3.54 kg/s (7.8 lb)/s, (CTS800-50) 4.01 kg/s (8.84 lb)/s. Pressure ratio (all announced versions) 15.5.

Combustion Chamber

Annular reverse-flow, with 12 fuel nozzles around rear rim.

Compressor Turbine

Two stages with cooled 1st-stage blades. EGT (CTS800-4) 876ºC, (CTS800-50) 882ºC.

Power Turbine

Two stages, widely separated from compressor turbine by rear bearing assembly. Rotor blades with tip shrouds. EGT (CTS800-4) 614ºC, (CTS800-50) 616ºC.

Output

Drive at 23,000 rpm to front, with speed sensors and phase-shift torque-measuring system. (Speed-reduction gearbox gives output at 6,000 to 6,600 rpm).

Accessories

Grouped above engine with drive from HP shaft. Self-contained 28 V electrical system with permanent-magnet alternator. Suction-lift fuel pump.

Starting

Electric or pneumatic starter capable (MA4080 interface) and high-energy ignition system.

Control System

FADEC with ARINC 429 interface.

Fuel Specification

JP-4, JP-5 (MIL-T-5624 and -83133); JP-8. Emergency, DF-A, DF-1, DF-2.

Oil System

Self-contained with integral tank of 4.1 litres (1.08 US gallons, 0.9 Imp gallon) with sight glass, air/oil radiator and oil filter impending bypass indicator; magnetic chip detector.

Oil Specification

MIL-L-7808, MIL-L-23699.

Dimensions

Length overall:	
T800-LHT-801	856.0 mm (33.7 in)
CTS800-2, -50, -54	856.0 mm (33.7 in)
CTS800-4	1,047.2 mm (41.23 in)
Width:	
T800-LHT-801	550 mm (21.66 in)
CTS800-2, -4, -50, -54	561.6 mm (22.11 in)
Height:	
T800-LHT-801	662 mm (26.07 in)
CTS800-2, -50, -54	690.9 mm (27.2 in)
CTS800-4	699.3 mm (27.53 in)

Weight, Dry

T800-LHT-801, CTS800-2	149.7 kg (330 lb)
CTS800-4	173.7 kg (383 lb)

Performance Ratings

(S/L)

T800-LHT-800:	
Contingency	1,043 kW (1,399 shp)
Max (normal use)	995 kW (1,334 shp)
Intermediate (30 min)	824 kW (1,239 shp)
Continuous	774 kW (1,038 shp)
T800-LHT-801:	
Contingency	1,221 kW (1,638 shp)
Max (normal use)	1,166 kW (1,563 shp)
Intermediate (30 min)	1,089 kW (1,460 shp)
Continuous	918 kW (1,231 shp)
CTS800-2:	
30 s OEI	1,215 kW (1,630 shp)

2 min OEI	1,130 kW (1,516 shp)
Continuous OEI and T-O	1,024 kW (1,373 shp)
Continuous	954 kW (1,279 shp)
CTS800-4:	
30 s OEI	1,208 kW (1,620 shp)
2 min OEI	1,123 kW (1,506 shp)
Continuous OEI	1,016 kW (1,362 shp)
T-O	1,016 kW (1,362 shp)
Continuous	946 kW (1,268 shp)
CTS800-50:	
30 s OEI	1,421 kW (1,906 shp)
2 min OEI	1,286 kW (1,724 shp)
Continuous OEI and T-O	1,186 kW (1,591 shp)
Continuous	1,094 kW (1,467 shp)

Specific Fuel Consumption

(T-O)

T800-800	76.02 µg/J (0.450 lb/h/shp)
CTS800-2	77.9 µg/J (0.461 lb/h/shp)
CTS800-4	78.5 µg/J (0.465 lb/h/shp)
CTS800-50	76.85 µg/J (0.455 lb/h/shp)

Turboprop versions:
LHTEC has discontinued the turboprop versions described previously. However, in January 1997, LHTEC signed a contract with Federal Express to power the Ayres Loadmaster LM200 aircraft with the LHTEC CTP800-4T engine.

CTP800-4T Twin Power

T-O rating 2,013 kW (2,700 shp). The CTP800 engine is a Propulsion System Unit (PSU) consisting of two modified CTS800 engine power sections running into a GKN Westland combining gearbox driving a single 3.96 m (13 ft) Hamilton Sundstrand 568F-11 propeller with six `scimitar' blades. In February 2000, LHTEC President Mike Bush said ``The PSU is on time and delivering the performance we expected. We have completed 300 hours of testing and have had no problems". The first PSU was delivered to Ayres in June 2000. The Loadmaster installation was scheduled to receive FAA certification in mid-2000. The initial FedEx order for 50 PSUs was later augmented by exercising an option for a further 25 and a further 175 are on option, but the entire Loadmaster programme has been badly delayed by the bankruptcy of Let of the Czech Republic, which was sharing airframe manufacture.

Dimensions

Length overall (to propeller flange)	1,752.6 mm (69.003 in)
Width	1,251.7 mm (49.276 in)
Height	739.0 mm (28.82 in)

Weight, Dry

	517.1 kg (1,140 lb)

Performance Ratings

(S/L)

APR (each power section)	1,007 kW (1,350 shp)
T-O	2,013 kW (2,700 shp)
Max Climb	1,938 kW (2,599 shp)
Max Cruise	1,676 kW (2,247 shp)

Specific Fuel Consumption

(T-O)

	78.5 µg/J (0.465 lb/h/shp)

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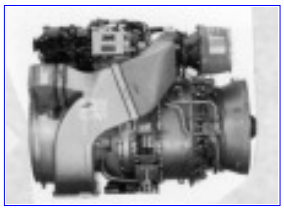
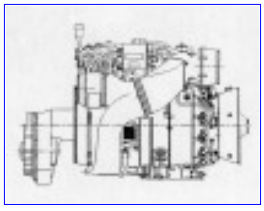
Cutaway drawing of basic T800



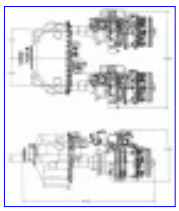
T800-LHT-800 (left side)

T800-LHT-800 (right side)

CTS800-4N
(1999)



Side elevation of CTS800-4N commercial version with integral gearbox



Top and side elevations of the CTP800-4T Twin Power turboprop engine with combining gearbox
(1999)

The CTP800-4T flight-test unit on static test
(2001)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 18 September 2001

Jane's Aero-Engines 11

GE AIRCRAFT ENGINES

GENERAL ELECTRIC TF34 and CF34

In April 1968, GE was awarded a Naval Air Systems Command contract for this completely new turbofan, which had won a 1965 competition to provide an engine for the proposed VS(X) carrier-based anti-submarine aircraft. It was a condition that development should keep within a budget of US$96 million. This was achieved, and the engine became the T34, the VS(X) becoming the Lockheed S-3A Viking. Subsequently, several versions have been developed:

TF34-GE-2

Initial production version. T-O rating 41.3 kN (9,275 lb st). MQT completed August 1972, S-3A production total 187, US Navy service from February 1974. Later GE and the Navy defined a 4,000-hour TBO. In 1974 a modified form of this engine was selected to provide thrust for the Sikorsky S-72 RSRA.

TF34-GE-100

T-O rating 40.3 kN (9,065 lb st). Selected in 1970 to power the USAF Fairchild A-10A. Re-engineered to minimise unit price. Full-length fan duct and arranged for side mounting. Flew the first A-10A in May 1972. MQT October 1974. A-10A production total 733. In front-line use over Kosovo in 1999,

by which time GE was urging the USAF to re-engine the A-10 with a military version of the CF34-8 (described below). As the A-10 is expected to remain in the USAF/ANG inventory until 2028 it is claimed by GE that the lower operating costs of the later engine (derated to about 48.91 kN; 11,000 lb st) would result in an overall saving of US$3 billion to US$4 billion. The re-engining programme would, says GE, "pay for itself in nine years". An important bonus would be significantly enhanced flight performance. With the Dash-100 engine, loss of either engine results in a negative climb rate. By August 2001 the USAF was completing a Requirements Document for upgrading the 366 A-10A and OA-10A aircraft remaining in inventory service. It focuses on several major new avionics and precision weapons, but according to Lockheed Martin Systems Integration at Owego, New York (which supports the aircraft in the absence of Fairchild), the item causing the most concern is the engine. For several reasons, greater thrust is needed. An obvious answer would be a TF34 based on (for example) the CF34-10. A completely new engine has not been ruled out, but funding for it could not be found for this one application.

TF34-GE-400A

Replaced the TF34-GE-2 in the S-3A and derived aircraft from January 1975. Ratings were unchanged, but modifications included rearranged external piping, an adaptive control system for optimising accessory power extraction, and a simplified system for minimising rocket-gas ingestion.

CF34-1A

Commercial turbofan announced in April 1976. Certificated in August 1982. Powers Challenger 601-1A.

CF34-3A

Powers Challenger 601-3A.

CF34-3A1

Same ratings as CF34-3A. Powers Canadair Regional Jet 100 and Challenger 601-3R.

CF34-3B

T-O rating as CF34-3A but maintained to 30ºC (86ºF), with climb and cruise ratings increased. Certificated in May 1995 for service on the CL-604 in April 1996.

CF34-3B1

CF34-3B with airline-type ratings. Certificated concurrently with CF34-3B. Selected for Canadair Regional Jet 200 and Russian Tu-324 project. By August 2001 40 customers had ordered almost 1,800 CF34-3 engines, plus a further 1,550 options. This has required production to be ramped up to more than 400 in calendar 2001. At that time total CF34-3 flight time was in excess of 7.2 million hours.

CF34-8

Major growth version, initially planned with T-O rating 61.35 kN (13,790 lb st). Differences are shown in a drawing and described in the text below. This engine, originally designated Dash-8, was redesigned in 1996. GE sees a market for 2,000 aircraft in the 50- to 90-seat class, and is teamed with IHI

of Japan for development and production. The Japanese company is a 30 per cent risk-sharing partner on all CF34-8 versions, responsible for the complete fan, LP turbine and accessory drive gearbox. Extensive component testing at Lynn and at European Gas Turbines at Whetstone, UK, is said to have equalled or exceeded all targets. The first core engine went on test in December 1997, and the first of 15 complete CF34-8 development engines was completed in the same month. The first engine was fired up at the Small Commercial Turbofan Department at Lynn, Massachusetts, on 20 February 1998. After 10 hours of calibration and facility checkout, the power was advanced to 64.50 kN (14,500 lb st), 5 per cent above the initial certification rating with `some components exceeding design point performance'. In March 1998, the core went on test at Evendale, while the second complete engine went on test at the Tokyo facility of IHI. All versions, by mid-1999 the 8C, 8D and 8E, share identical turbomachinery and 87 per cent parts commonality, as well as a common FADEC. A derived version has been proposed as a retrofit to the USAF/ANG A-10 (see TF34-GE-100 above).

CF34-8C1

First variant of the so-called Growth versions, designated C for Canadair. Selected in February 1995 for stretched Canadair Regional Jet 700 (CRJ-700). In March 1998 Bombardier's Short Brothers delivered the first nose cowl for a CRJ-700. It is being used for fan-blade-off testing. According to the Belfast company, ``The complete nacelle can be used on a left or right engine, and the reverser will not only be more effective than conventional designs but will also be much simpler." Costs are expected to be sharply reduced by the elimination of blocker doors; the only moving part is the translating cowl. Between 10 March and mid-May 1999 the 8C1 completed an exhaustive series of 29 test flights (see photo of 747 testbed) ``with outstanding results". FAA certification was awarded ahead of schedule in November 1999. JAA certification followed in May 2000. The 8C1 engine entered service with the French airline Brit Air in January 2001.

CF34-8C5

Selected to power Canadair CRJ900. Like other Growth engines, to be in 64.5 kN (14,500 lb st) class. Growth engine flight test began on GE's 747 on 5 April 2001. CRJ900 flight testing began on 23 May 2001 with an 8C5 in the left position and an 8C1 on the right, on a 4-hour shakedown flight, followed by a performance-evaluation flight lasting nearly 3 hours. Two days later testing continued with two 8C5 engines, and a second CRJ900 was to enter flight test in late 2001. CRJ900 certification is due in third quarter 2002, with EIS with Mesa following in 2003.

CF34-8D3

Derivative of the CF34-8C configured for underwing installation. The CF34-8D family will have a thermodynamic rating in the 66.7 kN (15,000 lb st) class. Designated D for Dornier. Under development for the Dornier-designed Fairchild Aerospace 728JET regional airliner and the derived Envoy 7 business jet. Differs from 8C in installation and wing-mounted nacelle. The 728JET programme was launched on 29 April 1999 by a massive buy of 60 firm plus 60 options for Lufthansa City Line. The Envoy 7 was launched on 14 June 1999 by an order for 25 for the Flight Options fractional-ownership company. The first engine to test began six months behind the published schedule in December 2000. Flight testing slipped from second quarter 2001 to second quarter 2002. FAA certification of the CF34-8D3 is now targeted for September 2002, with EIS with Lufthansa slipped from May 2002 to 2003 (quarter not specified). Partners include Hurel-Dubois for the Papillon reverser and Aermacchi for nacelle and engine buildup. Fairchild Aerospace expects to use the CF34-8D derated to about Dash-8C1 level to power the projected short-fuselage (55-passenger) 528JET.

CF34-8E

Designated E for Embraer, and selected for 70-passenger ERJ-170 at T-O rating in the 62.28 kN (14,000 lb st) class. Very similar to CF34-8D but with different underwing installation. FETT was scheduled for June 2000, but slipped to `third quarter'. FAA Certification was expected in September 2001, but has slipped to the second quarter of 2002. The ERJ-170 is scheduled to fly in the fourth quarter of 2001 and enter service with Crossair in fourth

quarter 2002.

CF34-10D

The largely new Dash-10 is briefly described in the next section (CF34-10E). Rated in the 80 kN (18,000 lb st) class, the CF-34-10D will power the Dornier 928JET. It will have an underwing mounting similar to that of the 728JET. Industrial partners are the same as those for the CF34-10E.

CF34-10E

Though described by GE as a derivative engine, the fantastic success of earlier versions has enabled this version to be a substantially new engine. T-O ratings of different versions will be in the 82.26 kN (18,500 lb st) class, double that of early versions. The CF34-E will power the Embraer ERJ-190, with an underwing installation similar to that of the ERJ-170. Features of all Dash-10 versions include a fan scaled down from the GE90 (94B or possibly 115B), and HPC scaled from that of the CFM Tech 56, and a largely redesigned LPT. Partners include IHI for the LPT and other items, as for the Dash-8 family, and Techspace Aero of Belgium. TA's share of about 8 per cent will include the entire four-stage booster module (LPC). The reversers for the D and E versions of CF34-10 could be the Bombardier (Shorts) natural blockage type or the Papillon (see `CIVIL ENGINES') which is to be supplied by MHD, a company formed in June 1999 by Hurel-Dubois of France and Aermacchi of Italy. CF34-10 testing is expected to begin in first quarter 2002, for FAA Certification in early 2003.

At 31 March 1999 the civil CF34 engine powered over 500 active aircraft, with a total flight time exceeding 3.1 million hours and increasing at a fast-accelerating rate. In the preceding 12 months IFSD rate for all CF34 engines was 0.005 per 1,000 hours, while SVR was 0.063 per 1,000 hours. Engines in corporate aircraft are operated to an FAA-approved fixed maintenance cycle of 6,000 hours. Despatch reliability was steady at 99.98 per cent. In March 1999 a total of 277 CRJ aircraft were in service with 18 airlines, and firm and option orders for the CRJ family alone stood at 1,006. CF34 orders in 1998 exceeded US$1.5 billion, and to meet demand the CF34 production rate was increased from 238 in 1999 to almost 300 in 2000 and to over 400 in 2001.

According to Frank Klaus, General Manager of GE's Small Commercial Engine operation, "It cost about US$60 million to develop the TF34 into a commercial engine, and there were many who doubted it was a good investment. When Bombardier wanted a bigger engine for the CRJ700 many in GE advised extreme caution, and regarded the market forecast of 400 as wildly optimistic. Then the regional market exploded. I am thrilled and humbled to be in the middle of it. We have a backlog of more than US$7 billion on CF34 engines alone, excluding services...It's like a surfer riding the biggest wave he's ever seen."

Except where otherwise indicated, the following description refers to versions before the Dash-8C:

Type

Two-shaft turbofan.

Intake

Plain pitot, without struts, rings or guide vanes.

Fan

Single stage, diameter 1,118 mm (44.0 in), with 29 blades of forged Ti alloy, without part-span shrouds, rotating with spinner (pointed on CF34). Blades replaceable with engine installed. Bypass ratio (typical TF34 and CF34-1 through -3) 6.2; CF34-8C1, 4.8; CF34-8D, 5.1; CF34-10, 5.3. Mass flow (typical TF34) 153 kg (388 lb)/s at 7,365 rpm; CF34-1 through -3, 151 kg (332lb)/s at 7,300 rpm; CF34-8C1, 200 kg (440.9 lb)/s. Typical pressure ratio (except CF34-8) 1.5; CF34-8C1, 1.8; CF34-8D, 1.9. CF34-1 and -3 fan cases have Kevlar-box containment. The CF34-8C has a completely new fan with 28 advanced-design wide-chord blades. Diameter in the CF34-8 was to have been 1,158 mm (45.6 in); diameter in the Dash-8C1 and -8D is

1,173.5 mm (46.2 in), giving still greater increase in mass flow. Testing of this fan has been carried out at European Gas Turbines at Rugby, England. The new fan has solid containment, recently proved in the blade-out test. The CF34-10 has a further completely new wide-chord fan with a diameter of 1,346 mm (53.0 in).

LP Compressor

The CF34-10 introduces a four-stage LP compressor rotating with the fan.

Compressor

14 stages, with IGVs and first five stators variable. First nine stages Ti alloy, remainder high-Ni alloy. Typical core mass flow 21.3 kg (47 lb)/s at 17,710 rpm, with pressure ratio 14, giving OPR 21.0. The CF34-8 was to have a new 11-stage spool derived from the F414 fighter engine, with blisks welded into pairs of groups; after redesign, the Dash-8C1 has an HP compressor with only 10 stages, handling higher mass flow, with five variable stators. Stages 2-8 are scaled from the F414. Aerodynamic testing of this compressor began in September 1996. CF34-8C1 OPR is 28.0.

Combustion Chamber

Annular, with machined-ring Hastelloy liner and front dome providing ports for primer nozzles, igniters and low-pressure injection to 18 carburetting burners. CF34-8C1 will have a totally new machined-ring chamber based on GE90 technology, with many thousands of laser-drilled holes giving film cooling.

HP Turbine

Two stages, with first-stage nozzle vanes having film and impingement cooling; convection-cooled rotor blades, first stage air-cooled. TF34 TET 1,225ºC. CF34-8C1, both stages cooled.

LP Turbine

Four stages with shrouded blades. The CF34-8 and -10 have an improved rotor with discs inertia-welded into two pairs.

Accessories

A radial drive from the front of the HP shaft powers a horseshoe `banana' gearbox around the underside of the compressor case. Up to eight pads available.

Control System

Before CF34-8C1, hydromechanical, with electronic amplifier. Contamination-resistant carburetting fuel control. CF34-8C1, dual-channel FADEC. The CF34-10 family will almost certainly be FADEC controlled.

Fuel Specification

TF34, JP-4 or JP-5.

Oil System

Enclosed, pressurised dual system, with vent along centre shaft.

Dimensions

Length:	

TF34	2,540 mm (100.0 in)
CF34, see table	
Diameter:	
TF34-2 and -400	1,321 mm (52.0 in)
TF34-100	1,245 mm (49.0 in)
CF34, see table	

Weight, Dry

TF34-2	661 kg (1,458 lb)
TF34-100	653 kg (1,440 lb)
CF34, see table	

Performance Rating

T-O: See model listing

Specific Fuel Consumption

T-O, as above:	
TF34-400	10.3 mg/Ns (0.363 lb/h/lb st)
TF34-100	10.5 mg/Ns (0.370 lb/h/lb st)
CF34, see table	

Contract Price

October 1998, CF34-3A1 for 30 CRJ-100 and CF34-8C1 for 20 CRJ-700, US$140 million; April 2001, CF34-3B1 for 150 CRJ-200, US$600 million.

CF34 variant	**CF34-1A**	**CF34-3A**	**CF34-3A1**	**CF34-3B**	**CF34-3B1**	**CF34-8C1**	**CF34-8D**
S/L flat rating	15ºC/59ºF	21.1ºC/70ºF	21.1ºC/70ºF	30ºC/86ºF	30ºC/86ºF	30ºC/86ºF	30ºC/86ºF
T-O thrust	38.48 kN	38.8 kN	38.8 kN	38.8 kN	38.8 kN*	56.38 kN	58.73 kN
	(8,650 lb st)	(8,729 lb st)	(8,729 lb st)	(8,729 lb st)	(8,729 lb st)*	(12,670 lb st)	(13,200 lb st)
Length	2,616 mm	2,616 mm	2,616 mm	2,616 mm	2,616 mm	3,264 mm	3,251 mm
	(103.0 in)	(103.0 in)	(103.0 in)	(103.0 in)	(103.0 in)	(128.5 in)	(128.0 in)
Maximum diameter	1,245 mm	1,245 mm	1,245 mm	1,245 mm	1,245 mm	1,321 mm	1,321 mm

	(49.00 in)	(49.00 in)	(49.00 in)	(49.00 in)	(49.00 in)	(52.00 in)	(52.00 in)
Weight, Dry	717 kg	737 kg	751 kg	757.5 kg	757.5 kg	1,066 kg	1,120.4 kg
	(1,580 lb)	(1,625 lb)	(1,655 lb)	(1,670 lb)	(1,670 lb)	(2,350 lb)	(2,470 lb)
TET (T-O, 5 min)	857ºC	871ºC	898.9ºC	898.9ºC	898.9ºC	-	-
	(1,575ºF)	(1,600ºF)	(1,650ºF)	(1,650ºF)	(1,650ºF)	-	-
SFC (T-O/APR)	10.18 mg/Ns	10.10 mg/Ns	10.10 mg/Ns	9.80 mg/Ns	9.80 mg/Ns	-	-
	(0.360 lb/h/lb)	(0.357 lb/h/lb)	(0.357 lb/h/lb)	(0.346 lb/h/lb)	(0.346 lb/h/lb)	-	-
SFC (37K, M0.74)	20.13 mg/Ns	19.92 mg/Ns	19.92 mg/Ns	19.49 mg/Ns	19.49 mg/Ns	18.78 mg/Ns	18.96 mg/Ns
	(0.712 lb/h/lb)	(0.704 lb/h/lb)	(0.704 lb/h/lb)	(0.689 lb/h/lb)	(0.689 lb/h/lb)	(0.664 lb/h/lb)	(0.670 lb/h/lb)

* Flat rated to 22.78ºC (73.0ºF)

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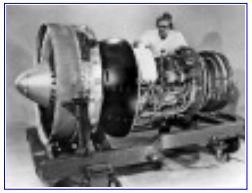
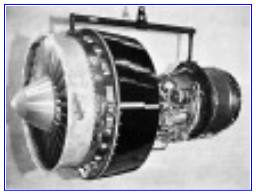


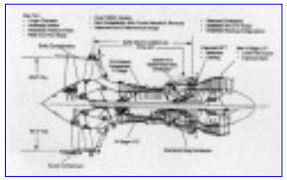
TF34-100



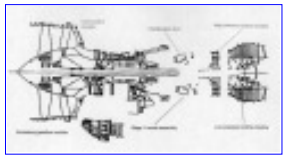
TF34-400A

Cutaway drawing of typical CF34-3

CF34-3A1/-3B/-3B1
(1998)



Comparison between CF34-3A1 (lower half) and the original CF34-8C1 (upper half)

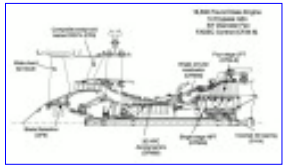


Modules for CF34-8C1

CF34-8C1 on crosswind test
(2000)

Flight testing the CF34-8C1 on GE's 747 testbed
(2000)

CF34-8C1
(2001)



Longitudinal section through CF34-10
(2001)

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 04 July 2001

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General Electric was a pioneer of aircraft-engine turbosuperchargers, but did not make aero-engines until 1941 when it was picked by General H H Arnold to produce an American version of the Whittle turbojet. GE was already working on axial compressors and a turboprop. This launched it into the

design and production of military engines. In October 1953, a Small Aircraft Engine department was formed at Lynn, Massachusetts, which ever since has produced small jet and helicopter engines. The headquarters of the Aircraft Gas Turbine department (as it then was) moved to Evendale, a suburb of Cincinnati in Ohio. Here, by adding an aft fan behind a basically military turbojet, GE got into airline propulsion systems.

Today, GEAE is one of the world's biggest players in aero-engines. Its partnerships with other companies are shown in the `Collaborative Programmes' graphical insert in the preliminary sections of this product. They include SNECMA (France) in CFMI and Honeywell in CFE. Indeed, in mid-2001 the hot news was a possible merger between GE and Honeywell. Another partner is Allison Advanced Development Co, on the JSF (see in International section) and the USAF IHPTET (Integrated HP turbine-engine technology) R&D programme.

In 1998, CFMI sold 1,126 engines, while GE sold a further 86 CF6 and GE90 engines to bring the combined total to a claimed 54 per cent of the world total of engines for large commercial aircraft. GEAE revenues in 1999 were US$10.56 billion.

GE Engine Services, Inc, a unit of GE Aircraft Engines and a wholly-owned subsidiary of General Electric Company, enjoyed its most outstanding sales year in 1999 with revenues exceeding US$5 billion. The 1998 total was nearly US$5 billion in sales. The 1997 total was US$3 billion and the 1996 figure US$2.3 billion. GE Engine Services acquired Greenwich Air Services, Inc, a multibillion dollar enterprise operating nine overhaul and repair sites in the United States and Scotland. In September 1997, GE completed acquisition of UNC Inc, for US$330,541,230 cash plus assumption of UNC debt. UNC specialises in the support of corporate and military jets. In 1998, GE VARIG Engine Services was opened to serve the Latin American market. In September 1998, Engine Services signed a US$100 million contract to maintain the JT9D engines of Evergreen International, the work to be done at Nantgarw (Wales). In February 1999, it signed an agreement with Bedek Aviation of Israel for joint marketing of aircraft and engine maintenance to customers worldwide. In November 1999 an agreement was signed with IHI (see under Japan) and All Nippon Airways to acquire a controlling interest in the ANA-IHI engine maintenance company. In December 1999 the GE Engine Services subsidiary On Wing Support opened a new facility at Los Angeles. In January 2000 GEAE opened an internet-based Customer Web Center (CWC). Within 11 months 360 customers were online and CWC business had exceeded US$1 billion.

Not including On Wing Support Inc, GE Engine Services operates overhaul and repair facilities at: Xiamen, China; Nantgarw, Wales; Singapore; Kuala Lumpur, Malaysia; Ontario, California; Stother Field, Kansas; Miami, Florida; Dallas/Fort Worth, Texas; Prestwick, Scotland; Petropolis, Brazil; Rio de Janeiro, Brazil; and Cincinnati, Ohio; as well as at the six Total Aircraft Service Centers of Garrett Aviation Services.

USAF Support

Following prolonged debate about how best to support engines in service, either by outsourcing or by in-house effort, a long-term contract was signed on 1 November 1999. The text reads: ``General Electric Company, Cincinnati, Ohio, is being awarded a US$2,800 million indefinite-delivery/indefinite-quantity contract F33657-98-D-0019 to provide for various quantities of the F110 and F118 engines in support of the F-15, F-16 and B-2 aircraft. These efforts also support equipment, field support and retrofit kits. Further funds will be obligated as individual delivery orders are issued. Expected completion date is November 2014. On 7 December 1999 a further multi-year contract followed: $1,975,500,000 indefinite-delivery/indefinite-quantity contract F33657-99-D-2050 to provide for technical services from January 2000 through December 2014 for the Component Improvement Program supporting the F110 engine on the F-16 aircraft, the F101 engine on the B-1

aircraft, the F118 engine on the B-2 aircraft, the J79 engine on the F-4 aircraft, the TF34 engine on the A-10 aircraft, the TF39 engine on the C-5 aircraft, and the J85 engine on the T-38 aircraft. Expected completion date is 31 December 2014. Aeronautical Systems Center, Wright-Patterson AFB, Ohio, is the contracting activity."

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 04 July 2001

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ENGINE ALLIANCE - ENGINE ALLIANCE, A JOINT COMPANY OF GEAE AND P&W

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On 8 May 1996, the above companies issued the following statement:

``GE Aircraft Engines and Pratt & Whitney have signed an agreement to jointly develop a new jet engine that will bring significantly lower operating costs to the growth Boeing 747-500X/600X aircraft.

``GE and P&W agreed to begin a joint engine development programme after determining that a new centreline engine, rated at 72,000 to 84,000 lb thrust, would be able to provide substantially better weight, fuel burn, noise, and cash operating costs to the growth 747 aircraft currently being studied by the Boeing Company.

``By combining expertise and resources, GE and P&W will be able to provide Boeing and the airline industry with the best engine for the aircraft in a timely fashion and at a significantly lower cost.

``Plans call for the engine to be developed and marketed by a separate joint venture owned equally by GE and P&W. The companies are defining the engine's technical specifications and the structure of the 50/50 joint venture. Management for the venture will be held solely by the two companies, though additional global revenue sharing participants will be considered for the programme. GE and P&W reviewed the project with Boeing, which urged the two companies to explore the joint programme."

This announcement was of the greatest interest. Apart from current joint effort on the propulsion system of a future SST, this is the first time the two rival giants have collaborated. Moreover, the proposed engine will lie in a thrust class well within the capability of the partners' existing products.

The collaborative project has been driven by several factors. Chief among these was an intensification of competition, and in particular the outstanding technical success of the Rolls-Royce Trent (in this market, the Trent 900). Another factor is generally declining profit margins, in part resulting from this increased competition and resultant price cutting. A third factor is the spiralling cost of developing new large engines.

The thrust needed could easily be met by existing engines. The PW4084 entered service at 343 kN (77,000 lb st), with obvious growth capability. The GE90-76B is spot-on for thrust, and GE would dearly like to find a second application for this all-new engine. The fact that neither engine has been offered may in part stem from Boeing's wish not to repeat the process of having to certificate three different engines, but a more telling reason is that the Trent 900 is recognised as a very formidable competitor.

At the 1996 Farnborough Air Show, the two company presidents, E F `Gene' Murphy and Karl J Krapek, gave a presentation on this programme which, by its hard-hitting nature, inevitably showed the respect they feel for the Rolls rival. At the same time, one must ponder on where this leaves the PW4000 and GE90.

Mr Murphy said, in part ``When Boeing began to define a program for 747 growth, it aimed to reduce DOC by 10 per cent. Both Pratt & Whitney and ourselves studied possible derived engines. We both concluded that no such engine can meet all the requirements. Clearly, 747 growth needs a new centreline engine, but neither of us was willing to commit the huge investment. Boeing urged us to combine forces. The result is the Engine Alliance. This is a new company, committed to the long term, with the support of the world's two largest jet manufacturers".

Mr Krapek said ``Our joint venture is only five months old, but its lineage is impressive. I think of it as the Dream Team of aerospace. Since we began discussions in March (1996) GE and Pratt have

assigned top people to the Alliance. While the initial engine is called the GP7176 (76,000 lb thrust), I'm confident that the entire GP7000 family will be a gold medal winner in all its configurations. It will have a nine per cent fuel burn advantage over current offerings. It will also be two to three points better in fuel burn than any competitor's derivative" (Rolls-Royce contests this assertion).

The Engine Alliance is a limited-liability company, whose staff is hoped never to exceed 25. All design and development is being handled by dedicated teams under Hughes at GE and Harper at P&W. On the other hand, the Alliance will have its own marketing staff, and all GP7000 engines will be sold independently of other products from the two partners.

To minimise development cost and risk, and enable the Alliance to meet its EIS target of December 2000, no attempt will be made to introduce completely new materials or technologies, but the GP7176 will make use of `critical technologies developed by the partners for other engine programmes'. At the same time, it will be all-new; no parts will be common to existing engines.

In the original *issue 0* edition of this binder, it was speculated that GE would have overall responsibility for the HP compressor, combustor and HP turbine, while Pratt & Whitney would take on the fan and LP section, a reduction gearbox and high-speed LP turbine. In the event, this guess was wrong in one respect: P&W is designing a traditional multistage LP turbine with direct drive to the fan, but has now gone ahead with its own engine, with a 3:1 reduction gearbox (see PW8000). The Connecticut partner will make up its 50 per cent share by undertaking assembly and test.

US Federal regulators gave the Alliance the go-ahead in 1996. The European Commission approved the venture in September 1999.

Design definition was to be completed in December 1996, detail design and initial procurement was to begin in January 1997, the FETT (First Engine To Test) was scheduled for mid-1998, and FAA certification requirements were to be completed by the end of 1999. As noted, entry to service was planned in December 2000. This demanding timescale stemmed from Boeing's urgent need to sell as many 747-500X/-600X aircraft as possible before Airbus could launch the A3XX. In the event, customer interest was lukewarm, and in February 1997 Boeing abandoned these proposals, saying it would instead concentrate on stretching the 767 and on the 747-400IGW. Both these projects interested the Alliance, and the increasingly firm specification for the A3XX made this also a target, but for a more powerful GP7000 engine.

That there is a market for a new engine in this thrust bracket was never doubted. In 1996, GE estimated the total that might be won as `3,000 engines', but a year later they increased this to `4,500-5,000 engines by 2016'. Airbus then estimated that by 2014 there would be a `market for 1,380 airliners of 500 seats and above', almost all of which would be four-engined. A similar estimate was repeated in June 1997 by Rolls-Royce, whose Trent 900 achieved the first MoU for propulsion of the A3XX. In 1998 Airbus revised its estimate to 1,332 by 2017. On the other hand, on 10 November 1998 the Engine Alliance announced that its A3XX market estimate is a mere 430 aircraft by 2020. The Editor cannot help recalling the grossly inaccurate estimates of the possible market for the 747 in the late 1960s before that aircraft flew.

According to the Alliance, `Boeing's decision to cancel the 747-500X/-600X does not eliminate the need for new high-capacity four-engine aircraft . . . Boeing is studying derivatives of the 747-400 with increased range and size, while Airbus continues to develop an all-new aircraft . . . Thus, two engine models are being developed using the same core with variations in the low spool to meet the different thrust and installation requirements . . . the GP7000 can be developed to meet all of the schedules proposed by Airbus and Boeing.'

By 1999 the waters had been to some degree muddied by the decision of one of the partners to target the 60K market with an enlarged conception of its PW8000, with a geared fan. Such an engine, if successful in all respects, could seriously damage the market for traditional direct-drive engines in the

same thrust class. Pratt & Whitney has been careful to suggest that the PW8000 will be aimed at a thrust category slightly below that of the Alliance engines, but the difference is slight, and the PW8000 could hardly be expected never to grow in power.

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 04 July 2001

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WILLIAMS INTERNATIONAL

WILLIAMS ROLLS FJ44

US military designation: F129

Williams International planned the FJ44 as its first engine intended specifically for conventional manned aircraft, especially light business jets and training aircraft. Design of what was at first called the WR44 began in 1971, using the WR19 as a basis. BPR and OPR were both increased. Prototype engines were tested in a two-man VTOL vehicle in 1973. At a rating of 3.78 kN (850 lb st), the first WR44 was chosen for the Foxjet, but the engine was then again redesigned to increase BPR and thrust, being redesignated FJ44. In 1982, Hawker Siddeley of the UK licensed Chandler-Evans Inc to use the British company's digital technology in the engine's control system. The first FJ44 `achieved design thrust at a very modest TET'.

Since then different FJ44 versions have appeared. All are characterised by simplicity, low cost, high reliability and the ability to replace turboprops in aircraft operating from small airports. All versions allow fan or hot section replacement while installed on aircraft.

Production engines are assembled at Williams' Ogden plant. In 1989, Rolls-Royce joined as a partner to assist design, production and support of the FJ44 worldwide. The two companies formed Williams Rolls Inc for this purpose. Since then engineering development has been very successful, and several customers have led to a growing number of versions:

FJ44-1A

T-O rating 8.45 kN (1,900 lb st). For Cessna CitationJet, Sino-Swearingen SJ30, Century Jet (previously called Paragon Spirit), IAR/AEDCO Noga VI, Promavia Jet Squalus, Contender 202 and possibly Promavia ATTA. Claimed ``quietest engine in production, less than half permitted Stage 3 limits''.

FJ44-1B, F129-WR-100

T-O rating 6.67 kN (1,500 lb st). For Cessna 526 JPATS contender.

FJ44-1C

T-O rating 6.67 kN (1,500 lb st). For Saab SK60, 244 ordered by Royal Swedish AF for re-engining 105 aircraft. First flight with new engines 6 October 1995. Deliveries of re-engined aircraft 6 September 1996 to 1998. Also powers Alberta Aerospace Phoenix.

FJ44-1D, F129

T-O rating 8.45 kN (1,900 lb st). For Lockheed Martin DarkStar (unmanned low-observable, Tier III Minus programme).

The FJ44-1 completed initial FAA certification on schedule in March 1992, and first production deliveries followed in the same year. JAA certification followed. By third quarter 1999 over 1,000 engines had been delivered.

FJ44-2A

T-O rating 10.23 kN (2,300 lb st), flat-rated to 22ºC. Core similar to FJ44-1, but new fan and intermediate compressor components to increase airflow, pressure ratio and efficiency. FAA certification completed in July 1997. Powers Raytheon Premier I (production of which was increased from 48 to 60 per year in September 1999), Sino Swearingen SJ30-2 and Scaled Composites Proteus, as well as other aircraft to be announced. A version of the FJ44-2 has been tested to a pressure altitude of 19,812 m (65,000 ft).

FJ44-2C

T-O rating 10.675 kN (2,400 lb st), flat-rated to 22ºC. Certified April 2000. Powers Cessna 525A Citation CJ2.

Type

Two-shaft turbofan.

Fan

Single-stage with integrally bladed rotor in titanium with 20 wide-chord shroudless blades. BPR (Dash-1) 3.28. FJ44-2A has an advanced fan of higher FPR, with wide-chord sweptback blades, with diameter increased to 500 mm (19.69 in). Mass flow (FJ44-1A) 29.575 kg (65.2 lb)/s, (FJ44-1C) 26.49 kg (58.4 lb)/s.

LP Compressor

Single-stage axial rotating with the fan on the LP shaft. FJ44-2 has a three-stage core booster.

HP Compressor

Single-stage centrifugal. OPR (FJ44-1A) 12.7, (FJ44-1C) 10.3. BPR (1A) 3.3, (1C) 3.4.

Combustion Chamber

Radial delivery from compressor leads to folded annular combustor.

HP Turbine

Single stage with inserted blades.

LP Turbine

Two stages with inserted blades.

Jetpipe

Single fixed-area nozzle for combined flows from core and full-length fan duct. FJ44-2, 18-lobe mixer.

Control System

Hydromechanical unit (HMU) on CitationJet and FJ44-2C, electronic with HMU reversion on SK60 and FJ44-2A.

Dimensions

See GA drawings

Weight, Dry

FJ44-1A	205 kg (452 lb)
FJ44-1C	208 kg (459 lb)
FJ44-2A	not final

Performance Ratings

T-O, S/L: See model listing	
Max continuous (11,000 m, 36,090 ft, M0.7)	2.25 kN (506 lb)

Specific Fuel Consumption

T-O, S/L:	
FJ44-1A	13.45 mg/Ns (0.475 lb/h/lb st)
FJ44-1C	12.91 mg/Ns (0.456 lb/h/lb st)
FJ44-2A	lower than Dash-1

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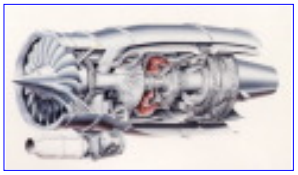


Cutaway drawing of FJ44-1

General arrangement of FJ44-2 (2000)

FJ44-1 (2000)

FJ44-1A (Paul Jackson) (2000)

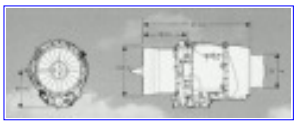
FJ44-1C (Paul Jackson) (2000)

General arrangement of FJ44-1 (2000)

FJ44-2 (2000)

FJ44-2A (Paul Jackson) (2000)

FJ44-2C (Paul Jackson) (2000)



General arrangement of FJ44-2 (2000)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 04 July 2001

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UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY F119

This advanced augmented turbofan was selected by the US Air Force over the rival General Electric F120 to power the Lockheed F-22A Advanced Tactical Fighter in April 1991. Basic requirements were the simplest and most robust design for maximum reliability and maintainability, supercruise (supersonic persistence) capability without afterburner and a 2-D (two-dimensional) nozzle incorporating limited thrust vectoring. The rival companies began formal development of their competing designs in 1983. From the outset Pratt & Whitney chose to offer a mature low-risk engine, even if in some respects it might appear to be less advanced than the rival F120. This philosophy was followed when, as the result of weight growth in the competing ATF aircraft, the thrust requirements were upgraded in early 1988. By this time the design had long been fixed and the first YF119 demonstrator was about to run. The decision was taken to meet the extra thrust requirement by a subsequent slight increase in fan diameter, leaving the rest of the engine unaltered.

Bench testing began in December 1988. In the summer of 1990, F119 engines made 65 flights totalling 153 hours, with no stalls or shutdowns. By April 1991, engines on ground test had run over 3,000 hours, including 1,500 with the 2-D nozzle. Since then total test hours have exceeded 6,500, most with the SCFN (see later under Nozzle).

F119-PW-100

The engine for the Lockheed Martin/Boeing F-22A EMD (Engineering and Manufacturing

Development) has been in progress since August 1991. It has `significantly fewer and more durable components than previous fighter engines'. The first shipment to Lockheed Martin took place in September 1996 for installation in the first flight test F-22A Raptor, which was to have flown in May 1997 but did not do so until 7 September 1997. P&W is delivering 27 engines to power the nine EMD aircraft. By the end of 1998 flight testing of the two F-22A prototypes had generally been very successful, though the second aircraft experienced a two-axis vibration while flying at high power at altitude. The left engine was replaced, but the condition could not be replicated. This slightly delayed transfer of this aircraft to Edwards until 26 August 1998. At that time aircraft test hours were still under 60, but the schedule was then accelerated in order as quickly as possible to reach the magic figure of 183 hours discussed by Congress as a prerequisite for production. This figure was achieved in early 1999

Despite an increasing emphasis on cost-reduction, Pratt & Whitney was, in August 1997, awarded a US$232,871,029 increase in funding `to provide for restructuring the EMD contract'; this extends the programme by 15 months, deletes one flight-test engine and adds two development engines. Thus, engines to final Dash-100 standard were due to enter production in 1999, for delivery from February 2001. EIS is scheduled for 2004. On current planning, the USAF will purchase 777 engines, to power 339 aircraft.

Compared to the F100, the F119-100 has `40 per cent fewer parts'. It is expected to require `75 per cent fewer shop visits and 50 per cent less support equipment'. According to Pratt, ``Reliability is being demonstrated at levels twice that of today's fighter engines, and it requires only half the maintenance manpower".

JSF-119

As related under JSF in the International section, all three proposals for the multiservice Joint Strike Fighter selected derivatives of the F119 as first engine choice. Originally, each version was identified by an SE (single-engine) designation, but this is now falling into disuse. For each competitor aircraft, there are two versions of the engine, one CTOL and the other STOVL. All JSF119 versions are more powerful than the F119-PW-100.

JSF119-SE615

The McDonnell Douglas team proposed to use an engine almost identical to the F119-PW-100. This team was eliminated in November 1996.

The two remaining competing teams propose to use engines with major changes, but retaining `a common F119 core, common materials and manufacturing processes, common controls and externals architecture, and common support systems'. New features include advanced diagnostic and prognostic systems, automated field data and test systems and automatic logistics support. Significantly modified versions are required for the STOVL version of each aircraft. For both JSF119 versions the programme schedule is: Preliminary Design Review and start of manufacture (first chip cut from new fan blisk), March 1997; Critical Design Review, November 1997; First Engine To Test, summer 1998 (see below); delivery of flight-test engines, late 1999; JSF first flight (CTOL version of both aircraft) March 2000.

JSF119-611

Engine for the Lockheed Martin X-35. It has a fan intermediate in diameter between that of the F-22A engine and the JSF119-614. The engine for the CTOL X-35A went on test at West Palm Beach on 11

June 1998. The engine for the STOVL X-35B version was to go on test before the end of 1998. Compared with the F-22A engine, this features increased TET and uprated LP turbine stators in order to transmit 18,643 kW (25,000 shp) through a flexibly mounted shaft to a clutch and bevel gearbox driving a forward lift fan. This complete shaft/fan system is on test at Rolls-Royce Indianapolis. The engine's main nozzle is axisymmetric and fully variable, and is attached by three rotating segments giving 110° vector travel. On each side of the midship case are high-capacity bleeds serving roll-control arms. These, and the main nozzle, are also the responsibility of Rolls-Royce.

JSF119-614

Engine for the Boeing X-32. This has a significantly larger fan raising BPR to over 0.6, making it arguably the most powerful fighter engine ever, with a sea-level rating of at least 180 kN (40,500 lb st) without augmentation. The CTOL version for the X-32A went on test on 18 June 1998, and the STOVL version for the X-32B was scheduled to go on test at the end of 1998. This has valves to divert the entire mass flow out through two Harrier-type side nozzles which not only vector but also retract into the airframe for high-speed flight. Upstream on the underside of the engine is a jet screen to control aerodynamics and in particular keep hot jet gas from being reingested. The nozzles and screen are Rolls-Royce contributions. The main-engine nozzle is a Pratt & Whitney 2-D design with limited vertical vectoring to enhance inflight manoeuvrability.

Diagrams and further details of both JSF119 STOVL versions appear under JSF in the International section, together with a photograph of machining the first blisk fan rotor for the JSF119-614.

The following description refers specifically to the F119-PW-100:

Type

Two-shaft augmented turbofan.

Intake

Anti-iced fixed ring, trailing-flap guide vanes and central cone supporting LP front bearing.

Fan

Development engines: three widely separated stages with wide-chord shroudless blades of titanium alloy integral with each hub. To reduce weight, the first-stage blades are hollow. Integrally stiffened fan duct split for easy access. BPR about 0.45.

Compressor

Six-stage spool with integrally bladed rotors with blades of low aspect ratio and high stage loading. Stators of Alloy C high-strength burn-resistant titanium. IGVs and first two stator stages variable. Rotation in opposite direction to the fan. Integrally stiffened casing split into halves for access. OPR in the 35 class.

Combustion Chamber

Short high-intensity annular of patented Floatwall type, with multiple airblast nozzles for virtually smokeless burning.

HP Turbine

Single stage, with high stage loading. Advanced single-crystal blade material with multipass convection

and advanced film cooling.

LP Turbine

Single stage, rotating in opposition to HP spool.

Afterburner

Single spray ring for combustion downstream of both fan duct and core. Alloy C high-strength burn-resistant titanium in both augmentor and nozzle.

Nozzle

P&W's third-generation Two-Dimensional Convergent Divergent (2DCD) is the nozzle for the F119-PW-100 engine. This offers variable throat area, variable throat to exit area ratio and ±20º pitch vectoring. Features include advanced cooling techniques and lightweight and affordable structural materials. The nozzle is able to vector from stop to stop in one second. The vector angle is controlled by the aircraft flight-control system through the FADEC.

Accessories

Grouped for immediate access from below, driven by radial shaft from HP spool. All LRUs mounted only one deep.

Control System

`Third-generation' FADEC, with integrated engine/aircraft controls.

Dimensions

Generally similar to F100.

Weight, Dry

Similar to F100.

Performance Rating

S/L	155.6 kN (35,000 lb st) class

Contract Price

30 December 1999, US$179,965,669 modification to fixed-price contract for 12 F119-PW-100 engines and associated preliminary field support for F-22 Production Representative Test Vehicle programme.

UPDATED

Two-dimensional convergent/divergent nozzle

Cutaway drawing of F119-PW-100 (for inlet/fan, see drawing of F100-PW-229IPE fan)
(1999)

F119-PW-100 on test, showing nozzle in full afterburner

Pratt & Whitney F119-PW-100 with 2DCD nozzle wide open

The first CTOL JSF119-611 on test in summer 1998 (1999)

The first CTOL JSF119-614 on test in summer 1998 (1999)

Cutaway drawing of a baseline CTOL JSF119 (1999)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 04 July 2001

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GE AIRCRAFT ENGINES

GENERAL ELECTRIC F110

The **F110** (previously F101 DFE) is a fighter engine derivative of the F101. It first ran in late 1979. In early 1984 the USAF selected the F110 to power the majority of future F-16 aircraft.

The following are current versions of the F110:

F110-GE-100

Initial USAF engine, also selected by Bahrain, Egypt, Greece, Israel and Turkey. Delivery of production F-16C/D aircraft with F110 engines began in mid-1986 with EIS in October 1986. The US Navy selected the F110-100 to power its F-16Ns for the adversary role in its Top Gun programme. At 30 June 1997, 1,610,000 hours had been flown in 812 F-16s in current use, excluding those of Israel; the high-time engine had then logged 4,634 flight hours and unscheduled SVR was 3.65 per 1,000 flight hours. By late 1996, nearly 1,600 engines had flown over 2 million hours worldwide.

F110-GE-100A

Modification of -100 for Israel, giving additional emergency thrust at low level.

F110-GE-100B

Upgraded engine with FADEC and improved combustor and turbine. Inspection interval is 4,000 cycles.

F110-GE-400

Powers F-14B and F-14D Tomcat for the US Navy. First production aircraft with this engine delivered to Navy in November 1987 for EIS in April 1988. Tomcats powered with this engine show a significant improvement in fuel consumption and the ability to catapult launch without use of an afterburner, resulting in a 61 per cent time-to-climb reduction and a 62 per cent improvement in mission range. By 30 June 1997, 477,900 hours had been flown in 127 F-14 aircraft. The high-time engine had accumulated 3,776 hours, and the unscheduled SVR was 2.6 per 1,000 hours.

F110-GE-129

This GE Improved Performance Engine (IPE) is the successor to the F110-GE-100. Through the use of improved design and materials, higher operating temperatures, speeds and pressures, the GE IPE increases thrust levels by as much as 30 per cent in certain areas of the flight envelope, while retaining more than 80 per cent parts commonality. The IPE's digital electronic control has 50 per cent fewer parts than previous controls and offers substantially improved reliability. The IPE first flew in an F-16C/D in August 1988 and a very successful USAF field service evaluation programme was completed in mid-1992. Service entry was achieved in January 1992. The engine has also been selected to power F-16s for Bahrain, Chile, Egypt, Israel, Turkey and the United Arab Emirates, as well as for Japan's F-2 (see below). By mid-2001 the number of -129 engines ordered exceeded 2,100, for more than half the global F-16C/D fleet. By 30 June 1998, a total of 420 GE-129 engines had flown over 275,000 hours in the F-16. The unscheduled SVR was 1.46 per 1,000 hours (12 month rolling average). In December 1999 the USAF awarded GE a US$12,663,575 contract modification to provide 264 retrofit kits for the engines already in service. These kits are to ``maximise safety and reduce maintenance requirements by reducing high screech levels and resolving exhaust burn anomalies''. Expected completion date is 30 December 2002.

In May 1996, the -129 completed flight testing in an F-15E, and qualification on this aircraft was awarded in September 1997, opening the way to GE-powered versions of the F-15. In April 1997, a field service evaluation began which involved 1,900 engine flight hours on two F-15E aircraft, in 600 sorties. No spare engine or extra maintenance was called for. The evaluation, completed in December 1998, exceeded all objectives. With slightly different external dressing, the -129 will also go into production in 1999, in partnership with Japanese industry, for the Mitsubishi F-2.

The US Air Force awarded GE a development contract to design and validate an F110 engine ejector nozzle kit, derived from GE's Low-Observable Axisymmetric (LO-Axi) exhaust system, for USAF F-16C/Ds. During tests, the LO AXI system has reduced nozzle hardware temperatures by 278-500ºC (500-900ºF) while operating at maximum afterburner conditions. The F110 engine ejector nozzle kit, designed for existing and future GE F110-powered F-16C/Ds, will provide improved durability through design features that draw air from the engine nacelle to cool the nozzle's divergent flaps and seals. This will substantially reduce nozzle temperatures, thereby increasing nozzle operating life up to fourfold, compared with the average operating life of current exhaust systems. In addition, the ejector nozzle kit, with almost 400 fewer parts than the current F110 nozzle, will reduce maintenance time requirements by as much as 90 per cent. Flight testing of the ejector nozzle on an F-16C/D is expected to be completed in 1999.

F110-GE-129EFE

Since 1996, GE has tested a new fan comprising three wide-chord blisks with aerodynamics based on the F118 and structural design similar to that of the F120. This increases both mass flow and pressure ratio. In a self-funded programme, this new fan has now been combined in the Enhanced Fighter Engine (EFE) with a radial-type augmentor to some degree derived from that of the smaller F414 engine, which reduces the number of parts by 25 per cent, significantly lowering maintenance costs. The F110-GE-129EFE is installationally interchangeable with the original GE-129, and requires no airframe changes to F-16C/D or F-15E aircraft. In

addition, existing GE-129 engines can be upgraded to EFE configuration. The EFE has a demonstrated growth capability to 160 kN (36,000 lb st), but operating at GE-129 thrust would reduce TET 55ºC and make possible dramatically increased TBO - typically from 4,000 to 6,000 hours - and reduced maintenance. The EFE is to undergo performance and endurance testing at Arnold in 2000. Qualification is targeted for mid-2001.

F110-GE-132

Following four years of work on the F110-GE-129EFE, this new version of the F110 was launched in March 2000 by an order from the United Arab Emirates for 80 F-16 Block 60 aircraft. The new engine follows 129EFE but with a further developed radial afterburner derived more closely upon the F120 (JSF) engine and with a PMR-15 graphite-composite fan duct. Flight testing and Full Qualification are scheduled for 2002, with deliveries due in the same year. GE expects to sell kits (mainly the blisk fan and new afterburner hardware) for converting F110-GE-129 engines installed in F-16 Block 50 aircraft. Modification to Dash-132 standard gives the thrust cited below or, by retaining the existing Dash-129 rating, an engine life increased up to 50 per cent. The upgraded engine can be installed in Block 50 aircraft without structural or mount modifications.

Under USAF contract, GE has tested an ejector nozzle which is available for retrofit to most F110 engines. By using the large engine-bay airflow to form a cooling film on the inside wall of the nozzle, the metal temperature is reduced by `hundreds of degrees'. The ejector nozzle is also simplified and has reduced IR signature. Scale model testing began in early 1999, with endurance testing and qualification flight testing scheduled for 2000.

By April 2000 more than 2,200 F110 engines had entered service. GE claims ``more than 80 per cent of orders for F-16C/D aircraft, with the best safety and reliability record on that aircraft".

Intake

Direct pitot, with LP front bearing supported by 17 (-129, 13) forward-sloping inlet guide vanes, with variable-incidence trailing flaps. Bleed air anti-icing.

Fan

Three stages. Solid titanium rotor blades, first stage with part-span snubbers. Casing has special containment features. Mass flow (-100 and variants) 122.4 kg (269.8 lb)/s, (-400) 120.2 kg (265.0 lb)/s, (-129) 122.5 kg (270 lb)/s, (-129EFE) normally limited by F-16 inlet to 125.0 kg (275.6 lb)/s. Bypass ratio 0.76. New fan on -129EFE, see above.

Compressor

Nine stages, derived from that of the F101-102. Inlet guide vanes and next three stator stages variable. Overall pressure ratio, F110-100 and -400, 30.4; F110-129, 30.7.

Combustion Chamber

Annular, scroll type, with multiple airstream burners. Inner liner of step-wall construction incorporates first-stage turbine nozzles.

HP Turbine

Single stage with air-cooled rotor blades carried in large flat disc.

LP Turbine

Two stages with rotor blades carried in rings driving via conical disc.

Fan Duct

Integrally ribbed metal, split into upper/lower halves.

Afterburner

High-intensity, with one ring of multiple radial spray tubes in core air flow upstream of two flameholder gutters, plus one ring of nozzles upstream of a gutter in the bypass flow. Corrugated liner laser drilled for cooling air.

Nozzle

Hydraulically actuated multiflap type with inner petals forming convergent/divergent profile when fully open. In 1990, an F110-129 IPE was successfully tested with an Axisymmetric Vectoring Exhaust Nozzle (AVEN) and a similar nozzle undertook outstanding flight trials on an F-16 from 2 July 1993. The AVEN vectors 20º in any direction at up to 60º/s, generating side forces up to 2,948 kg (6,500 lb).

Accessories

Grouped along underside of fan case and duct, driven via bevel gear and tower shaft from front of HP shaft.

Dimensions

Length:	
F110-GE-100	4,630 mm (182.3 in)
F110-GE-400	5,900 mm (232.3 in)
F110-GE-129	4,620 mm (181.9 in)
Diameter (all)	1,180 mm (46.5 in)

Weight, Dry

F110-GE-100	1,492 kg (3,289 lb)
F110-GE-400	1,599 kg (3,525 lb)
F110-GE-129	1,791 kg (3,940 lb)

Performance Ratings (S/L)

Max dry (MIL):	
F110-GE-100	78.06 kN (17,530 lb st)
F110-GE-400	71.60 kN (16,080 lb st)
F110-GF-129	75.7 kN (17,000 lb st)
Max a/b:	
F110-GE-100	124.6 kN (28,000 lb st)
F110-GE-400	119.3 kN (26,800 lb st)
F110-GE-129	129.0 kN (29,000 lb st)
F110-GE-129EFE	151.4 kN (34,000 lb st)

Specific Fuel Consumption (F110-GE-129)

Max dry	18.13 mg/Ns (0.64 lb/h/lb st)
Max a/b	48.16 mg/Ns (1.74 lb/h/lb st)

Contract Price

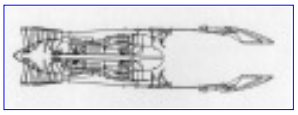
April 1998, F110-GE-129, face-value increase, four engines, US$18,331,268; December 1999, 32

F110-GE-100B engines, US$105,285,304; March 2000, F110-GE-132 engines for 80 F-16 Block 60, ``more than US$400 million".

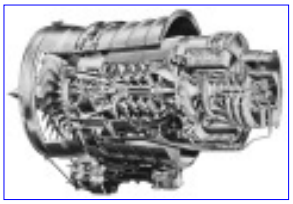
UPDATED

F110-GE-100

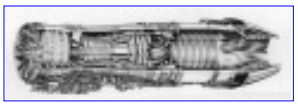
F110-GE-400



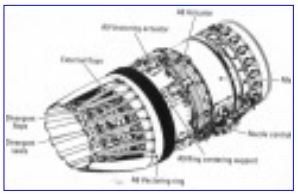
Longitudinal section through F110-GE-100



F110-GE-129



Cutaway drawing of F110-GE-129



Major components of F110 AVEN

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 04 July 2001

Jane's Aero-Engines 10

ENGINE ALLIANCE

ENGINE ALLIANCE GP7000

In May 1998, having previously signed an MoU for the RR Trent 900, Airbus Industrie concluded a similar agreement with Engine Alliance for an engine to power the A3XX (now launched as the A380). It reflected the wishes of several potential customers in thus being able to offer a choice of propulsion.

In 1998, following discussions with Boeing, the Alliance began tailoring a second family of engines, with designations beginning '71' to suit propulsion requirements of the various proposed growth 747s. By 1999 these studies had firmed up on the **GP7167**, to be rated at 297.9 kN (67,000 lb st). This engine would have had a 2,565-mm (101-in) fan, three-stage LPC, and four-stage LP turbine. Dry weight was estimated at 5,126 kg (11,300 lb). On 29 March 2001 Boeing announced that it was abandoning plans to produce a 747X, and was also cutting back on the development of the longer-ranged version of the 767-400ER, but the GP7167 could be resurrected for any application requiring an all-new engine in this thrust class.

GP7200

This is the family designation of engines proposed for the A380. In mid-1999 development was proceeding at two thrust levels. Natural weight growth had by 2001 caused the Alliance to add a third thrust level, and these are:

GP7267

T-O rating 297.9 kN (67,000 lb st) to 30ºC (86ºF). Proposed for A380-100 at initial weights.

GP7270

T-O rating 320.12 kN (72,000 lb st).

GP7275

T-O rating 333.46 kN (75,000 lb st) to 30ºC (86ºF). Proposed for A380-100R and A380-200.

GP7277

T-O rating 342.35 kN (77,000 lb st) to 30ºC (86ºF). Proposed for increased-weight A380 versions.

In addition to the above four thrust levels, the Alliance will certificate a further upgraded version, with a thrust of 362.36 kN (81,500 lb st). If required, this will be offered, probably as the **GP7282**. It should be emphasized that, as at present (mid-2001) conceived, all GP7200 engines will be physically similar.

Recognising the severe competition of the Trent 900, the partners are making exceptional efforts to beat the British engine on cruise fuel efficiency, and if possible even on weight - an area where the Trent family have previously proved superior to the existing US engines. Although President Bruce Hughes insists ``The GP7200 is a derivative, not an all-new engine like the GE90", the Alliance is at the same time investigating any way it can find to reduce weight. One novel idea is to use frangible fan bearing housings so that, following fan-blade failure, the housing breaks instead of transmitting severe off-balance forces to the structure. Such an arrangement would permit reduction in the weight of the mounting pylon, but (to the Editor) would appear to exacerbate the problem of containment.

The original programme plan for the A3XX engine was: selection of final configuration, start of 1999; start of detail design 1999-2000; start of core testing, October 1999; FETT, June 2001; start of flight testing 2001-2002; FAR Pt 33 certification, November 2002; start of A380 flight test with GP7200, spring 2003; EIS, July to August 2004.

In early 1999 President Hughes said the start of core-engine testing was regarded as ``firm, regardless of any slippage in the A3XX program". Testing of a core rig (mainly the responsibility of GE) was still on schedule for October 1999, but the remainder of the programme has slipped. By mid-1999 the programme milestones had been amended to: Initial A380 commercial proposals, December 2000; start of GP7200 detailed design, May 2001; FETT, July 2002; flying testbed, January 2003; GP7200 certification, November 2003; A380 first flight with GP7200, April 2004; A380 certification, June 2005; EIS, July 2005. See later for actual programme.

The first core was assembled at General Electric at Evendale and went on test at the beginning of March 2000. By mid-March about 70 hours had been run, mainly on performance evaluation and the scheduling of the variable-stator system. This preliminary core testing was completed by the end of May 2000.

The Alliance intends to offer a succession of packages introducing new technology ``to enhance customer value". In June 1999 the first, called Phase I, featured ``low risk, high confidence" items intended to reduce TSFC by about 1 per cent, and weight by ``a substantial amount". A Phase II package could be committed by engine launch, and a Phase III package would bring in post-EIS technologies.

In December 1999 the Airbus supervisory board announced that it was satisfied with the A3XX

technical specification, and authorised the management to begin discussions with airlines on numbers and timing, with a view to EIS in 2005. The Engine Alliance at once issued the following statement:

``The Engine Alliance is delighted with the ongoing progress of the A3XX. Fully integrated with the A3XX development program, the Alliance is following a development and certification schedule which supports the aircraft's 2005 entry into service data.

``For the past 12 months the Alliance has been infusing additional technology and improved performance features into the GP7200 design to assure it meets or exceeds the A3XX requirement in terms of performance, weight, noise and emissions. In addition to an engine core test in Cincinnati (Evendale) in the first quarter of 2000, the Alliance is conducting numerous component development tests which will lead to the more detailed design work tailored to meet the aircraft's specific requirements".

On 19 December 2000 the A3XX, having met the criteria stipulated by Airbus management for a go-ahead with 50 orders and 42 options from six customers, was formally launched as the A380. Several further orders have been announced since. By this time the GP7200 engine had undergone significant changes, strongly influenced by progress with the GE90. These changes include an increase in fan diameter from 2,794 mm (110 in) to the figure given below, an additional LPC stage and an additional HPT stage to a total of six. These changes may be expected to have significantly increased the engine length over the previously published figure of 4,547 mm (179 in) and the dry weight over the January 2000 figure for the bare engine of 5,854 kg (12,906 lb).

The following is based on a release issued by the Alliance partners on 30 April 2001:

``Development remains on schedule, with the successful completion of the second full-scale core engine test. In addition to verifying the GP7000 core performance for the A380, this test supported the GE90-115B engine, which shares a common core architecture and aerodynamic technology level with the GP7000. The core, which features an advanced nine-stage high-pressure compressor (HPC), represents the latest configuration in GE's strategic continuous improvement plan for an HPC architecture which is common for the GP7000 and the GE90-115B.

``The GP7000 is benefiting from a heritage of development and verification testing which began with Build 1A, and has progressed to the current Build 6. The latest core build ran for more than 160 hours at GE's Evendale, Ohio, altitude-simulation facility. The tests logged approximately 1,000 data points from more than 1,000 pieces of instrumentation to validate compressor efficiency and operability. The company ran the core Build 5 test in mid-2000, successfully testing several key features, including a swept first-stage HPC airfoil for increased flow and efficiency, and the elimination of the variable vane feature of the stage-4 compressor vane stage which can now be fixed to reduce overall engine complexity and improve cost of ownership.

``'We're very excited about the results of this very important test program,' said Lloyd Thompson, President of the GE-P&W Engine Alliance. 'After we completed the core Build 5 test last year, we felt we could further improve the industry's best compressor. These latest tests confirmed those predictions. This core met all of our requirements for efficiency, and exceeded our expectations for operability.

``'The world-class performance of the GP7000 core has been validated through this testing, giving high confidence that the GP7000 will meet or exceed all performance requirements for the A380 aircraft. Right now, we are the only engine manufacturer testing critical components for the Airbus A380; we're ensuring that we give our customers mature technology at entry into service.'

``Extensive GP7200 component tests are ongoing, with Core Build 7 scheduled to go on test in early 2002 as part of the GP7000 strategy continually to improve performance and mature technology. The company will begin the detailed engine-design phase in early 2003, and will run the first Full Engine Test in April 2004. The GP7200 is scheduled to be certificated at 81,500 lb st [see above] by FAA/JAR

in July 2005. First flight on the A380 is scheduled for January 2006, followed by EIS later in that year."

Though the GP7200 is beyond question a superb engine, the Alliance partners had to watch a succession of A380 customers select the competitor engine. All that changed on 29 May 2001, when Air France picked the GP7270 as the powerplant of ten firm A380-800 aircraft for delivery from fourth quarter 2006. In announcing this, the Alliance also revealed that Snecma of France and MTU of Germany are revenue-sharing partners (it did not say ``risk-sharing'') in the GP7200 programme. Within the Alliance itself, GE Aircraft Engines is responsible for the complete core (HPC, combustor and HPT) and control system, while Pratt & Whitney is responsible for the fan, LPC, LPT, gear train and accessory gearbox. GE will assemble cores at Durham, North Carolina, sending them to Pratt & Whitney for assembly and test at Middletown, Connecticut.

The following data refer to the GP7200 at EIS:

Type

Two-shaft turbofan of high BPR.

Fan

Originally derived from PW4084/PW4090. Diameter 2,946 mm (116 in). Single-stage, with wide-chord swept-back snubberless blades of hollow titanium. In May 2001 it was announced that Pratt & Whitney Canada had successfully completed testing of a 42-per-cent-scaled fan. This testing was said to validate ``performance improvements of the 3-D aerodynamic design''. The cutaway drawings show the progress towards the final shape of blade. From the outset it has been intended that the blade tips should be rebated into the case, but the latest drawings no longer show the case as being of lightweight sandwich construction; it will be an aluminium isogrid fabrication. Instead of originally schemed dual forward bearings, the fan has a single high-capacity ball bearing, see remarks above about frangibility. The mass flow was published in 2000 as 1,179 kg (2,600 lb)/s, when fan diameter was 2,794 mm (110 in). The actual figure is now likely to be closer to 1,361 kg (3,000 lb)/s. Cruise BPR, originally set at 8, has now matured at 8.7, the highest for any airline engine.

LP Compressor

The GP7200 was originally conceived with a two-stage core booster, rotating with the fan, the first stage rotor having aerofoils perpendicular to the engine axis. By 2000 the GP7200 had a four-stage LPC, with controlled-diffusion aerofoils of low aspect ratio derived from PW4084/4090 technology, all with pronounced backwards lean. By 2001 a fifth stage had been added, though OPC had not changed from the previously published value (see below).

HP Compressor

Like the entire core, a 72 per cent scale from GE90, with GE/NASA E^3 features. Nine stages (originally 10) with variable IGVs and next four stators. Spool PR 19. OPR (max climb) 43.9.

Combustion Chamber

The original GP7176 would have used GE90 technology with dual-annular burners, but in 1997 this was replaced by a single annular type derived from that of the latest CF6 engines. All air blasts in through a divergent nozzle ring aimed at the single ring of burners. Thermal barrier coatings. An announced design objective is to reduce emissions to 40 per cent below the standards set by ICAO in 1998.

HP Turbine

Two stages, scaled from GE90 at same scale factor as the HPC. Advanced aircooled blades with thermal barrier coatings.

LP Turbine

Originally five stages, now six. Derived from PW4056/4152. Efficient high-slope design, with active clearance control managed by FADEC.

Reverser

In fan duct only.

Mounting

Pylon attachments on rear of fan case and top of LP turbine case.

Accessories

On underside of core, driven from front of HPC. The drive shaft, originally inclined back, is now to be radial.

Control System

BAE Systems Fadec III, with improved diagnostics.

Dimensions

Diameter	3,150 mm (124 in)
Length	4,750 mm (187 in)

Performance Ratings

T-O, flat-rated to 30ºC (86ºF)	311.23 kN (70,000 lb st)
Cruise (10,670 m; 35,000 ft, M0.85, ISA)	56.19 kN (12,633 lb)

Specific Fuel Consumption

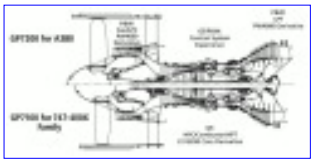
10% better than 747-400.

Contract Price

May 2001, engines for ten A380-800, ``nearly US$900 million".

UPDATED

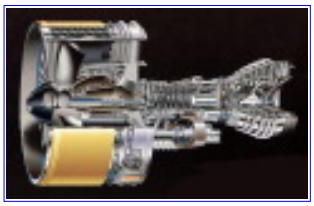
Half longitudinal section of GP7176 of 1996-97 (1999)



Half longitudinal sections of GP7275 (upper), GP7167 (lower) (2000)

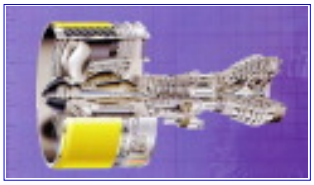
Cutaway drawing of GP7200 propulsion system as it was in 1999 (1999)

Cutaway drawing of GP7275 (January 2000) (2000)



Cutaway drawing of GP7200 (December 2000) (2000)

Preparation of GP7200 core for testing (2000)



Cutaway drawing of GP7200 (June 2001) (2001)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 04 July 2001

Jane's Aero-Engines 10

GE AIRCRAFT ENGINES

GENERAL ELECTRIC CF6-80G2

Announced in April 1998, this advanced version of the CF6 family is intended to be certificated initially at about 302.4 kN (68,000 lb st), with considerable growth potential. It is intended to provide a minimum-change fit for the A330-200, 767-400ERX and projected 747-400X. GE states that the G2 engine will not conflict with the GP7000 engine jointly launched with Pratt & Whitney (see under Engine Alliance at the start of the US section), even though it offers exactly the same level of thrust.

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 04 July 2001

Jane's Aero-Engines 10

WILLIAMS - WILLIAMS INTERNATIONAL

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This company was formed in 1955 by Sam Williams, who believed that gas-turbine jet engines could be effective propulsion systems in very small sizes. He established a design office and factory at Walled Lake, which has subsequently grown considerably. Williams gained the major shares of the large markets for cruise-missile propulsion, as well as important contracts to power targets and unmanned reconnaissance vehicles. To produce the various versions of F107 cruise-missile engine a large new plant was built at Ogden, Utah. This remains the principal source for production engines, with capacity for large-scale output.

In 1989, Rolls-Royce joined Williams to produce the FJ44 for manned aircraft. The two partners formed a joint company, Williams Rolls Inc, with offices at Walled Lake.

The following pages are devoted to Williams engines for manned aircraft. Williams engines for unmanned vehicles include the following:

F107-WR-402: turbofan, 3.11 kN (700 lb st) class, Tomahawk cruise missile.

F112-WR-100: turbofan, classified performance, Advanced Cruise Missile.

F121-WR-100: turbojet, 0.67 kN (150 lb st), Tacit Rainbow defence-suppression.

WJ38-15: turbojet, 6.67 kN (1,500 lb st), future stand-off weapons.

WJ119 (IPM): turbojet, 0.445 kN (100 lb st) Integrated Propulsion Module for fibre-optic-guided missiles.

WTS117: turboshaft, 93.25 kW (125 shp), Canadair CL-327 UAV.

Further details of these engines can be found in *Jane's Unmanned Aerial Vehicles and Targets*.

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 04 July 2001

Jane's Aero-Engines 10

GE AIRCRAFT ENGINES

GENERAL ELECTRIC CF6

US military designation (CF6-50E): F103-GE-100

On 11 September 1967 General Electric announced the commitment of corporate funding for development of the CF6 turbofan for the then-forthcoming generation of wide-body transports. From the initial family of 142 to 160 kN (32,000 to 36,000 lb st) CF6 two-shaft engines announced in September 1967 to cover the anticipated thrust requirements of the Lockheed and McDonnell Douglas wide-body projects, the CF6 evolved through a series of variants to the CF6-6D, flat-rated at 178 kN (40,000 lb) to 31ºC and tailored to the DC-10 Series 10 intermediate-range transport. An announcement that this engine had been selected by United Air Lines and American Airlines was made on 25 April 1968.

Basic configuration of the CF6-6 comprises a 1¼-stage fan driven by a five-stage LP turbine, energised by a slightly modified TF39 core engine, consisting of a 16-stage HP compressor, annular combustor and two-stage turbine. Modifications were introduced to enable the accessory systems to suit airline installation requirements, while other changes were aimed at enhancing reliability, durability and maintainability.

CF6-6D

Initial 178 kN (40,000 lb st) version of the engine in production for intermediate range DC-10 Series 10.

First ran on 21 October 1968 and 18 days later attained 203.5 kN (45,750 lb st). Following a series of successful factory and outdoor tests, the engine was released for production in February 1969. The second CF6-6D, built to the production configuration, first ran in May 1969. Delivery of flight test engines to McDonnell Douglas started in late 1969, with aircraft first flight following in September 1970. Certification of the CF6-6D for commercial service was granted by the FAA in September 1970, and the engine entered airline service in the DC-10 Series 10 in August 1971.

CF6-6D1

In August 1971 this growth version was FAA certificated and offered to take advantage of the demonstrated margin of the -6D. The D1 rating was increased by 1,000 lb to 182.4 kN (41,000 lb st) at 28.9ºC. By 1979 more than 400 -6D and -6D1 engines had been delivered. At 30 April 1997 these engines were still operating in 87 DC-10-10 aircraft. Total flight time was 24.1 million hours, the high-time engine having logged 66,705 hours and the high-cycle engine 25,181 cycles. Engine-caused SVR was 0.236 (one per 4,237 hours), and dispatch reliability 99.99 per cent.

CF6-6K

This version of the -6D1, rated at 184.6 kN (41,500 lb st) at 31.1ºC, was planned for certification in June 1981. It featured improved sfc, improved performance retention and further improved reliability, and was planned for applications currently using CF6-6D1 engines. Mechanical changes included improved-performance fan, -50 type HP turbine and external-case LP turbine case cooling.

CF6-45A and -45B

Economical derated CF6-50E (described later) giving flat rating of 206.8 kN (46,500 lb st) to 31.6ºC (-45A for 747SR) or 43.3ºC (-45B for 747SP).

CF6-50A

Announced by GE in January 1969, the 218 kN (49,000 lb st) CF6-50A was a growth version of the CF6-6. The increased thrust is achieved by increased flow through the core engine (reducing the bypass ratio from 5.9 to 4.4) at slightly decreased turbine entry temperature. A major change was the introduction of two additional booster stages behind the fan, with no change in the external dimensions. To provide for flow matching between the two rotors, variable bypass doors are incorporated between the LP and HP compressors. A 41 per cent scale model fan with three-stage compressor and variable bypass doors started testing in January 1969. FAA certification testing was completed in March 1972. The CF6-50A entered airline service in December 1972 in the DC-10 Series 30. The CF6-50 series also powers the Airbus A300 and some versions of Boeing 747, including the E-4B.

CF6-50C

The CF6-50C is rated at 226.8 kN (51,000 lb st) up to 30ºC. Higher thrust is provided by an increase in turbine temperature, with improved cooling of hot-section components. Certificated November 1973.

CF6-50C1/E1

Rated 233.5 kN (52,500 lb st) to 30ºC.

CF6-50C2/E2

Similar to C1 and E1 but improved sfc and EGT margins and new fan case and blades with improved birdstrike resistance. Certification 1978.

CF6-50E

(Military designation **F103-GE-100 and -101**). This engine is rated to give 233.5 kN (52,500 lb st) up to 26ºC. Certificated November 1973. Powers some versions of Boeing 747-200, E-4B and USAF KC-10A. By 30 April 1997, all types of CF6-50 engine were operating in a total of 551 aircraft and had flown 87 million hours. The high-time engine had logged 84,056 hours, and the high-cycle engine 28,877 cycles. Engine-caused SVR was 0.166 (one per 6,024 hours) and dispatch reliability 99.92 per cent. The F103-GE-100/101 had flown 2.1 million hours, with the high-time engine on 12,960 hours; unscheduled SVR was 0.193 per 1,000 hours.

In October 1996 the Russian Aviastar industrial complex at Ulyanovsk announced that it was preparing an Antonov An-124 heavy airlifter for Western markets. One of the modifications was to fit CF6 engines. At the time of writing it was not known which version of the CF6 was involved. Meanwhile, Antonov itself, at Kiev, Ukraine, is independently modifying an An-124 for export markets.

With maturity, and with millions of hours flown, CF6 engines suffered uncontained break-ups from 1991 onwards. The first affected the CF6-50 of a Korean A300B; the titanium-alloy HP spool began to break up due to what FAA inspectors called ``dwell-time fatigue, a new phenomenon''. According to the FAA, ``The cracks propagate much faster than we've ever seen before''. In 1993 one of the two CF6-80 engines of an American Airlines 767 suffered uncontained HP spool break-up. A near-catastrophe occurred on 25 April 2000 when one CF6-6D of a fully-loaded Continental DC-10 suffered an uncontained break-up on take-off from Newark, also knocking out the right engine; the pilots flew for 34 minutes, dumping fuel, on the tail engine. It was discovered that the fault in this case was failure of turbine NGV (stator) locking pins, allowing the complete nozzle ring to rotate at high speed, wear through the casing and disintegrate. In June 2000 an HP spool again broke up on a 767 of Varig, and in September 2000 Continental suffered NGV pin failure on two DC-10s taking off from Newark and Amsterdam, and a third during line maintenance. The FAA and GE moved swiftly to find corrective modifications, Roger Seager, GE's CF6 Project Manager, saying, ``I can't afford another engine failure''.

CF6-80

New family of engines, described later.

The following data relate to the CF6-6D, with the differing features of the CF6-50 series also detailed.

Type

Two-shaft high-bypass ratio turbofan.

Intake

Single forward-facing annular configuration.

Fan

Single-stage fan with integrally mounted single-stage LP compressor (described together as a 1¼-stage

fan), both driven by LP turbine. Fan has rotating spinner and omits inlet guide vanes. Blade-containment shroud provided against possible blade failure. The 38 fan rotor blades are individually removable from the thick-section disc bolted to forward conical extension of LP shaft system. Blade aerofoil has anti-vibration shrouds at two-thirds span. Fan exit air flow split between LP compressor and fan slipstream. Fan frame has 12 radial struts across slipstream exit. Fan frame provides support for LP and HP rotor front bearings, fan being overhung ahead of large-diameter ball-thrust bearing with rear roller bearing ahead of core engine. Blades, discs, spool of titanium; exit guide vanes of aluminium; fan frame and shaft of steel; spinner and fan case of aluminium alloy. Mass flow (6D) 593 kg (1,307 lb)/s, (6D1) 600 kg (1,323 lb)/s. Bypass ratio (6D, 6D1) 5.9. CF6-45 and CF6-50 have two added LP core-booster stages with bypass doors (see LP compressor). Mass flow (45A, 45B) 630 kg (1,389 lb)/s, (50A) 653 kg (1,439 lb)/s, (50C) 664.5 kg (1,465 lb)/s, (50C1, E, E1) 673 kg (1,484 lb)/s. Bypass ratio (all) 4.4.

LP Compressor

Single-stage compressor, acting as booster to air flow into core engine. Rotor blades carried on rear rim of tapered drum bolted to rear of fan disc. Stators cantilevered off short-chord shroud ring, supported by radial outer struts and radial/tangential inner struts located on fan front frame. Compressor exit flow free to balance between core engine and fan slipstream exit. Configuration of CF6-50 modified to three compressor booster stages carried on flanged rotor drum. Continuous shroud extends to fan front frame with 12 integral bypass doors located between canted radial struts in fan exit inner casing. These doors maintain proper flow matching between the fan/LP system and core by opening at low power settings to permit LP supercharged flow to bleed into the fan airstream. The doors are closed during take-off and cruise.

HP Compressor

Sixteen-stage compressor of near-constant tip diameter, with inlet guide vanes and first six stator rows having variable incidence. Provision for interstage air bleed for airframe use and engine cooling. Rotor of combined drum-and-disc construction with front stage and rear three stages overhung on conical shaft providing location on HP front bearing and HP main shaft. All rotor blades held in rabbeted discs and individually replaceable without rotor disassembly. Stages 1-7 blades forged titanium, 8-16 steel. Stages 1-10 disc titanium, 11-16 and aft casing Inconel 718. Casing split on horizontal centreline: stator vanes held in dovetail slots and replaceable individually. All stator vanes steel. Double-skin inner casing shrouds the LP main shaft. Outlet frame contains compressor diffuser and incorporates support structure for HP rotor mid-bearings. Overall pressure ratio (T-O), 24.3 (6D), 24.9 (6D1). Core air flow (-6D) 88 kg (194 lb)/s. CF6-50A has 15th and 16th stages removed to pass greater core air flow of 125 kg (276 lb)/s and reduce pressure and temperature of air entering combustion chamber. Titanium blades in stages 1-5, steel 6-14. Overall pressure ratio (6D) 24.2, (6D1) 24.7, (45A, 45B) 27.1, (50A) 28.6, (50C) 29.5, (50C1, E, E1) 30.3.

Combustion Chamber

Fully annular with comprehensive film cooling. Separate snout, dome and inner/outer skirts, with nozzles, igniter, leads and manifold externally removable. Dome contains ports for two igniters and axial swirler cups for 30 fuel nozzles. Igniters of high-voltage surface-gap type with energy level of 2.0 J, each igniter operated independently. Forced steel nozzles with liner of Hastelloy X. Nozzle and dome designed to minimise smoke, and entrance diffuser has gradual profile to assure low temperature gradient to turbine under all flight conditions. CF6-50 combustor is shorter, of improved material (HS 18-8), and can be removed with fuel nozzles in place.

HP Turbine

Two-stage air-cooled turbine with 1,330ºC entry temperature. Rotor blades are film and convection cooled. Rotor blades cast from René 80; discs and forward and rear shafts of Inconel 178. First-stage nozzle guide vanes supported at inner and outer ends; second-stage cantilevered from outer ends, with inner ends carrying interstage labyrinth seals. First-stage vanes cast from X40 and film cooled by compressor discharge pressure. Second-stage vanes are cast from René 80 material and are convection cooled. Vanes are welded into pairs to decrease number of gas leakage paths. Thin-section discs with heavy-section centreless hubs are bolted to front and rear conical shafts, including conical and arched inter-disc diaphragms. Configuration for CF6-50 is similar but introduces improved materials and cooling, and blades are not Siamesed but individual. Stage-1 rotor blades being improved by cast-in tips and internal aluminide coating.

LP Turbine

Five-stage constant tip-diameter turbine with nominal 871ºC inlet temperature. Rotor blades tip-shrouded and cast in René 77, not air-cooled. Forward and rear shafts, case and discs of Inconel 718. First-stage nozzle guide vanes supported at inner and outer ends; remaining stages are cantilevered from outer ends with inner ends carrying inter-stage labyrinth seals. Stages 1-3 guide vanes cast in six-vane segments in René 77, stages 4 and 5 cast in pairs in René 41. Vanes held in slots machined in the two half-stator casings. Drum and centreless disc construction, located on LP rotor by front and rear conical shafts attached to third- and fourth-stage discs. Drive to rotor by means of long fan midshaft. On CF6-50 a four-stage LP turbine is used, all stages being modified in geometry and cooled by seventh HP-stage compressor air instead of ninth.

Jetpipe

Fixed-area annular fan duct with outer cowl and engine cowl forming convergent plug nozzle for fan slipstream. Short-length fixed-area core jetpipe with convergent plug nozzle. Provision for reverser.

Reverser (Fan)

Annular cascade reverser with blocker doors across fan duct. For reverse thrust, rear portion of fan outer cowl translates aft on rotating ballscrews to uncover vanes. Blocker doors (16) flush-mounted in cowl on link arms hinged in inner cowl, rotate inwards to expose cascade vanes and block fan duct. Reverser hinged at top to open in L/R halves for access to HP casing and combustor.

Reverser (Turbine)

Post nozzle exit, cascade type. Two cascade screens are mounted in vertical plane on fixed pivot aft of turbine exhaust and are enclosed in fairing forming aerofoil-shaped plug. Aft translation of fairing uncovers cascades which open across nozzle exit and divert turbine exhaust radially outward and slightly forward in horizontal plane. Configuration for CF6-50 similar to fan thrust reverser with nine blocker doors, but not split. CF6-50 available with long fixed core nozzle; short nozzle also designed, for performance improvement. Acoustic treatment is provided in the nozzle flow path.

Mounting

Main thrust mount located on the inner fan frame; aft mount located on the turbine mid-frame.

Accessories

This consists of the inlet gearbox, radial gearbox, radial driveshaft, transfer gearbox, horizontal

driveshaft and accessory gearbox. The inlet gearbox is located in the forward sump of the engine. The gearbox transfers energy from the core-engine (HP) rotor to the radial driveshaft located in a housing aft of the bottom vertical strut of the fan frame. The transfer gearbox is mounted on the bottom of the fan frame. Accessory mounting pads are provided on both the forward and aft faces of the gearbox. The engine accessories mounted on the gearbox are starter, fuel pump, main engine control, lubrication pump and tachometer. Pads are also provided for mounting the aircraft hydraulic pumps, constant-speed drive and alternator.

Starting

Air-turbine starter mounted on the front of the accessory gearbox at the through shaft.

Control System

Hydromechanical fuel control system regulates steady-state fuel flow and schedules acceleration and deceleration fuel flow. It also schedules and powers variable-stator vane position. A governor in the Woodward control provides core-engine speed stability during steady-state operation. During transient operation, core-engine fuel flow is scheduled on the basis of throttle position, compressor inlet temperature, compressor discharge pressure and core-engine speed. The fuel control and fuel pump are mounted in the accessory package as an integrated unit which avoids interconnecting high-pressure fuel lines and potential leakage points (they are separable for change or maintenance). This configuration provides a single drive mounting flange. The filter, fuel/oil heat exchanger and control pressurising valves may be removed individually without removing the entire assembly. The fuel manifold is double-wall constructed for safety and mounted on the exterior of the engine. For CF6-50, fuel control is modified to provide scheduling function for LP compressor variable bypass doors.

Fuel Specification

Fuels conforming to ASTM-1655-65T, Jet A, Jet A1 and Jet B, and MIL-T-5624G2 grades JP-4 or JP-5 are authorised, but Jet A is primary specification.

Oil System

Dry-sump centre-vented system in which oil is pressure-fed to each engine component requiring lubrication. Oil is removed from the sump areas by scavenge pumps, passed through a fuel/oil heat exchanger and filter to the engine tank. Nominal lubrication system pressure is 2.07 to 6.21 bars (30 to 90 lb/sq in) above sump reference pressure. All pressure and scavenge pumps and filters are located in the lubrication centre on the forward side of the gearbox.

Oil Specification

Conforming to General Electric specification D50TFI classes A and B, equivalent to MIL-L-7808 or MIL-L-23699A.

Dimensions

Fan tip diameter	2,195 mm (86.4 in)
Fan case diameter	2,337 mm (92.0 in)
Max width (cold)	2,390 mm (94.1 in)
Max height (over gearbox)	2,675 mm (105.3 in)
Length overall (cold):	
CF6-6D	4,775 mm (188.0 in)

CF6-50 series	4,648 mm (183.0 in)

Weight, Dry

Basic engine:	
CF6-6D, -6D1	3,679 kg (8,111 lb)
CF6-45A/B, -50E, -50E1	3,851 kg (8,490 lb)
CF6-50A, -50C, -50C1	3,956 kg (8,721 lb)
Fan and turbine reverser:	
CF6-6D, -6D1	932 kg (2,054 lb)
CF6-50A, -50C	968 kg (2,135 lb)
CF6-50E	962 kg (2,121 lb)

Performance Ratings

Max T-O, uninstalled, ideal nozzle: See under model listings	
Max altitude and Mach No:	
CF6-6 and -50	13,700 m (45,000 ft) at M1.0
Max cruise thrust at 10,670 m (35,000 ft) and M0.85, flat rated to ISA, + 10ºC, uninstalled, real nozzle:	
CF6-6D	40.6 kN (9,120 lb st)
CF6-6D1	41.1 kN (9,250 lb st)
CF6-45A	48.9 kN (11,000 lb st)
CF6-50A, -50C	48.0 kN (10,800 lb st)
CF6-50E, -45B	50.3 kN (11,300 lb st)

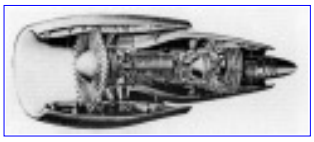
Specific Fuel Consumption

At T-O thrust, as above:	
CF6-6D	9.86 mg/Ns (0.348 lb/h/lb st)
CF6-6D1	9.91 mg/Ns (0.350 lb/h/lb st)
CF6-45A, -45B	10.28 mg/Ns (0.363 lb/h/lb st)
CF6-50A	10.90 mg/Ns (0.385 lb/h/lb st)
CF6-50C	11.05 mg/Ns (0.390 lb/h/lb st)
CF6-50E	10.65 mg/Ns (0.376 lb/h/lb st)
Oil consumption:	0.9 kg (2.0 lb)/h

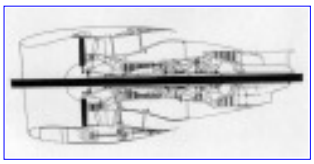
UPDATED

Vertical assembly of CF6-50 engines

Cutaway of CF6-6D nacelle



Cutaway of CF6-50 nacelle



Longitudinal sections through CF6-6 (upper half) and CF6-50

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 01 May 2001

Jane's Aero-Engines 10

GE/P&W - GE AIRCRAFT ENGINES

Turboshaft Engine Company, Lynn, Massachusetts 01905

Vice-President and General Manager, Military Engines: Russ Sparks

UNITED TECHNOLOGIES

Pratt & Whitney

Small Military Engines, West Palm Beach, Florida 33410

President, P&W Military Engines: Steve Finger

On 29 March 2000 GE and UTC signed a Memorandum of Understanding for a new helicopter engine which will compete for the US Army's CEP (Common Engine Program). This engine is required in the first instance to replace the T700 in future versions of the Black Hawk/Seahawk and Apache helicopters, and sebsequently in future designs.

CEP goals include an improvement of 60 per cent in O&S (operation and support) costs. There is no way these objectives can be met by an improved version of any existing engine. The partners began pre-development work almost immediately, in anticipation of an Army CEP competition. The venture is `expected to reflect a 60:40 GE/P&W ownership and work-split structure for the CEP'.

By 2001, problems involving GE's relationship with not only P&W but also Honeywell had caused the CEP to be put on ice. The partners await a decision by the new Administration.

UPDATED

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AERO-ENGINES - TURBOFAN/TURBOJET, UNITED STATES OF AMERICA

Date Posted: 09 January 2001

Jane's Aero-Engines 09

UNITED TECHNOLOGIES PRATT & WHITNEY

P&W/TCM

In May 2000, it was announced that Pratt & Whitney Small Military Engines group and Teledyne Continental Motors had signed an agreement ``to discuss teaming arrangements covering the development, manufacture and support of small gas turbines". Such an agreement could extend to engines for all kinds of surface applications, but initially it is intended to cover jet engines of up to 71.1 kN (16,000 lb st) for unmanned military aircraft.

In 1998, United Technologies Pratt & Whitney formed the Small Military Engine group in order to recover lost ground resulting from failure to follow such engines as the JT12. The group's General Manager, Kevin Farrell, said ``We want to put our logo on all systems, manned and unmanned, from 50 lb thrust upwards". It was expected that a further statement would be made by the fall (autumn) of 2000.

Pratt had previously entered into a teaming agreement with General Electric Aircraft Engines (see GE/P&W), covering turboshaft engines.

NEW ENTRY

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AERO-ENGINES - TURBOPROP/TURBOSHAFT, UNITED STATES OF AMERICA

Date Posted: 09 January 2001

Jane's Aero-Engines 09

GE AIRCRAFT ENGINES

GENERAL ELECTRIC CT7

The designation CT7 covers a family of turboprop and turboshaft engines, all derived from the military T700. In general, the description of the T700 applies, except for elimination of military/naval features, a different group of accessories and, in the turboprop, addition of a remote reduction gearbox. In April 1999, GE delivered the 11,300th engine in the combined T700/CT7 family. At that time, these engines had logged more than 30 million flight hours in 24 types of aeroplanes and helicopters, including more than 14 million in airline service.

Turboshaft versions

Development has been continuous since the first version, the -2A, was certificated in April 1977. The following versions have been produced:

CT7-2A

Powers Bell 214ST. EIS June 1982.

CT7-2B

Formerly powered Westland 30-200 and -300; no longer operating.

CT7-2D

Commercial equivalent of T700-701 and 401. Powers Sikorsky S-70.

CT7-2D1

Equivalent of T700-701C and 401C. Certificated June 1989. Powers S-70. It is intended that the same engine produced by Rybinsk-GE (see under Russia) shall be an option on the Kamov Ka-64.

CT7-6

Step-3 growth engine co-developed by GE with Alfa Romeo Avio and FiatAvio (about 25 per cent each) for EH 101. Delivered from April 1988.

CT7-6A

Marinised version equivalent to T700/T6A for EH 101 naval versions. CT7-6 and -6A are FAA/RAI/BCAA certificated. Deliveries began January 1997. Powered all 37,200 hours of EH 101 development test, including October 1999 flight to Canada by PP9. At this time, PP9 and PP8 had completed 4,000 hours of an intensive programme with no IFSD.

CT7-6D

Commercial version originally intended for Sikorsky S-92 Helibus. Now fitted only to the first S-92 prototype, which began flight testing in January 1999.

CT7-7E

Latest growth version, with FADEC, 3-D flow path, advanced materials and improved cooling. Exceeded 2,013 kW (2,700 shp) during first 10 hours of testing and 2,240 kW (3,000 shp) subsequently. Targeted for FAA certification 1999.

CT7-8

Commercial version with FADEC and inlet particle separator, for Sikorsky S-92 Helibus. Developed jointly with FiatAvio. In April 1999 this engine was undergoing FAA certification testing, including water, ice and foreign-object ingestion testing delegated to the Canadian National Research Council in Ottawa, Ontario. This engine will power the Nos 2, 3 and 4 prototypes and all S-92 production helicopters. First flight in testbed May 1999. Medium-bird test passed September 1999, certification intended at the end of that year. First flight of an S-92 with CT7-8 engines 30 October 1999. On 24 January 2000 first 40 hours in S-92 said to have `exceeded expectations'. FAR Pt 33 certification finally awarded October 2000, to permit delivery of production engines to begin in mid-2001.

CT7-11

Projected further growth version in 2,240 kW (3,000 shp) class.

By 31 July 1997, CT7 turboshaft engines were in service powering 68 helicopters. They had then

flown 566,600 hours, the high-time engine having recorded over 11,900 hours. Mean time between engined-caused shop visits was approximately 4,000 hours.

The detailed description of the T700 (see later) applies in general to the CT7 turboshaft versions. The following are major differences:

Dimensions

Length	1,194 mm (47.0 in)

Diameter:	
CT7-2, -8	635 mm (25.0 in)
CT7-6D	660 mm (26.0 in)

Weight, Dry

CT7-2A	194.6 kg (429 lb)
CT7-2D	201 kg (442 lb)
CT7-2D1	212 kg (466 lb)
CT7-6, -6A	220 kg (485 lb)
CT7-6D	229 kg (504 lb)
CT7-8	not finalised

Performance Ratings
(S/L, ISA):

CT7-2A, -2D and -2D1:	
2.5 min OEI	1,286 kW (1,725 shp)
30 min OEI and T-O	1,212 kW (1,625 shp)
Continuous	1,189 kW (1,595 shp)
CT7-6 and -6A:	
2.5 min and continuous OEI and T-O	1,491 kW (2,000 shp)
Continuous	1,281 kW (1,718 shp)
CT7-6D:	
30 s OEI	1,734 kW (2,325 shp)
2 min OEI	1,637 kW (2,195 shp)
OEI continuous	1,570 kW (2,105 shp)
T-O	1,514 kW (2,030 shp)
Continuous	1,336 kW (1,792 shp)
CT7-8:	
T-O	1,879 kW (2,520 shp)
OEI continuous	1,789 kW (2,400 shp)

Specific Fuel Consumption

(continuous rating, S/L, ISA):

CT7-2A, -2D and -2D1	79.9 Ág/J (0.473 lb/h/shp)
CT7-6 and -6A	79.4 Ág/J (0.470 lb/h/shp)
CT7-6D	76.9 Ág/J (0.455 lb/h/shp)
CT7-8	79.06 Ág/J (0.468 lb/h/shp)

Turboprop versions
All turboprop versions in current use have a remote propeller gearbox carried on struts ahead of the inlet and above the inlet air duct (except for the first version listed below which has not been put into production):

CT7-3

Projected version with Hamilton Standard gearbox centred in an enlarged annular air inlet.

CT7-5A

Certificated August 1983. Produced as 5A1 and 5A2 to power initial Saab 340. EIS June 1984.

CT7-7A

Similar to -5A. Powers initial series Airtech CN-235.

CT7-9B

Growth engine, with improved aerodynamics, materials and turbine cooling. Powers Saab 340, driving Dowty R390 or Hamilton 14RF-21 slow-running four-blade propeller. Option on Sukhoi S-80.

CT7-9C

Powers Airtech CN-235, driving Hamilton Standard 14RF-21 four-blade propeller.

CT7-9C3

Hot/high growth derivative incorporating improved centrifugal compressor and stronger drive shaft to new Hamilton Standard 3.66 m (12 ft) four-blade propeller. Certificated 29 July 1998. Powers CN-235-300 and -330.

CT7-9D2

Powers Let L-610G, driving Hamilton Standard 14RF-23 four-blade propeller.

CT7-11

A turboprop version of this growth engine is being studied.
By 31 July 1998, CT7 turboprops were in service powering 598 Saab 340 (production of which has

been terminated) and CN-235 aircraft. At that time, engine flight hours were 14.1 million, the high-time engine having logged 26,370 hours and the high-cycle engine 30,155 cycles. Dispatch reliability was 99.99 per cent.

Control System
CT7-5A and -7A, analog hydromechanical; CT7-9 models, DEC/hydromechanical.

Dimensions

Length	2,438 mm (96.0 in)
Height of gas generator	737 mm (29.0 in)

Weight, Dry

CT7-5A, 7A	356.3 kg (783 lb)
CT7-9B, C and D	366.3 kg (805 lb)

Performance Ratings
(S/L, ISA):

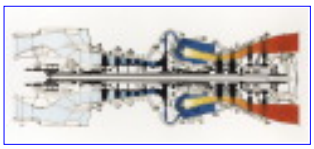
T-O:	
CT7-5A	1,294 kW (1,735 shp)
CT7-7A	1,268 kW (1,700 shp)
CT7-9B, C and D	1,305 kW (1,750 shp)
CT7-9C3	`1,417 kW (1,900 shp) class'
APR:	
CT7-9B, -9C	1,394 kW (1,870 shp)
CT7-9D	1,447 kW (1,940 shp)
Max cruise (4,572 m; 15,000 ft):	
CT7-5A, -7A	978 kW (1,312 shp)
CT7-9B, -9D	1,052 kW (1,411 shp)
CT7-9C	1,118 kW (1,499 shp)

Specific Fuel Consumption
(T-O, as above):

CT7-5A	80.4 Ág/J (0.476 lb/h/shp)
CT7-7A	80.08 Ág/J (0.474 lb/h/shp)
CT7-9B, -9C, -9D	77.9 Ág/J (0.461 lb/h/shp)
CT7-9D	76.87 Ág/J (0.455 lb/h/shp)

Contract Price
According to Rybinsk-GE, a CT7-9 costs `US$750,000 to US$850,000'.

UPDATED



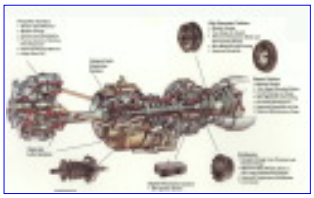
Longitudinal section through typical CT7 gas generator

CT7-5A (see overleaf)

Modules forming CT7-9

General Electric CT7-9 free turbine turboprop

CT7-9B



Cutaway drawing of CT7-9B

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 09 January 2001

Jane's Aero-Engines 09

WILLIAMS INTERNATIONAL

WILLIAMS INTERNATIONAL FJX

This small turbofan is being developed under the GAP (General Aviation Propulsion) programme. A co-operative effort with NASA, this is intended `to revolutionise and revive the once-flourishing American light-aircraft industry'.

Key features will be the lowest possible capital and operating costs, minimal fuel burn, very low noise and emissions and light weight. A basic objective is that, in mass production, this engine should be priced at a level similar to that of the piston engines it is designed to replace (target said to be US$50,000).

In late 2000, a detailed description was still not possible, but the technology (but not any part) is generally similar to that of the company's larger engines.

FJX-1

T-O rating 3.11 kN (700 lb st). Powers second Chichester-Miles Leopard which has been flying since April 1997. This engine will not be produced in quantity.

FJX-2

T-O rating 3.11 kN (700 lb st) class. The first complete engine ran in December 1998. In late 1999, development was on schedule for flight demonstration in the Williams V-JET II at the EAA Oshkosh AirVenture 2000 meeting. This aircraft first flew in 1997 with different interim engines, but has been superseded by the Eclipse 500 (see below). Among many potential customers for this engine is the British Chichester-Miles company, for the third Leopard prototype. Another application is a jet derivative of the Piper Aerostar. Production engines designed for a 7,500-hour TBO could be available in 2003.

EJ-22

T-O rating 3.42 kN (770 lb st). Variant being developed for twin-engined Eclipse 500. Similar to FJX-2 apart from increased ratings. Four development engines had run 200 hours by mid-2000, with certification planned for 2003. Sam Williams is delighted at the rapid progress of Eclipse Aviation, and has terminated the V-Jet II initiative.

Type

Two-shaft turbofan.

Dimensions

Length	1,041 mm (41.0 in)
Diameter	368.3 mm (14.5 in)

Weight, Dry

EJ-22	about 39.55 kg (85 lb)

Performance Rating

See model descriptions	

UPDATED

FJX-2 mockup (2000)

Preliminary general-arrangement drawing of FJX-2 (2000)

FJX-2 (Paul Jackson) (2000)

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AERO-ENGINES - TURBOJET, UNITED STATES OF AMERICA

Date Posted: 09 January 2001

Jane's Aero-Engines 09

GE AIRCRAFT ENGINES

GENERAL ELECTRIC YF120

One of the most advanced jet engines ever flown, this began life as a candidate engine for the Advanced Tactical Fighter (ATF). It was rejected in favour of the rival YF119 and now, five years later, has come back into the picture as a rival to the YF119-derived main engine of the various JSF (JAST) submissions. (See JSF in International section.)

Throughout the 1970s and 1980s GE worked on two principal military research and development programmes, the Advanced Turbine-Engine Gas Generator (ATEGG) and the Joint Technology Demonstrator Engine (JTDE). These involved teams at both Evendale and Lynn, and testing of complete engines included outdoor runs at Peebles.The third-generation ATEGG programme was funded by both the USAF and Navy, and centred on the GE23, the first ATEGG variable-cycle engine. This was a turbofan at take-off and during low-speed loiter, but for supercruise (supersonic cruise) the main bypass flow could be shut off. The resulting engine was a turbojet, with just a very small bypass flow mainly for cooling. This gave optimum propulsive efficiency in all phases of flight.

Later in the 1970s the fourth-generation ATEGG was aimed explicitly at M2 fighter propulsion. The resulting GE29 incorporated lightweight low-cost compressor advances developed at Lynn. This led to the fifth-generation GE33/XTE45, which matched variable-cycle technology with what GE call `the first totally new compressor design in 25 years'. The GE33 demo engine first ran in 1986, and by early 1987 had run 128 successful hours. A second build, with significant improvements, provided a core for a new JTDE which was the basis of the XF120. Like some earlier engines this combined variable-cycle features with a 2D (two-dimensional) variable-area, variable-profile variable-direction nozzle, able to be vectored in full afterburner.

YF120-GE-100

Two of these engines powered the first Lockheed YF-22 on its first flight on 29 September 1990, and another pair powered the second Northrop YF-23 on its first flight on 26 October 1990. All these engines had the 2D nozzle. In 1987-88 the ATF prototypes had grown in weight and drag. To match the greater thrust requirement the YF120 was partly redesigned, and when it first flew was more powerful than the rival YF119. The latter was modified with slightly longer fan blades to increase thrust, and was finally selected mainly on the grounds of low risk.

Since this decision GE has collaborated with Allison Engine Co (now Rolls-Royce Corporation) on the Integrated High-Performance Turbine-Engine Technology (IHPTET) programme, and also more recently as propulsion team-mates on the JSF programme in development of the GEA-FXL lift fan. In March 1996 Rolls-Royce, which had purchased Allison a year earlier, joined GE and Allison Advanced Development Co (AADC) in the JSF programme. This move was welcomed by the JSF programme office, which naturally prefers an element of competition between bidders, and also by the UK MoD which prefers participation by UK industry.

YF120-F

In addition to development of the GEA-FXL, the GE/AADC/RR team entered the competition for the JSF main cruise engine with a fixed-cycle variant of the F120, designated YF120-F. Claimed to feature the world's largest core for a tactical fighter engine, the YF120-F will be able to perform the JSF's demanding tri-service mission at relatively low operating temperatures, thereby reducing overall programme risk and cost. Since May 1996 the **JSF-F120** has been a firm programme, as JSF alternative power plant. The core first ran at AADC in August 2000. This engine is described in the JSF entry in the International section.

For historical interest, the following were features of the YF120-GE-100:

Type
Two-shaft variable-cycle turbofan/turbojet.

Fan
Two stages with exceptionally large blisk stages. Fan duct fitted with a ring of blocker doors which can be closed (turbojet configuration) in supersonic cruising flight.

Compressor
Multi-stage axial spool with several variable stators. Exceptionally high work per stage. Fixed bypass between first and second stages. Overall pressure ratio 35+.

Combustion Chamber
Compact dual-annular.

HP Turbine
Single stage, highly loaded, with cooled stator and rotor. Very large blading and flow capacity make this the world's highest-energy tactical engine core.

LP Turbine
Single stage, vaneless (no nozzle ring between rotors), counter-rotating, cooled.

Afterburner
Advanced design giving high augmentation in short axial burn length. Cooled radial flameholders.

Nozzle
Fully variable in area and profile, with vectoring capability. Two-dimensional convergent/divergent (2DCD) design with low-observability features.

Accessories
Conventional, grouped on underside.

Starting
Conventional air-starter assisted, windmill or cross-bleed.

Control System
Dual-channel FADEC.

Dimensions

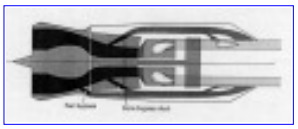
Length	4,242 mm (167.0 in)
Overall diameter	1,067 mm (42.0 in)

Weight, Dry

	1,860 kg (4,100 lb)

Thrust Ratings
In 155.6 kN (35,000 lb st) class; capable of extended supersonic flight (supercruise) without augmentation

UPDATED



F120 variable geometry: upper half, bypass turbojet with reheat; lower half, turbofan mode

YF120-GE-100 for ATF

YF120-FX for JSF

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 23 November 2000

Jane's Aero-Engines 09

GE AIRCRAFT ENGINES

GENERAL ELECTRIC F101

The F101-GE-100 was the augmented turbofan designed for the US Air Force's original B-1 strategic aircraft competition. It was preceded by the GE9 demonstrator engine under the sponsorship of the Air Force Systems Command. From it stemmed the F110, described separately, and the **F101-GE-102** augmented turbofan for the Rockwell B-1B.

Like the earlier model, the F101-GE-102 is a dual-rotor design with a bypass ratio close to 2. Design criteria were durability and operability, while maintaining high turbine temperatures to maximise performance and operating efficiency. The engine incorporates a simplified exhaust nozzle to reduce weight, and neutral position bleed air extraction ports to permit build-up engine assemblies to be installed in any engine position. It provides efficient operation during high-altitude cruise and low-level penetration.

Much progress was made on the F101 engine between the cancellation of B-1 production in June 1977 and launch of the B-1B. A continued engineering development (CED) programme was initiated by the Air Force and GE in 1976 and progressed through 1981. CED objectives were: acceleration of engine maturity, extension of component life, and reduction of engine ownership costs. GE also developed engine repair procedures, minimised manufacturing costs, and refined manufacturing techniques, including engine health monitoring and trending.

At the conclusion of the original B-1 flight test programme in May 1981, F101 engines had powered four B-1 aircraft on 347 flights for a total of 7,600 engine flight hours.

From its inception, the F101 has been designed to incorporate technology that would lessen the impact of exhaust emissions and noise, and enhance fuel conservation. The engine is virtually smokeless.

Delivery of the B-1B to Strategic Air Command units began in July 1985. The 100th and last aircraft was delivered in April 1988. By 30 April 1997, flight time of the engines in active service with 98 B-1B aircraft totalled 1,080,000 hours, with the high-time engine on 4,493 hours. Unscheduled SVR was then 2.35 per 1,000 hours (12 month rolling average). Problems of fan containment have been solved by a new case, but fan blades are now being modified.

In May 2000, the 92nd B-1B became the first of its type to log more than 5,000 hours. At that time the total B-1B flight time was 339,000 hours.

Fan

17-strut front frame with variable trailing guide vanes. The two fan stages have solid titanium blades with tip shrouds. Inlet guide vanes and fan vanes are installed in a horizontally split casing, which permits blades and vanes to be individually replaceable. In May 1999 GE Cincinnati was awarded a USAF contract for US$9,954,800 for 164 fan-blade modification kits. Pressure ratio is over 2. Mass flow 159.7 kg (352 lb)/s.

Compressor

Nine stages, with pressure ratio over 11. First three vane stages and inlet guide vanes variable. Horizontally split casing, forward section of titanium and aft section of steel. Inertia welded discs make continuous steel spool. Blades and vanes individually replaceable. Borescope inspection of vanes, blades and clearances. Overall engine pressure ratio 26.5. Bypass ratio 2.01.

Combustion Chamber
Very short annular, with dual cone nozzles to inject fuel into dome area. Dual high- and low-energy igniters.

HP Turbine
Single-stage, high energy extraction, high tip speed design. Blades and vanes are hollow airfoils which are convective and film cooled. Stationary shroud is segmented and cooled, to provide tip clearance control. TET 1,427ºC class.

LP Turbine
Two stages, tip-shrouded and uncooled. Blades individually replaceable, and second-stage vanes replaceable in segmented groups.

Fan Duct
Upper and lower halves of titanium alloy chem-milled to leave circumferential and closely spaced longitudinal ribs.

Jetpipe
Single pipe with convergent-divergent nozzle. Area variations obtained by hydraulic actuators translating actuation ring which positions flaps and seals through cams and links.

Afterburner
Mixed flow type, with convoluted flow mixer to provide efficient mixing and burning of both fan and core streams. Fan and core flows mix in plane of flameholder where automatic ignition begins on inner ring. Radial flameholders in core stream, for lightoff and stable high-altitude operation. Smooth and continuous temperature rise over the entire modulation range.

Accessories
Twin engines in each nacelle are linked with APU to an accessory-drive gearbox mounting two 115 V IDG alternators, two 276 bar (4,000 lb/sq in) hydraulic pumps and a pneumatic (air-turbine) starter.

Control System
Hydromechanical with electronic supervision of MFC (main fuel control) and fan speed. In 1998, Hamilton Standard was awarded a contract for an entirely new advanced FADEC control. The 450 replacement systems will be delivered from third quarter 2002.

Dimensions

Length	4,590 mm (180.7 in)
Diameter	1,402 mm (55.2 in)

Weight, Dry

2,023 kg (4,460 lb)

Performance Ratings

Max dry (MIL)	75.6 kN (17,000 lb st) class
Max a/b	186.9 kN (30,780 lb st) class

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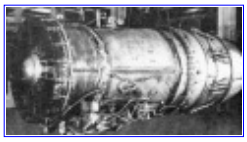
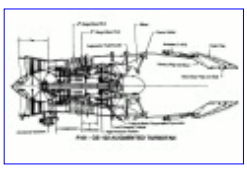

F101-GE-100

F101-GE-102

F101-GE-102


Longitudinal section through F101-GE-102 (1999)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 23 November 2000

Jane's Aero-Engines 09

GE AIRCRAFT ENGINES

GENERAL ELECTRIC TF39

This turbofan was one of the most significant aero-engines in history. It was the first of the giant engines of high bypass ratio that make possible today's large subsonic transport aircraft. It also introduced today's era of much lower specific fuel consumption and reduced noise and emissions.

In 1962 Gen Bernard Schriever, USAF, launched Project Forecast, which asked industry for data upon which strategic decisions could be based. One of the results was the launch of a requirement for a military airlifter of unprecedented size, called CX-HLS. Whittle's calculations in 1936 had shown the importance of high bypass ratio, and to power the giant transport GE set up a team under Don Berkey which submitted an astonishing proposal in which a slim `core engine' drove a huge `1½-stage' fan with a diameter of 2.4 m (8 ft). The USAF instructed GE to build a demonstrator, and the resulting GE1/6 designed for about half the required thrust demonstrated an sfc of 9.52 mg/Ns (0.336 lb/h/lb st), a figure never before even approached.

The result was that in October 1965 GE won the contract to power the big airlifter, which had become the C-5A Galaxy, with the biggest single award in the company's history, valued at US$459,055,000. This covered design, development and delivery of the first 58 examples of an engine designated TF39-GE-1. The first engine began testing in December 1965. First flight with an engine on the right inboard pylon of a B-52E was on 9 June 1967, and the C-5A first flew on 30 June 1968.

TF39-GE-1

Original production engine, fitted to 81 Lockheed C-5A aircraft. T-O rating 182.8 kN (41,100 lb st), flat rated to 31.94ºC. Total production 464 engines, with last delivery 1971, all now converted to -1C standard. Flight time to date, with 77 active aircraft, about 3.5 million hours.

TF39-GE-1C

Upgraded engine incorporating experience with the TF39-1 and CF6 to increase durability and performance retention. Fitted to 50 C-5B aircraft. T-O rating 191.3 kN (43,000 lb st), flat rated to 27.5ºC. Total production 200 engines, with last delivery 1988.

At 30 April 1997 the GE-1C engine was in service on 106 active C-5A and C-5B aircraft, and flight time totalled 6.5 million hours. The high-time engine had accumulated 16,416 hours, and unscheduled SVR was 0.37 per 1,000 flight hours. In July 1997 GE was awarded a US$21,949,601 contract, to run to 31 July 2002, to improve reliability.

Eventually, despite the above efforts, the decision was taken in summer 2000 to replace the TF39 with a more modern engine. On 4 August 2000, it was announced that the REPR (reliability enhancement and re-engining program) choice was the General Electric CF6-80C2L1F (see CF6-80C2). If fully implemented the REPR could involve over 500 engines for 126 aircraft. These would replace the TF39 in 2005-09

Type
Two-shaft turbofan of high bypass ratio. Designed for ready division into three modules: fan, gas generator and LP turbine.

Fan
A unique `1½-stage' design, with a flow-splitter ring dividing the inner flow path from the outer. The first fan stage comprises a rotor fitting inside the splitter ring, with 25 slender blades of solid titanium, with part-span shrouds. Outboard of the splitter are fixed guide vanes carrying the outer fan case. The second stage comprises full-length blades, with part-span shrouds, with an overall fan diameter of 2,443 mm (96.2 in). Mass flow 700 kg (1,541 lb)/s. Bypass ratio 8.0. Provides 85 per cent of T-O thrust.

Compressor
16-stage axial, with IGVs and first six stator stages variable. Interstage bleed for airframe services and engine cooling. Inlet diameter 749 mm (29.5 in). Mass flow (TF39-1) 78 kg (172 lb)/s, with pressure ratio 16.1. Mass flow (TF39-1C) 87 kg (192 lb)/s, with pressure ratio 14.7, giving OPR of 22.

Combustion Chamber
High-efficiency annular, with all components film-cooled. The snout, dome and inner/outer skirt sections are all separable, and the fuel nozzles, igniters and manifold are externally removable with the engine on-wing.

HP Turbine
Two-stage, with film and convection cooling on first-stage nozzles (guide vanes) and rotor blades, and convection cooling of second stage blades. TET approximately 1,316ºC. In May 1997 the decision was taken to modify the HP turbine, and the Air Force awarded a contract valued at US$235,058,275 for 790 (best estimated quantity) HP turbine kits to be retrofitted, with completion in October 2002.

LP Turbine
Six stages, with uncooled shrouded rotor blades. Entry temperature approximately 871ºC. Case split into upper/lower halves for replacement of damaged blades or vanes. In the TF39-1C, the case is externally cooled for active clearance control.

Fan Duct
Comprises a fixed forward fan duct, inner splitter, inner cowl, support assembly, and a reverser with a translating cowl and blocker doors driven by four synchronised actuators. Annular propelling nozzle around the core cowl.

Jetpipe
The core jetpipe leads to a fixed nozzle with a central `plug' bullet.

Accessories
Driven by an inlet gearbox, transfer gearbox off the HP spool and radial and horizontal shafts. Mounting pads

for two hydraulic pumps, CSD-alternator, starter, forward scavenge pump, lube/scavenge pump, fuel pump, tachometer generator and air/oil separator, all mounted on the aft section of the transfer gearbox under the core. The oil tank and fuel heater are above the core on the left side.

Dimensions

Length, flange to flange	4,801 mm (189.0 in)
Length overall	6,880 mm (271.0 in)
Diameter	2,604 mm (102.5 in)

Weight, Dry

TF39-1	3,187 kg (7,026 lb)
TF39-1C	3,583 kg (7,900 lb)

Performance Ratings

See model listing

Specific Fuel Consumption

TF39-1C at T-O rating	8.93 mg/Ns (0.315 lb/h/lb st)

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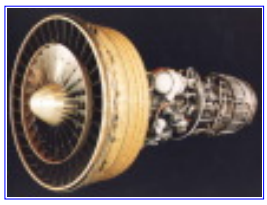
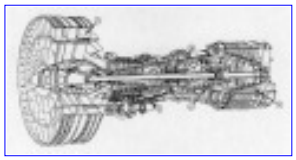


TF39-GE-1C



Cutaway TF39: 1, fan case strut; 2, variable-stator compressor; 3, combustion chamber; 4, HP turbine; 5, LP turbine

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 30 August 2000

Jane's Aero-Engines 08

UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY PW6000

Having sold over 14,500 JT8D engines, Pratt & Whitney decided that it could hardly walk away from the vast potential market for replacements. Accordingly, this new engine is aimed at the thrust range 71.15 to 102.3 kN (16,000 to 23,000 lb st).

Pratt & Whitney claims that "the engine will distinguish itself by virtue of its unmatched low cost of ownership. It will provide airline operators a lower initial investment cost, lower maintenence cost and better operating economics than other engines available in this thrust class. Simplicity is the key. The PW6000 builds on proven technology, gleaned from other advanced PW engine programs to deliver exceptional reliability and component durability".

The company plans to cover this entire thrust range with a single size of fan. The lower thrust (below 20,000 to 21,000 lb) range will be covered using conventional materials in the turbines. For thrusts above this range, the physical geometry of the engine will remain unchanged. The thrust increase will be handled by improved materials and/or coatings in the burner, high and low turbine. The core is derived from the XTC66 demonstrator, testing of which was successfully completed in July 1998.

The PW6000 concept was defined in the 1994-95 time period. Cycle optimisation studies, preliminary design and rig tests have continued from that time period until today. The planned 36-month certification schedule (based on the PW4084 and PW4168 programmes) allowed Pratt & Whitney the luxury of waiting to launch the PW6000 until it was selected for an application. In this way, the engine could be optimised for that particular application.

In January 1998, Pratt & Whitney announced a major programme which is intended to re-engine

some 500 aircraft of the 707 and KC-135 families with the JT8D-219, the very engine the PW6000 is designed to replace. This potentially massive re-engining market, with four engines in each airframe, could have been a great help in launching the new-generation engine, and to build the programme around the JT8D can only have been on grounds of cost and timing.

Thus, ignoring the need to replace older engines, all the obvious applications for the PW6000 have been twin-jets in the 100-seat class. Unfortunately for Pratt, some of the most promising airframes simply went away. The multinational A31X was abandoned (though Airbus, AVIC and Alenia continue to study possible projects), Far East financial (and other) problems made IPTN put the N2130 on a back-burner and the MD-95 became the Boeing 717 wedded to BMW Rolls-Royce. Thus, by the summer of 1998, Pratt's marketing efforts became solidly focused on the Airbus A319M5.

At the 1998 Farnborough Air Show, Pratt & Whitney President Karl Krapek described the application as ``absolutely critical". He said ``We are determined that, if Airbus do the M5, they will select our engine. We cannot give things away, but we have - belatedly - taken a leaf out of GE's book and driven our costs down to unheard-of levels, so we can offer Airbus the best engine at a really unbeatable price. . . . Once we have launched on the M5, we will fight to get on the 717 and I believe we can even get on future 737s."

Only two days later Pratt's unprecedented efforts - Krapek: "We are pulling out all the stops" - paid off. Airbus announced that the M5 was going ahead as the A318, typically seating 108 mixed-class, and that its launch engine would be the PW6000. All previous Airbus aircraft had been launched with GE, CFM or Rolls engines. Formal launch of the A318 was announced on 26 April 1999. At this time the new aircraft had attracted 109 orders and commitments. Airbus then said ``The PW6000 has been selected by several major A318 customers, and remains the engine of preference for the programme. Certain operators of CFM56-powered aircraft have not announced their engine decisions, and wish to study a possible CFM alternative being discussed with the GE/SNECMA joint venture". The principal customer not to announce an engine decision was Air France. This is despite CFMI's 1998 announcement that it would not seek to get aboard the A318 ``as second-choice supplier in a narrow potential market".

The engine is being developed at two initial thrust levels:

PW6122

Derated to 100.24 kN (22,100 lb st). Launch engine for the A318 at an MTO of 59 tonnes (130,070 lb).

PW6124

Fully rated at 105.91 kN (23,800 lb st). Available for A318 IGW versions.

Well into 1998, the PW6000 design had still not become firm. The engine then had a three-stage LP compressor rotating with the fan and a six-stage HP spool. These features are shown in 1997 artwork reproduced here for comparative purposes. These drawings also show the originally projected short fan duct and plain core nozzle. Today, the PW6000 has a four-stage LP compressor, an advanced five-stage HP spool and a mixer nozzle.

By mid-1999 considerable component testing had been completed, including rig-testing the complete compressor, and the first complete engine, No 872, went on test on 29 July 1999. It experienced repeated hung starts, but after slightly changing the vane (stator) schedules the engine ran as predicted on 30 July. By the start of 2000 this first engine had completed preliminary tests and been transferred to a high-altitude cell in the Willgoos laboratory. Two further PW6000s had run, the 873 endurance engine and the No 871 stress telemetry engine. By 2000 No 873 had been shipped to partner MTU in Germany

for two months of endurance testing.

Basic sea-level testing on engine 872 was completed in October 1999, together with combustor rig testing. By the end of the year lubrication system evaluation was essentially complete, and the first FBO (Fan-Blade Off) and birdstrike tests had been run. Full birdstrike testing will involve two tests, one with a 2.5 lb and two 1.5 lb birds together and the other with a single 6 lb bird. Flight testing on inboard station of Boeing 720, with complete control software, was due in August 2000, with FAR Pt 33 certification scheduled for October 2001. The A318 should fly in September 2001, with FAR/JAR certification targeted for August 2002 and EIS one month later.

MTU of Germany, specialist in LP turbines, was from the outset a preferred supplier, but in June 1999 it joined the programme as an 18 per cent risk-sharing partner. A basic design objective for the PW6000 was ``33 per cent fewer blades and vanes than competitor engines". Pratt says it has beaten this target, claiming ``62 per cent fewer airfoils than the CFM56-7B and 52 per cent (later changed to about 46 per cent) fewer than the BR715". The announced number of airfoils in the engine is 2,049. The engine comprises ten modules, a major design aim being to match all lifed components so that all modules can be rated for 25,000 cycles. Among the `25K' items are the fan hub, LPC drum, complete HPC rotor, HPT rotor and LPT disks.

Type

Two-shaft turbofan.

Fan

Single stage, with 24 wide-chord blades, for low cost made of solid titanium alloy. Design to centrifuge foreign matter into fan duct, this process assisted by canted and swept fan exit guide vanes. Robust titanium case, with no need for fragile acoustic panels behind fan. Diameter 1,435 mm (56.5 in). BPR (6122) 5.1, (6124) 4.9.

LP Compressor

Four stages, with rotor blades inserted into thin-wall drum bolted to the rear of the fan hub. No variable stators. Case machined from one-piece titanium forging.

HP Compressor

Five stages with 3-D controlled-diffusion rotor blading incorporating end-bends and bows. All disks of high-nickel alloy. Rotor blades in stages 1,2 of titanium alloy, stages 3,4,5 of high-nickel alloy. First three stators variable. Casing stages 1,2 split, stages 3,4,5 one-piece ring. Pressure ratio 11. OPR, max cruise, 35 k, M 0.72, (6122) 26.6, (6124) 28.7.

Combustion Chamber

Annular, with Floatwall segmented-panel technology. Designed to Talon-2 (Technology for Affordable LO NO_x) using laser-drilled holes to meter flame temperatures. Single ring of airblast fuel nozzles.

HP Turbine

Single stage, with stators (vanes) and rotor blades of single-crystal material. ``No superalloys or protective coatings". TET ``about 200ºF (111ºC) lower than competitor engines".

LP Turbine

Three stages, of considerably greater diameter than the HP turbine. Supplied by MTU (16 per cent partner). Both turbines contained in the same creep-resistant casing of Waspalloy. Cooled first-stage stators. Whole unit assigned to MTU. One-piece exhaust case in C263 alloy fabricated by Howmet at Le Creusot, France.

Jetpipe

Long fan duct and core with 12-lobed mixer made of Inconel 625.

Reverser
In fan duct only. Comlete nacelle assigned to Aircelle (Hispano-Suiza/Airbus).

Accessories
All LRUs mounted one deep. Secured by three standard sizes of bolt, design aim to remove any item in 15 min. Colour-coded wire harnesses.

Control System
FADEC, with self-trimming dual-channel microcomputer giving extensive fault-detection and self-diagnostic capability. To preserve commonality with other Airbus aircraft, based on N_1 (fan speed) instead of EPR, eliminating external pressurized pipes.

The following data are preliminary projected figures:

Dimensions

Length, excl nozzle	2,743 mm (108 in)
Diameter	1,524 mm (60 in)

Weight, Dry

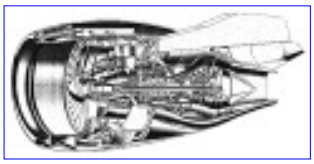
	2,313 kg (5,100 lb)

Performance Ratings
see text

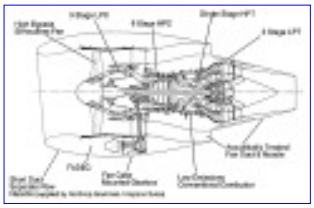
Specific Fuel Consumption
Lower than competitor engines

Contract Price
Target, "in the neighbourhood of US$3 million". A deal with America West for engines for 15 aircraft plus spares and long-term support is valued at "about US$500 million".

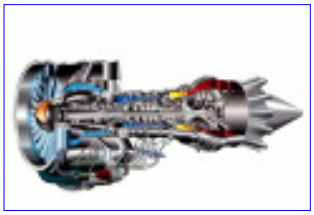
UPDATED



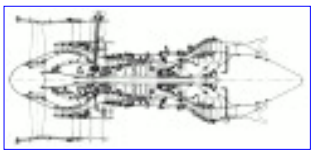
Cutaway drawing showing 1997 installed engine (1998)



Longitudinal section through 1997 PW6000 (1998)



Cutaway drawing of mid-1999 showing definitive PW6000 (1999)



Longitudinal section of October 1998 showing definitive PW6000 (1999)

Checking the HP spool of the first PW6000, which went on test in July 1999 (2000)

PW6000 engine (2000)

PW6000 last bolt ceremony (2000)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 30 August 2000

Jane's Aero-Engines 08

UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY PW2000

US military designation: F117

This turbofan stemmed from a `clean-sheet' design of the late 1960s designated JT10D. With the participation of MTU and Fiat, this first ran at a design thrust in the 106.76 kN (24,000 lb st) class in August 1974. In 1979 the 757 emerged as the principal application, and the JT10D was accordingly redesigned to offer considerably increased thrust. To offer what was claimed to be unrivalled fuel economy both BPR and OPR were also increased, the final baseline specification being reached in 1980.

A formal risk sharing agreement was signed in July 1977 between Pratt & Whitney and MTU (21.2 per cent) and FiatAvio (4 per cent). In 1978 Volvo joined as a manufacturing partner with a 4 per cent stake, leaving Pratt & Whitney with a 70.8 per cent share.

The first engine began testing in December 1981. By this time a new designation system had been adopted, in which this engine became the PW2000 family, the last two digits denoting the T-O thrust rating in thousands of pounds. The following versions have been announced:

PW2037

Initial production engine for 757, with T-O rating of 170.1 kN (38,250 lb st). FPR 1.63, BPR 6.0, OPR 27.0. FAA certification December 1983, first flight on 757 14 March 1984, aircraft entered service with this engine 1 December 1984, qualified for 120 min ETOPS December 1989, extended to 180-minutes April 1992.

PW2040

Uprated engine for 757PF, with T-O rating of 181.9 kN (40,900 lb st). FPR/BPR as PW2037, OPR 27.9. Certificated on 757 freighter at 185.5 kN (41,700 lb st) in September 1987. Reliability problems with the PW2037/2040 had a serious effect on the proportion of 757 aircraft powered by these engines. Pratt & Whitney has fought back with determination. In March 1994 a durability-improvement package became available, with new fan blades, new HP turbine blades and new fan-duct acoustic treatment. In 1995 an Upgrade Package was marketed, concentrating on the HP system. In detail, these changes comprise: extra acoustic treatment in the fan case and fan exit case; new fan blades with a more robust leading-edge; more airfoils in the first four rows (two stator, two rotor) of the core booster to increase core flow 8 per cent, reducing combustor exit temperature by 44ºC (80ºF) and EGT by 28ºC (50ºF); increased capture area of the booster 2.5 bleed to extract more dirt; improved first HP turbine stator (vane) cooling and thermal barrier coating; second-generation monocrystal HP turbine rotor blades with thermal barrier coating, increasing the metal temperature margin by a staggering 167ºC (300ºF); improved internal and external cooling of the second HP rotor blades, together with supercooled thin ceramic air seals with counterflow convection cooling; and increased FADEC computer capacity for modulation of turbine cooling air, improved fault checks and expanded recording capability. These very significant changes result in what is called the PW2000-RTC (Reduced Temperature Configuration). A further improvement, in the Upgrade Package, is to apply advanced blade-tip rub materials (developed on the PW4000) to the HP spool. These modifications, which are available as fleet upgrade kits, are claimed not only to extend life but also to reduce maintenance cost by 22 per cent. This sustained effort has improved the engine's prospects, and in 1996 it was picked by TWA, United, Finnair and (as the PW2040 rather than the F117) for the USAF C-32A, a military version of the 757-200 for Executive and Congressional use.

PW2043

FAA certificated February 1995 at 194.56 kN (43,734 lb st). FPR 1.73, BPR 5.34, OPR 31.2. This engine differs in minor FADEC changes and is aimed at growth versions of the 757.

PW2337

Configured to match nacelle and airframe demands of the Il'yushin Il-96M, rated at 170.1 kN (38,250 lb st). Flight testing of the PW-engined aircraft began in April 1993, with certification originally scheduled for 1995. Despite significant US funding (which includes US$80 million from Pratt & Whitney) the programme has slipped roughly nine months with each passing year. A single ll-96T freighter has been built, and this flew on 16 May 1997 and received FAA certification on 2 June 1999. Instead of being certificated and in production by 1995, by 2000 no ll-96M passenger aircraft had even been completed. If this aircraft could be flown and certificated to FAR Pt 25 it would release funds for the purchase of engines, an impasse yet to be resolved. According to ll'yushin the ll-96M is intended eventually to be powered by the PS-90A2 (see Aviadvigatel, Russia).

F117-PW-100

Essentially the same as the PW2040, rated at 181.9 kN (40,900 lb st). Powers C-17A Globemaster III. Certificated in December 1988, and entered USAF service in June 1993. Sensing British interest in this aircraft, Rolls-Royce offered the 535E4B engine, as preferred by most 757 customers, but a strong Buy

American lobby in the UK military urged adoption of the US engine by the RAF and the RAF aircraft will have the F117.

By 2000 the total number of PW2000 engines shipped was 883, of which about 240 are F117 engines for the USAF. Total flight time of all versions was 18,733,679 hours.

Type
Two-shaft turbofan.

Fan
Single stage, with 36 solid titanium alloy blades with part-span shrouds. Diameter 1,994 mm (78.5 in). Mass flow (2037) 608 kg (1,340 lb)/s.

LP Compressor
Four stages, rotating with fan. Controlled-diffusion aerofoils with thick leading and trailing edges.

HP Compressor
Twelve stages, with variable stator vanes on first five stages. Active clearance on last eight stages. Controlled-diffusion aerofoils.

Combustion Chamber
Annular with flame tube fabricated in nickel alloy. Single-pipe fuel nozzles.

HP Turbine
Two stages, with air-cooled rotor blades cast as single crystals in PW1480 and PW1484 alloy, held in discs sintered from PW1100 nickel-based powder. Active clearance control.

LP Turbine
Five stages, with active clearance control.

Control System
FADEC, with two computers for redundancy.

Dimensions

Length (all)	3,729 mm (146.8 in)
Diameter:	
PW2037, 2040	2,154 mm (84.8 in)
F117	2,146 mm (84.5 in)

Weight, Dry

PW2037	3,311 kg (7,300 lb)
PW2040	3,309 kg (7,295 lb)
F117	3,220 kg (7,100 lb)

Performance Ratings
See model listing

Specific Fuel Consumption

T-O, ideal nozzle:	

PW2037	9.69 mg/Ns (0.342 lb/h/lb st)
F117	9.97 mg/Ns (0.352 lb/h/lb st)
Cruise (10,670 m; 35,000 ft at M0.8)	15.95 mg/Ns (0.563 lb/h/lb)

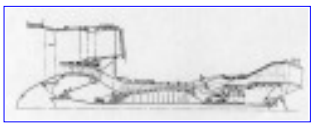
Contract Price

Installed engines and spares for four C-32A (USAF 757), US$54 million. Eight installed engines and spares for Finnair, about US$58 million

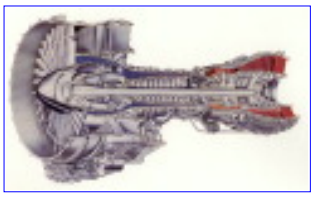
UPDATED

PW2000 first engine to test

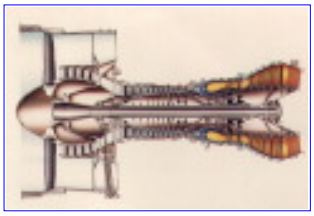
PW2037



Longitudinal section through PW2000



Cutaway drawing of PW2000 series



PW2000 section drawing showing fan containment and active clearance control

PW2040

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 30 August 2000

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UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JT9D

US military designation (JT9D-7): F105-PW-100

Based on technology stemming from the USAF heavy freighter propulsion of 1965, the JT9D was the first civil example of the new era of very large, high-bypass ratio turbofans on which the design of the present generation of wide-body commercial transports rests.

The main advances in the JT9D were: (1) improved fan design to achieve the desired pressure ratio at high efficiency from a single stage with no inlet guide vanes; (2) improved compressor to attain a pressure ratio of 24 in 15 stages, compared with 14 in 16 stages for the JT3D; (3) improved combustion chamber to give greater temperature rise in appreciably shorter length than in previous engines, with lower pressure loss and better exit temperature distribution (and able to use smoke-reduction technology from the outset); (4) new high-temperature materials and cooling systems to allow a substantial rise in turbine gas temperature; (5) a controlled-vortex turbine design, allowing much higher stage-loadings (effectively eliminating two turbine stages); and (6) design features which enabled thrust to be more than doubled with considerably less noise.

In its basic design the JT9D is compact, being shorter than the JT3D, and has two shafts, each supported in two bearings. In cruising flight the installed sfc is 22 to 23 per cent lower than for the JT3D or JT8D. Careful attention has been paid to maintenance. The engine is made in 10 modules which can be individually removed in short times, as demonstrated in numerous tests. By adding brackets and twin rails to the airframe, the engine can be dismantled on the airframe. There are 21 borescope ports for inspecting all stages of blading and the combustion section; and provision is made to facilitate chip

detectors, eddy current, ultrasonic and radioisotope inspection. In the 1960s these were innovations.

First run of the JT9D was in December 1966 and first engine flight test, with the engine mounted on the starboard inboard pylon of a Boeing B-52E, was in June 1968. The first flight of the Boeing 747 occurred on 9 February 1969. The DC-10-40 flew on 28 February 1972.

Current versions include:

JT9D-7

Higher thrust version, with back-sloped compressor blades and vanes, short-cone hooded burners and detail turbine improvements. Certificated June 1971; powers the 747-200B, C, F and SR. By 1972 the pod cowl was simplified, eliminating noisy ring of blow-in inlet doors. Typical performance of -7 series: FPR 1.59, BPR 5.1. On 6 December 1997 at London and 24 February 1998 at Nairobi 747 aircraft suffered strikes from large birds which caused separation of the complete intake assembly. The London strike also caused separation of the fan cowls, jetpipe and exhaust cone, and the engine was classed Beyond Economic Repair. Among other things it was decided to review the 1971 decision to reduce the number of intake attachment bolts from 74 to 37 to speed maintenance.

JT9D-7A

Aerodynamic improvements provide increased thrust at the same turbine temperature and reduced specific fuel consumption. Certificated September 1972; powers 747-200 and 747SP.

JT9D-7F, -7J

JT9D-7A with first- and second-stage turbine rotor blades and second-stage stator vanes of directionally solidified material; -7J also has improved cooling, giving -7F T-O rating without water injection. The -7F was certificated in September 1974 and the -7J in August 1976. Pratt & Whitney offers a major upgrade from the -7A to the -7J, as well as a core-durability upgrade. These improve payload (especially with Stage 3 noise compliance) and are claimed to pay for themselves inside two years (see illustration).

JT9D-7Q, -7R

Described later.

JT9D-20, -20J

Similar to the D-7A and D-7J respectively, except for external configuration changes such as accessory gearbox under fan case. Certificated October 1972 (-20) and December 1986 (-20J).

JT9D-59A, -70A

First growth versions. Fan diameter approximately 25.4 mm (1 in) larger, with reprofiled blades; LP compressor has a zero (fourth) stage and is completely redesigned; burners recontoured, an HP turbine carbon seal is added, HP turbine rotor blades are of directionally solidified PWA 1422 superalloy, and HP turbine annulus is of greater area. FPR 1.60, BPR 4.9. Both models certificated December 1974; configured for installation in a common nacelle, developed jointly by P&W and Rohr Industries, for the 747 (-70A) or DC-10 and A300B (-59A). The **--59D** and **-70D** are higher thrust versions.

JT9D-7Q Series

These have the same gas path as the -59A and -70A but an exterior configured like the -7 for installation in the Boeing 747-200 nacelle. The combination of improved performance, reduced drag and reduced propulsion system weight significantly improved 747-200 performance. The -7Q was certificated in October 1978. This series spans a thrust range of 236 to 249 kN (53,000 to 56,000 lb st). Typical performance, FPR 1.63, BPR 4.9.

JT9D-7R4 Series

This family comprises seven models (7R4D to 7R4H), with common fan, LP and HP compressor, LP turbine and gearbox modules, incorporating the latest technology and materials. Compared with the 1977 D-7A the -7R4 series has a larger fan with wide-chord blades, a zero stage on the LP compressor, improved combustor, single-crystal HP turbine blades, increased diameter LP turbine, a supervisory electronic fuel control and many smaller changes. The new engines offer an sfc reduction of up to 8 per cent. Typical performance, FPR 1.67, BPR 4.8. The 7R4D, 7R4E and 7R4E4 for the Boeing 767 were certificated in November 1980, June 1982 and March 1985 respectively, the 7R4D1 and 7R4E1 for the A310 were certificated in April 1981, and the -7R4G2 for the 747 and the -7R4H1 for the A300B-600 in July 1982. ETOPS approval was gained in June 1985. Engines are offered as a complete package in the PW7R4 nacelle; this propulsion system was selected for the A300-600 and A310.

The JT9D entered service on the 747 on 21 January 1970. Production was completed in 1990 at 3,265 engines, of which 3,234 were shipped to customers. Early versions, excluding the JT9D-7R4, numbered 2,469, and by December 1999 these had logged a total flight time of 131,740,467 hours. Total shipments of JT9D-7R4 engines amounted to 765 engines, and by December 1999 these had logged a total flight time of 26,911,884 hours. In 2000 about 2,800 JT9D engines remained in service.

The following description applies to early versions of the JT9D, with data for later models given in parentheses:

Type
Two-shaft turbofan of high bypass ratio.

Intake
Direct pitot, annular fixed geometry (except that airframe inlet on early 747 aircraft has blow-in side doors around periphery). No inlet guide vanes ahead of fan. Air flow improved by rotating spinner.

Fan
Single stage, with 46 titanium blades of 4.6 aspect ratio (-7R4, 40 blades of 4.0 ar) and two part-span shrouds (-7R4, one shroud) dovetailed in titanium LP rotor. Downstream are 108 aluminium alloy exit guide vanes (96 on the -59A, -70A and -7Q, 84 on -7R4), followed by nine discharge-case radial struts. Fan case of stainless steel (7R4, titanium) and aluminium alloy, designed to contain fan blades. Discharge case lined with perforated acoustic material. Nominal air flow 684 kg (1,509 lb)/s at 3,650 rpm (-7, 698 kg; 1,540 lb/s at 3,750 rpm; -59A, -70A, -7Q, 744 kg; 1,640 lb/s at 3,430 rpm; -7R4G/H, 769 kg; 1,695 lb/s at 3,530 rpm).

LP Compressor
Three stages (JT9D-59A, -70A, -7Q, -7R4, four stages), rotating with fan. Rotor made up of rings, spacers and integral hub/disc splined to steel LP shaft and held by lock-nut ahead of fan and overhung ahead of main LP ball thrust bearing. Hydraulically opened bleed ring at LP exit to increase stall margin during deceleration and low-power operation. Rotor stages have 104, 132 and 130 (-7Q, 59A, -70, -7R4,

108, 120, 112, 100) dovetailed blades of titanium alloy. First stator stage (except -7R4) anti-iced by ninth-stage bleed air. Stator stages have 88, 128 and 126 (-7Q, -59, -70, 7R4, 96, 114, 116, 104, 88) titanium vanes and 120 (fourth stage) nickel alloy vanes, all riveted to outer rings. First stator nickel alloy, remainder corrosion-resistant iron alloy. Casing of aluminium alloy. Core air flow typically 118 kg (260 lb)/s (all versions).

HP Compressor

Eleven stages. All stages have rings or centreless discs with integral spacers carried on titanium hub/disc at third stage and nickel bolted hub at 11th stage. Rotor stages have 60, 84, 102, 100, 110, 108, 104, 94 and 100 dovetailed titanium blades and 102 and 90 nickel alloy blades. Stator has 76, 70, 80, 106, 100 and 112 steel vanes and 126, 146, 154, 158 and 92 vanes of nickel alloy, the last eight stages brazed to inner and outer rings. First three stator stages are variable, plus the intermediate IGV stage, positioned by hydraulic actuator to provide adequate stall margin for starting, acceleration and part-power operation. Casing of titanium alloys (last two stages, nickel alloy) has bleed ports supplying eighth-stage air for airframe requirements. Max HP speed: -7, 8,000 rpm; -7R4E4/G/H, 8,080 rpm. OPR, -7, 22.2; -7Q, -20, 21.1; -59A, -70, 24.5; -7R4D, 23.4; -7R4E, 24.2; -7R4G, 26.3; -7R4H, 26.7.

Combustion Chamber

The diffuser case incorporates two sets of bleed ports for 15th-stage air for airframe requirements. The forward set (absent from -7R4) takes air from the outside case via an integral manifold and the rear set bleeds air from the inner diameter via four of the 10 radial struts. The combustor itself is fabricated in nickel alloy and is annular. Ignition by dual AC 4 J capacitor system serving two plugs just above chamber centreline on each side.

HP Turbine

Two stages. Both have high-nickel discs carrying high-nickel blades in fir-tree roots; first stage has 116 air-cooled blades and second has 138 solid blades (air-cooled in -D7 and all subsequent models). Stators have 66 and 90 high-nickel alloy vanes, both rows air-cooled. (-7R4, single crystal alloy in first-stage blades to 222.4 kN, 50,000 lb st, and in first and second blades and second stator (vane) for higher-thrust models.) Turbine inlet temperature (max T-O): typically (-7) 1,243ºC; -59A, -70A: 1,350 to 1,370ºC; -7R4: 1,200 to 1,300ºC.

LP Turbine

Four stages. Stages have 122, 120, 110 and 102 solid nickel alloy blades held in fir-tree roots in discs of nickel alloy (last disc, iron alloy). Stators have 108, 126, 122 and 116 solid nickel alloy vanes. In 1982 an improved LP turbine was introduced to -7R4 production, consisting of four stages with integral spacers carried on a single bolted hub splined to the LP shaft (see cutaway). Exhaust gas temperature after turbine, typically 482ºC (-7, -20), 580ºC (-59A, -70, -7Q), 500ºC (-7R4D), 535ºC (-7R4E), 560ºC (-7R4G2) and 575ºC (-7R4H1).

Jetpipe

Fixed Inconel assembly.

Reverser

Fan duct reverser comprises a translating sleeve (the rearmost portion of fan duct) which moves aft, causing long links to close the blocker doors and simultaneously pulling aft the cascade vanes. Primary (core) reverser, largely of Inconel 625, uses fixed cascades which are uncovered by aft movement of translating sleeves to which are hinged blocker doors pulled by links against the central nozzle plug. No primary reverser is used on -59A, -70A, -7Q or -7R4.

Mounting

From above, in two planes. Front mount (-7, -7R4G2) is double flange at top of fan discharge case, absorbing vertical and side loads. On -20, -20J, -59A, -70A and 7R4 (except 7R4G2) the mount is rectangular block above intermediate case, taking vertical and side loads, and thrust brackets at 40º each side of vertical on intermediate case outer flange.

Accessories

Main accessory gearbox driven by tower bevel shaft from front of HP spool and mounted under central diffuser case (-20, -59A, -70A, under fan discharge case). Main driven accessories include CSD (IDG on -7R4 except G2) fuel pump and control, starter, hydraulic pump, alternator and N_2 tachometer; Boeing 747 includes primary reverser motor and the DC-10-40 a second hydraulic pump and a fuel boost pump. The box also includes numerous lubrication system items, and provides for hand-turning the HP spool during borescope inspection.

Starting

Pneumatic, by HamStan PS 700 or AiResearch ATS100-384 (DC-10, PS 700 only). Supplied at 2.76 to 3.10 bars (40 to 45 lb/sq in) from APU, ground cart or cross-bleed.

Control System

Pressure type with hydraulic control system operating at up to 76 bars (1,100 lb/sq in). Main components are fuel control, pump, fuel/air heater and fuel/oil heat exchanger. (-7R4 except G2 has digital electronic system to trim hydromechanical control; engine is operational with or without electronic system functioning). Provision for water injection, as customer option, with regulator, piping and spray nozzles, adds 18.1 kg (40 lb) to engine weight (not fitted to -7R4).

Fuel Specifications

P&W specification PWA 522.

Oil System

Pressure feed through fuel/oil cooler to four main bearings and return through scavenge pumps (-20 also centrifugal scavenge) to 18.8 to 37.6 litre (5 to 10 US gallon; 4.16 to 8.32 Imp gallon) tank.

Oil Specifications

PWA 521C (blend of synthetic and/or mineral oils).

Dimensions

JT9D-7, -7A, -7F, -7J, -20:	
Diameter	2,427 mm (95.56 in)
Length (flange to flange)	3,255 mm (128.15 in)
JT9D-59A, -70A, -7Q:	
Diameter	2,464 mm (97.0 in)
Length	3,358 mm (132.2 in)
JT9D-7R4D to H:	

Diameter	2,463 mm (96.98 in)
Length	3,371 mm (132.7 in)

Weight, Dry

Guaranteed, including standard equipment:	
JT9D-7, -7A, -7F, -7J	4,014 kg (8,850 lb)
JT9D-20	3,833 kg (8,450 lb)
JTRD-20J	3,883 kg (8,560 lb)
JT9D-59A	4,146 kg (9,140 lb)
JT9D-70A	4,153 kg (9,155 lb)
JTD9-7Q	4,216 kg (9,295 lb)
JT9D-7R4D, E, E4	4,039 kg (8,905 lb)
JT9D-7R4D1, E1	4,029 kg (8,885 lb)
JT9D-7R4G2	4,146 kg (9,140 lb)
JT9D-7R4H1	4,029 kg (8,885 lb)

Performance Ratings (ideal nozzles):

T-O thrust, dry:	
JT9D-7	202.8 kN (45,600 lb st) to 26.7ºC
JT9D-7A	205.7 kN (46,250 lb st) to 26.7ºC
JT9D-7F	213.5 kN (48,000 lb st) to 26.7ºC
JT9D-7J, -20J	222.4 kN (50,000 lb st) to 30ºC
JT9D-20	206.0 kN (46,300 lb st) to 28.9ºC
JT9D-59A, -70A, -7Q	236.0 kN (53,000 lb st) to 30ºC
JT9D-7R4D, D1	213.5 kN (48,000 lb st) to 33ºC
JT9D-7R4E, E1	222.4 kN (50,000 lb st) to 33ºC
JT9D-7R4E4	222.4 kN (50,000 lb st) to 45.6ºC
JT9D-7R4G2	243.4 kN (54,750 lb st) to 30ºC
JT9D-7R4H1	249.0 kN (56,000 lb st) to 30ºC
T-O thrust, wet:	
JT9D-7	210.0 kN (47,200 lb st) to 30ºC
JT9D-7A	212.4 kN (47,750 lb st) to 30ºC
JT9D-7F	222.4 kN (50,000 lb st) to 30ºC
JT9D-20	220.0 kN (49,400 lb st) to 30ºC
Max cruise thrust, 10,670 m (35,000 ft) at M0.85:	

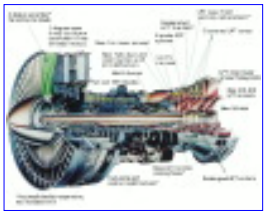
JT9D-7	45.4 kN (10,200 lb st)
JT9D-7A	48.2 kN (10,830 lb st)
JT9D-7F, -7J	49.2 kN (11,050 lb st)
JT9D-20, -20J	47.5 kN (10,680 lb st)
JT9D-59A, -70A, -7Q	53.2 kN (11,950 lb st)
JT9D-7R4D, D1	50.0 kN (11,250 lb st)
JT9D-7R4E, E1	52.0 kN (11,700 lb st)
JT9D-7R4E4, G2, H1	54.5 kN (12,250 lb st)

Specific Fuel Consumption (ideal nozzles):

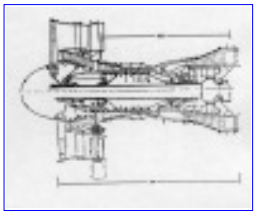
Max cruise, ISA + 10ºC, M0.85 at 10,670 m (35,000 ft):	
JT9D-7	17.55 mg/Ns (0.620 lb/h/lb)
JT9D-7A	17.69 mg/Ns (0.625 lb/h/lb)
JT9D-7F, -7Q, -59A, -70A	17.87 mg/Ns (0.631 lb/h/lb)
JT9D-20, -20J	17.67 mg/Ns (0.624 lb/h/lb)
JT9D-7R4D, D1	17.42 mg/Ns (0.615 lb/h/lb)
JT9D-7R4E, E1	17.55 mg/Ns (0.620 lb/h/lb)
JT9D-7R4G2	18.10 mg/Ns (0.639 lb/h/lb)
JT9D-7R4H1	17.79 mg/Ns (0.628 lb/h/lb)

UPDATED

The JT9D introduced a wholly new concept of engine size. This is the first JT9D prototype fan case in 1966

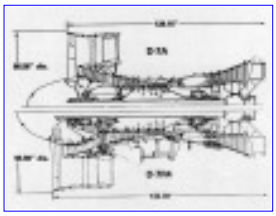


Modifications in the -7A to -7J Upgrade and the -7J core-durability kit

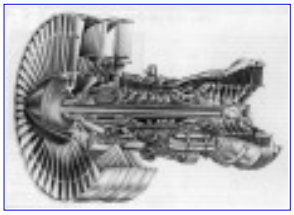


Comparative cross-sections of JT9D-20 (lower half) and JT9D-59

JT9D-59A



Comparative cross-sections of JT9D-7A (upper half) and JT9D-7R4



Cutaway drawing of JT9D-7R4

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 30 August 2000

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UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JT8D

This turbofan engine was developed as a company-sponsored project to power the Boeing 727. It was later selected for other types of aircraft, and supersonic military versions of the JT8D have been developed in Sweden by Volvo Flygmotor (see RM8 in that company's entry).

Construction of the JT8D is largely of steel and titanium. An annular bypass duct runs the full length of the engine, with balanced mixing of the hot and cold air streams in the tailpipe.

The JT8D entered commercial service on 1 February 1964. It rapidly became the most widely used airline engine in the world, and also gained flight time faster than any previous engine in history. By 1982, excluding the JT8D-200 and RM8, a total of 11,845 had been produced, of which 11,757 were shipped to customers. By the end of 1999 the total flight time of these engines had reached 454,272,272 hours.

The following versions were produced:

JT8D-7, -7A

Rated at 62.28 kN (14,000 lb st) to 28.9ºC. FPR 1.93, BPR 1.07, OPR 15.4. A few still power 727,737.

JT8D-9, -9A

Rated at 64.5 kN (14,500 lb st) to 28.9ºC. FPR 1.96, BPR 1.04, OPR 15.9. Specified for Boeing 727-100, -100C and -200, 737-200, -200C and T-43A, McDonnell Douglas DC-9-20, -30, -40, C-9A, C-9B and VC-9C, Aérospatiale Caravelle 12 and Kawasaki C-1. Deliveries began in July 1967. Produced under licence in Japan (see entry under Mitsubishi) until 1981.

JT8D-11

Rated at 66.7 kN (15,000 lb st) to 28.9ºC at S/L. Specified for McDonnell Douglas DC-9-20, -30 and -40 series aircraft. Deliveries began in November 1968. No longer in use.

JT8D-15

Rated at 69 kN (15,500 lb st) to 28.9ºC. FPR 2.02, BPR 1.03, OPR 16.6. FAA certification was received and deliveries began in April 1971. Powers Dassault Mercure, Boeing Advanced 727 and 737, and DC-9. Entered service 1972.

JT8D-15A

In 1982, new components in the Dash-15 engine resulted in a 5.5 per cent reduction in cruise fuel consumption, with unchanged thrust. FPR 2.05, BPR 1.08, OPR 16.4. The same parts in the Dash-17 produce the **JT8D-17A**, and when fitted to the Dash-17R the **JT8D-17AR** (thrust as -17R but FPR 2.16, BPR 0.96, OPR 18.5). Some can be incorporated in earlier JT8D engines, giving up to 3 per cent reduction in fuel consumption. Following uncontained turbine failures aboard Delta 727s on 15 January and 14 August 1996, the NTSB ordered an investigation into forged discs made prior to 1989, and periodic checks on them.

JT8D-17

Rated at 71.2 kN (16,000 lb st) to 28.9ºC. FPR 2.09, BPR 1.02, OPR 16.9. Certificated on 1 February 1974. Entered service July 1974. Powers Advanced versions of Boeing 727 and 737, and DC-9. For JT8D-17A, see JT8D-15A.

JT8D-17R

Normal T-O rating 72.95 kN (16,400 lb st) to 25ºC, but has capability of providing 4.448 kN (1,000 lb) additional thrust in the event of significant thrust loss on any other engine. Certificated at 77.40 kN (17,400 lb st) T-O rating in April 1976. FPR 2.21, BPR 0.97, OPR 18.2. Delivery of Advanced 727-200 August 1976 and certification with reserve-thrust feature November 1976. For JT8D-17AR, see JT8D-15A.

JT8D-200 Series

Described separately.

Since February 1970, all new JT8D engines have incorporated smoke-reduction hardware, and conversion kits are available for in-service engines. Maximum TBO for the JT8D is 16,800 hours.

Ageing problems, with both the original family of engines and the re-fanned -200 family, continue to cause concern. Problems have included disintegration of the fan hub, HP compressor discs and the LP turbine.

In March 1999 the NTSB recommended that the FAA should order operators to inspect compressor tie rods for cracks or corrosion. This alert was prompted by uncontained turbine failure of a JT8D-17A. In July 1999 the NTSB recommended that all JT8D engines should be required by the FAA to have a one-piece integral-boss combustion chamber outer case installed at the next shop visit.

Apart from this, the biggest problem facing JT8D operators is the need to meet stricter Stage 3 noise requirements (see feature article Environmental factors). Three types of JT8D hushkit are available; for the 727, developed in conjunction with Federal Express; for the DC-9, in conjunction with ABS Partnership; and for the 737, in conjunction with Nordam. Pratt & Whitney has also collaborated with BFGoodrich Aerospace on the Super 27 programme. This replaces the 727's outboard engines with JT8D-200 models and installs an exhaust mixer on the centre engine. This not only gives Stage 3 compliance but also increases payload and

range. In addition, AvAero offers a 737 kit using P&W-supplied parts. All hushbits involve substantial additions to the installed engine to increase airflow and promote rapid core mixing with an enlarged slower-moving jet. The Federal Express 727 kit comes in Light and Heavyweight versions, respectively priced at around US$1.9 million and US$2.575 million per shipset. The ABS DC-9 hushkit likewise is available in various forms (the maximum configuration is the subject of a diagram) priced at US$1.743 million per aircraft. By June 1997, airlines had ordered about 1,600 shipsets of JT8D hushkits.

Type
Axial flow two-spool turbofan.

Intake
Annular with 19 fixed inlet guide vanes.

Fan
Two-stage front fan. First stage has 27 titanium blades dovetailed into discs. First-stage blades have integral shroud at about 61 per cent span. Air flow: -7, -7A, 143 kg (315 lb)/s; -9, -9A, 145 kg (319 lb)/s; -11, -15, 146 kg (322 lb)/s; -17, 147 kg (324 lb)/s, -17R, 148 kg (326 lb)/s.

LP Compressor
Six-stage axial, integral with fan stages, on inner of two concentric shafts. Blades made of titanium. Shaft carried in double ball bearings, either half of each bearing being able to handle the complete loading.

HP Compressor
Seven-stage axial flow on outer hollow shaft which, like the inner shaft, is carried in double ball bearings. One-piece casing. Blades made of steel or titanium.

Combustion Chamber
Can-annular type with nine cylindrical flame tubes, each downstream of a single Duplex burner and discharging into a single annular nozzle.

HP Turbine
Single-stage axial flow. Solid blades in -9; air-cooled in -11 and later; guide vanes hollow and air-cooled in all models.

LP Turbine
Three-stage axial flow. Solid blades and guide vanes.

Dimensions

Diameter	1,080 mm (42.5 in)
Length	3,137 mm (123.5 in)

Weight, Dry

JT8D-7, -7A	1,454 kg (3,205 lb)
JT8D-9, -9A	1,532 kg (3,377 lb)
JT8D-11	1,537 kg (3,389 lb)
JT8D-15	1,549 kg (3,414 lb)
-15A	1,576 kg (3,474 lb)
JT8D-17	1,556 kg (3,430 lb)
-17A	1,577 kg (3,475 lb)
JT8D-17R	1,585 kg (3,495 lb)
-17AR	1,588 kg (3,500 lb)

Performance Ratings

T-O thrust (S/L, static): See model descriptions	
Max cruise thrust (10,670 m; 35,000 ft at M0.8):	
JT8D-7, -7A	16.13 kN (3,630 lb)
JT8D-9, -9A	18.2 kN (4,100 lb)
JT8D-11	17.6 kN (3,950 lb)
JT8D-15, -15A	18.2 kN (4,100 lb)
JT8D-17, -17R, -17A, -17AR	18.9 kN (4,240 lb)

Specific Fuel Consumption

T-O rating:

JT8D-7, -7A	16.57 mg/Ns (0.585 lb/h/lb st)
JT8D-9, 9A	16.85 mg/Ns (0.595 lb/h/lb st)
JT8D-11	17.56 mg/Ns (0.620 lb/h/lb st)
JT8D-15	17.84 mg/Ns (0.630 lb/h/lb st)
-15A	16.63 mg/Ns (0.587 lb/h/lb st)
JT8D-17	18.27 mg/Ns (0.645 lb/h/lb st)
-17A	17.05 mg/Ns (0.602 lb/h/lb st)
JT8D-17R	18.55 mg/Ns (0.655 lb/h/lb st)
-17AR	17.31 mg/Ns (0.611 lb/h/lb st)

Max cruise rating, as above:

JT8D-9, -9A	22.86 mg/Ns (0.807 lb/h/lb)
JT8D-11	23.14 mg/Ns (0.817 lb/h/lb)
JT8D-15	22.97 mg/Ns (0.811 lb/h/lb)
JT8D-17, -17R	23.37 mg/Ns (0.825 lb/h/lb)

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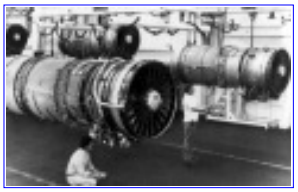
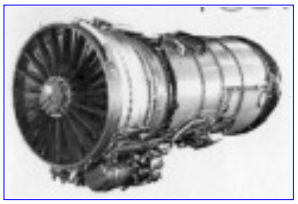
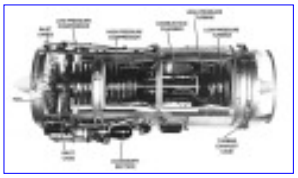


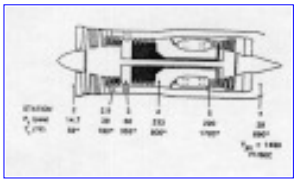
JT8D-15



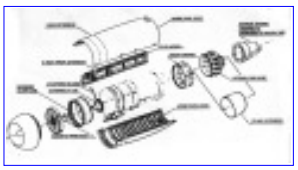
JT8D-15



Cutaway JT8D
(1998)



Internal pressures and temperatures in JT8D (14,000 lb rating)



Features of the ABS-P&W Stage 3 hushkit for the DC-9

Rear view of JT8D-200
(2000)

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 30 August 2000

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Pratt & Whitney, a long-established firm of precision engineers, formed Pratt & Whitney Aircraft in 1925 to build one product, the Wasp piston engine. Other designs followed, virtually all air-cooled radials, and in 1939-45 Pratt & Whitney delivered 363,619 engines with an aggregate horsepower of 603,814,723. No other company has ever even approached these figures. The move into gas turbines was assisted by a licence for the Rolls-Royce Nene turbojet, but the company really began to exploit the new technology with the design of the JT3 axial turbojet with separate LP and HP compressors. After a troubled start this became, as the J57, the most important military engine in the USA, and, as the JT3C, the engine that launched the civil Jet Age. Both appear in this publication.

Prodded by Rolls-Royce, the JT3C was refanned into the JT3D, which virtually killed the competition. To power the 727, the JT8D was produced, and with about 14,400 units sold this was until 1998 the best-selling civil engine of all time. In parallel with the JT8D, a new species of engine was designed, a giant turbofan of high bypass ratio to power the C-5A. Pratt & Whitney lost this competition, but its JT9D was picked by Boeing for the 747, and this was far more important than the C-5A contract. This engine pioneered the giant turbofan in airline service.

In 1976, the company was restructured, ceasing to be a Division of United Aircraft Corporation and instead becoming a business unit of United Technologies Corporation (UTC). The Florida research and development centre, originally created to produce rocket engines for space vehicles, was renamed the Government Products Division (today restyled as in the heading), and this has ever since been the centre for military engines. The J57 was followed by the TF30, which did little to help the company's proud logo `Dependable Engines'. The situation was transformed by the F100, which has gone from strength to strength and played a major role in Pratt & Whitney's winning of the most important fighter propulsion contract in the world today with the F119. In turn, the F119 is the baseline engine for the potentially even more important JSF programme.

Large Military Engines states that its 400-strong team working on the Integrated High-Performance Turbine-Engine Technology (IHPTET) programme is the only one to have successfully demonstrated all goals in Phase I. Its XTE-65/2 engine `delivered the required 30 per cent increase in T/W ratio and exceeded the 167ºC (300ºF) increase in core running temperature by 56ºC (100ºF) in September 1994'. The team is now testing Phase II and designing the Phase III system to achieve T/W double that of the F119 with other dramatic advances by 2003.

Commercial Engines developed the PW2000 in the `40K' class, and then simplified the JT9D into the dramatically superior PW4000, a family of engines with three types of LP turbine driving three sizes of fan to cover a range of thrusts from 50 to 100K. For the 25 to 30K bracket, the company took a 33 per cent share in IAE (see in International section) to produce the V2500. What the company failed to do was have a modern successor ready to take over from the JT8D.

It is mainly because of this engine that Pratt can claim "more than 29,000 P&W jet engines power over half the world's fleet of commercial aircraft. These engines have accumulated over one billion hours... a P&W-powered jet takes off about every three seconds". Under its previous President, Karl Krapek, Pratt determined to rectify this situation, and has already achieved a major foothold in having the PW6000 selected for the A318 before a complete engine had even run. The next objectives are to get PW6000 versions on the 717 and 737 while, following 10 years of work on gearboxes, the impressive PW8000 is searching for a market niche. One wonders what all this will do to the IAE partnership.

Like rival GE, Pratt is devoting great efforts to building up aftermarket services. It has now grouped all such operations in Pratt & Whitney Eagle Services, which has grown from US$250 million in 1993

to over US$1 billion, with plans to double this in the next five years. Since 1996, several large contracts have been signed with airlines, some for five- or 10-year support. Large commercial engines, including V2500, are handled at Cheshire, Connecticut, though other facilities (such as Eagle Services Asia, opened in 1998 at Singapore) serve local markets. All JT8D versions are overhauled at a new facility at Columbus, Georgia. Military engines are handled by East Hartford and (F100 modules, combustion chambers and augmentors) by San Antonio, Texas. Other major locations include North Berwick, Maine; Springdale, Arkansas; Maastricht, Netherlands (main centre for Europe); Dublin, Ireland; Singapore; and Taipei, Taiwan. In January 1996, a 24-hour telephone help desk went on line. In July 1997, a joint-venture company with Singapore Airlines was announced to tap the engine-overhaul market in the Asia-Pacific region.

In May 1997, GE's global dominance of engine maintenance was further challenged by the launch of a `thrust manager' scheme. This is to provide a guaranteed `cradle to grave' for all airline propulsion requirements.

Following prolonged debate about how best to support USAF engines in service, either by outsourcing or by in-house effort, a long-term contract was signed on 3 December 1999 (a similar contract was awarded four days later to GE). The text reads: United Technologies Corp, West Palm Beach, Florida, is being awarded a $2,685,400,000 indefinite-delivery/indefinite-quantity contract F33657-99-D-2051 to provide for technical services from January 2000 through December 2014 for the Component Improvement Program supporting the F100 engine on the F-15 and F-16 aircraft, the F119 engine on the F-22 aircraft, the TF33 engine on the B-52, C-141, E-3 and KC-135 aircraft, and the J57 engine on the KC-135 aircraft. Expected completion date is 31 December 2014. Aeronautical Systems Center, Wright-Patterson AFB, Ohio, is the contracting activity.

Together with AVIC's Chengdu Engine Co (see in Chinese section), Pratt & Whitney has formed Aerotech Manufacturing. This will produce `commercial aircraft engine-related components.' AVIC is also playing a role in the design of the PW6000.

Essentially, all the Pratt & Whitney commercial engine programmes involve foreign partners, but a totally new development was that in 1996 Large Commercial Engines entered into a team effort with General Electric on an engine for stretched 747s, described earlier under Engine Alliance.

Pratt & Whitney Canada, described separately under Canada, is the United Technologies company for gas-turbine engines for General Aviation. It is 97 per cent owned. It has delivered nearly 45,500 engines, bringing the combined total of gas-turbine aero-engines from Pratt & Whitney to an unrivalled 110,000 plus. Of these, more than 27,000 are large airline jet engines.

For 40 years, Pratt & Whitney has been a major producer of rocket engines for space launchers. Its latest product in this field is the RD-180, rated at 4,004 kN (900,000 lb), jointly developed with Energomash of Russia (the original designer) in a joint venture called RD Amross (from `America Rossiya'). In early 1997, this was selected by Lockheed Martin for the Atlas IIAR (now designated Atlas IIIA) in a 101-engine contract valued at US$1 billion. In July 1998, the first RD-180 went on test at Marshall Space Flight Center. First launch took place in 1999. This work is to lead to the Lockheed Martin Evolved Expendable Launch Vehicle. In early 2000 the US State Department was being urged to approve a five-year licence that would allow Pratt & Whitney and Lockheed Martin to co-produce the RD-180 in the USA.

In 1998, Pratt & Whitney had 31,600 employees and revenues totalled US$7.6 billion. Just over half this total was accounted for by large commercial engines. About one-sixth was contributed by military engines, one-sixth by Pratt & Whitney Canada and about US$1 billion by aftermarket repair and overhaul, excluding spare parts. Sales target for 2000 is US$7.4 billion, rising to double this in 2004.

In August 1999 United Technologies announced that, following cutbacks in other operating units, it

would by late 2000 reduce Pratt & Whitney's floor area from 1,350,000 m^2 (14,532,750 sq ft) to 1,080,000 m^2 (11,625,400 sq ft), and shed 6.8 per cent of its workforce. Some observers equated this with a reduction of 1,500 but arithmetic suggests 2,150. Most of the cuts affect West Palm Beach. Large Military Engines is being relocated to existing sites at Hartford and Middletown in Connecticut. About 1,300 staff will remain in Florida on Space Propulsion and on some engine testbeds. A further 900 redundancies are expected from closure of the Connecticut repair and overhaul operations.

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 30 August 2000

Jane's Aero-Engines 08

LHTEC - LIGHT HELICOPTER TURBINE ENGINE COMPANY

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These two companies jointly developed the Army Qualified T800-LHT-800 975 kW (1,300 shp) class turboshaft engine for the RAH-66 Comanche programme. The CTS800 is an FAA-certificated version for commercial applications. The CTP800 is a commercial turboprop derivative.

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 30 August 2000

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UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY PW8000

In March 1998, this engine was announced as a major competitor in the 111 to 156 kN (25,000 to 35,000 lb st) market. Though the primary objective at that time was to beat the most powerful versions of CFM56, the PW8000 was also intended to compete with the V2500 produced by IAE, in which Pratt & Whitney and MTU are partners. Together with the PW6000, with which it was to share almost a common core, it underscores Pratt's determination to restore its position in the narrow-body market (once a dominating 90 per cent). For many years, Pratt has exhorted its IAE partners to develop the V2500, and the fact that the PW8000 is not an IAE engine is significant.

Since the 1980s, Pratt & Whitney has been studying how best to design a geared turbofan. The problem of a direct-drive engine is that it is difficult to match the rotational speeds of the turbine and fan. As bypass ratio is increased (to improve fuel economy and reduce noise), the rotational speed of the larger fan must fall, demanding either a large, heavy and costly multistage turbine or the insertion of a speed-reducing gearbox. Long-established examples of geared turbofans include the AlliedSignal LF500 and TFE731.

At first, Pratt & Whitney called its proposed engine the ADP (Advanced Ducted Propulsor). Prolonged research and testing was carried out, in partnership with MTU of Germany and FiatAvio of Italy. Mechanical design of the gearbox, like other factors, is discussed in the engine description. One obvious problem is reliability, and another is dissipation of heat via lubricating oil. In the case of low-wing aircraft, the diameter of the engine might be a problem with underwing installations.

Accompanying drawings show the general layout of the PW8000 and a detail of the vital gearbox, as

it was conceived in 1998. Comparing the PW8000 with the contemporary Engine Alliance GP7000, we see that the latter needs a massive five-stage LP turbine to drive its fan and two-stage LPC. Another Pratt & Whitney engine, the PW4084 series, needs an LP turbine with seven stages. This is nice business for MTU, who are a world-class supplier of LP turbines but, in the longer term, the simpler turbine and geared drive may well prove the better answer. According to Pratt, once BPR exceeds about 8, a geared drive is desirable. In the case of the PW8000, the FPR has deliberately been kept low - much less than the 1.8 of the 94-in PW4000 - in order not to need a variable-pitch fan.

After much study Pratt decided to develop the PW8000 gearbox to transmit a T-O power of 23,862 kW (32,000 shp). Other data are given in a box by the cutaway drawing. In mid-1999 intensive bench testing was in progress with 12 units accumulating a planned 4,200 hours which includes endurance testing with intentional misalignment. A press release states ``Once a launch commitment is made...full engine certification will take about 30 months. If the full-scale program were to begin in 2002, a PW8000-powered aircraft could enter service in 2005".

Predictably, Pratt has found great difficulty in deciding which market to go for. The Editor was told that there would be no great problem in re-engining a developed 737 or A320 with a PW8000 in the 30K class, though the former would need new landing gears. His question ``You don't mind hurting the V2500?" was answered with a ``No". However, such an engine seemed in June 1999 to be unlikely to happen. The Editor was told ``We are examining a range of take-off thrusts from 25 to 65K [111-289 kN, 25,000-65,000 lb st] and could well hit the market at the upper end of this spectrum, despite the challenge of the gearbox for such an engine." The gearbox for a 60K engine would have to transmit powers up to double the 32,000 hp of the current unit.

By mid-1999 Pratt had recognised that the PW4000 family was having to meet severe competition from the proliferation of variants of the three-shaft RR Trent and that, despite its large customer base, it would within a decade have to be replaced by a fundamentally newer engine. What has made the position more difficult is the existence of the GP7000 being developed jointly with GE (which see). In the event, by June 1999 Pratt said it had decided to go for the 60K (267 kN) market in order to avoid conflict with the V2500 and compete against the Trent 500 on the A340-500/600, and also to provide a newer alternative to the CF6-80C2.

At the 1999 Paris Air Show the Editor was told that the PW8000 was being redesigned for this general level of thrust. The larger engine was to follow closely along the lines of the original version, but because of the greater financial risk a go-ahead depended upon the assembly of a comprehensive team of risk-sharing partners. MTU of Germany was said to be prepared to risk an increased share - possibly 25-28 per cent - in taking on not only the LP turbine but also the LP compressor. Discussions were also being held with several other possible partners, ``including FiatAvio and other companies at present participating in the PW4000". FiatAvio is a major partner on the gearbox.

Before the redesign, former President Krapek went on record as saying that the PW8000 could be created ``in 30 to 36 months, for an investment of some US$500 million." A year has gone by since then, and the 60K engine is clearly going to cost much more. What has not changed is the basic design objective of ``reducing fuel burn (compared with existing engines) by 10 per cent, cutting noise by 30 dB and reducing aircraft DOC by 8 to 10 per cent". An objective with the original PW8000 was to achieve 180-min ETOPS before EIS, but that presupposed twin-engined aircraft.

Despite the June 1999 choice of a thrust level up to 65K (289 kN), by 2000 all Pratt & Whitney public information on the proposed PW8000 continued to describe this engine as in the 25-35K bracket. Accordingly, the following description refers to the original conception of the PW8000 in the 25-35K thrust class.

Type

High-efficiency geared turbofan.

Fan
Single stage, with 26 shroudless wide-chord blades of solid titanium, developed with 3-D aerodynamic codes. Blades fixed in hub carried on double inclined-roller bearings. Light aluminium fan case with Kevlar wrapping. Structural ring of exit guide vanes followed by A-frame support downstream. Bypass ratio 10.

Gearbox
Single-stage epicyclic (called by Pratt a Star system), with five sets of double-helical star (planetary) gears held in a fixed star carrier. These drive the inner periphery of the surrounding ring gear, which is extended forwards to drive the fan. At take-off, the design shaft speeds are 9,000 rpm on the low spool and 3,200 on the fan, and power transmitted is 23,872 kW (32,000 shp). Two crucial design factors are flexure and heat dissipation. According to Pratt ``It was discovered that up to 80 per cent of the heat load the cooling oil has to handle comes from simply churning in the gearbox. Painstaking work came up with a lubricating system that places the oil only where it is needed and circulates it twice as fast as previous systems, increasing gear efficiency to over 99 per cent". To overcome the problem of flexure, causing severe wear on tooth edges, the entire box is mounted ``on a kind of bellows structure that is, in effect, a large spring". This has the form of inner and outer drums (on the inlet shaft and supporting the large-diameter bearing round the ring gear) with convolutions resembling a barometer aneroid, as shown in the detail drawing, together with an oil feed. Design mechanical efficiency, over 99 per cent. Estimated MTBR 30,000 hours.

LP Compressor
Three stages, with variable inlet guide vanes. Wide-chord aerofoils designed with advanced NASTAR aerodynamics, held in titanium drum EB-welded into one unit. The fact that this is not constrained to rotate at the same speed as the fan, but at up to 9,200 rpm, means that its diameter can be greatly reduced, making the engine lighter and giving a more direct overall gas path. Fan/core splitter of composite material with steel leading-edge.

HP Compressor
Five-stage axial with short, rigid rotor with very high stage loading. Basically that of the XTC66 (see PW6000). Wide-chord 3-D aerofoils with blunt leading-edges for erosion resistance. First two stator stages variable. Ring case construction to minimise gyroscopic deflection and enhance performance retention. Pratt says ``Because of the inherent efficiency . . . the number of stages in the engine can be reduced 40 per cent and the number of airfoils by about half".

Combustion Chamber
`Talon' (second-generation Floatwall) short annular low-emissions, similar to that of the PW6000. Cast one-piece diffuser and easily replaced liner segments of high-strength nickel alloy.

HP Turbine
Single stage. Blades of third-generation single-crystal superalloy with advanced internal cooling management system and advanced thermal barrier coating held in sintered powder-metal disk. Ceramic outer air seal and modulated active clearance control.

LP Turbine
Three stages, counter-rotating, with 3-D aerofoils for high work extraction at high rotational speed. Powder-metallurgy disk, and active clearance control. Diameter significantly less than for a direct fan drive turbine. Transition duct between HP/LP turbines has no aerodynamic fairings.

Jetpipe
Fixed pipe from core only.

Reverser
In fan duct only.

Control System
Fourth-generation FADEC, managing thrust, fuel flow, compressor vane position and engine diagnostics. Equipped with EAGLELink diagnostic system which automatically alerts destination about engine condition via satellite.

Dimensions

Fan diameter	1.93 m (76 in)
Length	3.15 m (124 in)

Weight, Dry

Complete propulsion system	3,629 kg (8,000 lb)

Performance Ratings (S/L, T-O)

	111-156 kN (25,000-35,000 lb st)

Weight	227 kg (500 lb)
Ring-gear pitch diameter	463.6 mm (18.25 in)
Transmitted power	up to 23,862 kW (32,000 shp)
Torque	25,357 N-m (18,700 lb-ft)
Mechanical efficiency	over 99 per cent
MTBR	30,000 hours

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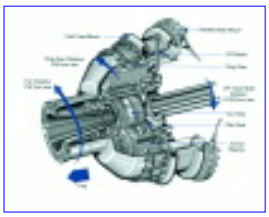
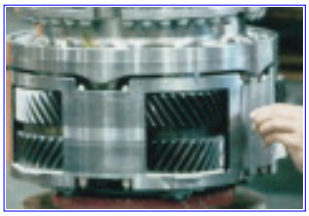
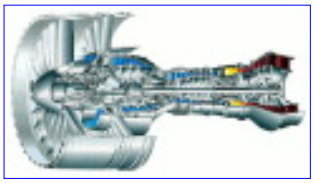


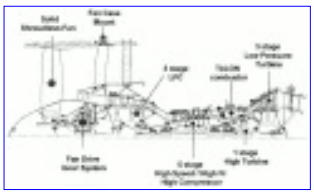
Detail of the PW8000 gearbox
(1999)



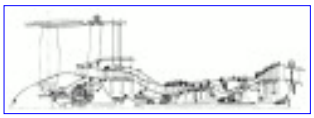
One of the PW8000 test gearboxes
(2000)



Cutaway of PW800
(1998)



PW8000 longitudinal section (1998)



A later PW8000 section drawing showing many changes in detail design (2000)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 30 August 2000

Jane's Aero-Engines 08

UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JT8D-200

The success of the JT3D, and recognition that BPR in the JT8D had been set too low for optimum noise and fuel economy, prompted Pratt & Whitney to undertake a cautious refan programme. This kept the JT8D family competitive for many further years, making it by far the world's most successful airline engine of all time. Though the spares business is huge, production of complete new engines tapered off in 1999 to a toal of 2,856, bringing the combined total of all JT8D engines (excluding the Swedish RM8) to 14,701, a total now surpassed by sales of the CFM56. In early December 1999 total flight time of JT8D-200 versions had reached 68,380,557 hours. In 1994 Pratt calculated that "60 per cent of all flights on existing commercial jet aircraft are made with JT8D engines".

JT8D-209

The first of the refanned engines to be committed to production. It combined the core of the JT8D-9 with a new LP spool derived from the NASA JT8D-refan programme. The new single-stage fan, retained in subsequent versions with increased air flow, has greater diameter and rotates with a new six-stage LP compressor (core booster) to give greater OPR. These and other changes are listed below. A Dash-209 prototype began flight testing on the YC-15 STOL transport on 4 March 1977. Later that month the Dash-200 series was commercially launched, to power the DC-9 Super 80, later redesignated MD-80. FAA certificated June 1979. Rated at 82.2 kN (18,500 lb st) to 25ºC, and 85.6 kN (19,250 lb st) following loss of thrust on any other engine. FPR 1.79, BPR 1.78, OPR 17.1. Entered service in October 1980, powering the MD-81.

JT8D-217

Rated at 88.96 kN (20,000 lb st), and 92.75 kN (20,850 lb st) following loss of thrust on any other engine. FPR

1.88, BPR 1.74, OPR 18.2. Certificated October 1980. Powers MD-82.

JT8D-217A

T-O thrust available to 28.9ºC or up to 1,525 m (5,000 ft). Certificated October 1981. Powers MD-82.

JT8D-217C

Incorporates JT8D-219 performance improvements to reduce sfc. Certificated February 1986. Powers MD-82 and -87.

JT8D-219

Rated at 93.4 kN (21,000 lb st), with a reserve thrust of 96.5 kN (21,700 lb st). FPR 1.91, BPR 1.74, OPR 19.4. Certificated February 1985. Powers MD-83 and other MD-80 aircraft. One of the few catastrophic failures of a Dash-200 engine occurred to the left-hand JT8D-219 of a Delta MD-88 on 6 July 1996; two passengers were killed by uncontained fan-hub failure on the ground. The cause was a fatigue crack from one of the 24 tiebolt holes. The hub had made 13,835 flights since new in 1989, and its failure has been traced back to a manufacturing deviation. P&W says `Dash-200s have never before experienced a single fan crack'.

In September 1996, Pratt & Whitney announced faster support for European operators via a new Dash-200 inventory management system at Alfa Romeo Avio. The Italian firm, one of the largest providers of JT8D overhaul service in Europe, will work through the P&W Aftermarket Center at Maastricht.

In January 1998, Pratt & Whitney and Seven Q Seven of Texas signed an agreement to certify the JT8D-219 as a replacement engine on Boeing 707-320 and KC-135 aircraft. This new application is intended to use new engines, installed in a nacelle derived from that of the MD-80 family with the pylon attachment moved from the side to overhead and with changes to the reverser. Nothing has been said about provisions for noise-suppressing, apart from the comment that the change from the JT3D will offer ``sharply improved emissions and noise characteristics''. The two partners are targeting nearly 300 707-320 family aircraft, including AWACS and J-STARS versions, as well as almost 200 KC-135 tankers, mainly serving with the Air Force Reserves and Air National Guard. First flight of a re-engined aircraft took place six months late in July 1999, and FAA certification has slipped a year to June 2000. One cannot help noticing that the JT8D is an old-technology engine, now to be replaced by the Pratt & Whitney PW6000 (which see). The 707/KC-135 market might have provided a marvellous launch-pad for the new engine.

Type
Two-spool turbofan.

Fan
Single-stage front fan has 34 titanium blades, with part-span shrouds. Mass flow: -209, 213 kg (469 lb)/s; -217 (all), 219 kg (483 lb)/s; -219, 221 kg (488 lb)/s.

LP Compressor
Six-stage axial, integral with fan.

HP Compressor
Seven-stage axial.

Combustion Chamber
Nine can-annular low-emissions burners with aerating fuel nozzles.

HP Turbine
Single stage. Air-cooled blades in -217, -217C and -219.

LP Turbine

Three-stage.

Nozzle
Forced mixer combining core jet and fan air flow, discharged through common nozzle.

Dimensions

Diameter	1,250 mm (49.2 in)
Length	3,911 mm (154 in)

Weight, Dry

JT8D-209	2,056 kg (4,533 lb)
JT8D-217, -217A	2,052 kg (4,524 lb)
JT8D-217C, -219	2,092 kg (4,612 lb)

Performance Ratings

T-O (S/L static): See model descriptions	
Max cruise thrust (10,670 m; 35,000 ft at M0.8):	
JT8D-209	22.0 kN (4,945 lb st)
JT8D-217, -217A, -217C	23.31 kN (5,240 lb st)
JT8D-219	23.35 kN (5,250 lb st)

Specific Fuel Consumption

Max cruise rating, as above:	
JT8D-209	20.50 mg/Ns (0.724 lb/h/lb)
JT8D-217, -217A	21.32 mg/Ns (0.753 lb/h/lb)
JT8D-217C	20.84 mg/Ns (0.736 lb/h/lb)
JT8D-219	20.87 mg/Ns (0.737 lb/h/lb)

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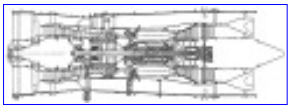
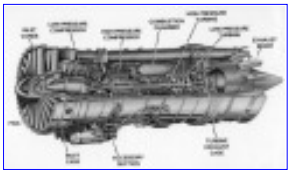
JT8D-219



Longitudinal section through JT8D-200 series (1998)



Cutaway drawing of JT8D-200 series

JT8D-200 series in production

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AERO-ENGINES - TURBOSHAFT, UNITED STATES OF AMERICA

Date Posted: 17 August 2000

Jane's Aero-Engines 08

GE AIRCRAFT ENGINES

GENERAL ELECTRIC T700

The T700 was selected in 1971 to power the US Army's Utility Tactical Transport Aircraft System (UTTAS). The first T700 went to test in 1973, and in 1976 it was the first turboshaft to pass current US military qualification standards. The engine went into production in 1978. The civil versions, designated CT7, are described separately.

T700-GE-700

First production model delivered from early 1978. Following description refers to this version, except where otherwise noted. Powers UH-60A Black Hawk.

T700-GE-701

First-step growth derivative. Powers AH-64 Apache.

T700-GE-401

Navalised first-step growth derivative. Powers SH-60B Seahawk, SH-2G Super Seasprite and AH-1W SuperCobra. In 1999 in production for US Marine Corps UH-1Y (replacing PT6T-3) and AH-1Z

upgrades. To be made under licence by IAR for Romanian-built AH-1RO.

T700-GE-701C

Second-step growth derivative. Went into production in 1989 for Black Hawk and Apache. Offered for future S-70 Black Hawk derivatives for international sales.

T700-GE-401C

Navalised second-step growth derivative. Powers Seahawks and derivatives. First production engines delivered in 1988 to power HH-60H; Royal Australian Navy also a launch customer.

T700/T6A

Step-3 growth engine co-developed with Alfa Romeo Avio and FiatAvio (about 25 per cent each) for EH 101. In January 1998, the Canadian government selected the T700/T6A1 as the engine of its fleet of EH101 Cormorant helicopters. These engines will be assembled and tested by GE Aircraft Engines Canada.

T700/T6E

Growth engine co-developed with Alfa Romeo Avio and FiatAvio (about 25 per cent each) for NH90. In September 1997, MTU joined as co-producer of NH90 engines. Italy selected this engine in September 1999; the other NH90 partners picked the RTM322.

GE has entered into a number of international agreements relative to the sale, production and maintenance of T700 and CT7 engines for helicopters and fixed-wing aircraft. Co-production and maintenance agreements have been struck with Alfa Romeo Avio in Italy, IHI (Ishikawajima-Harima Heavy Industries) in Japan, KAL (Korean Airlines) and SSA (Samsung Aerospace) in Korea, AIDC in Taiwan and HDH (Hawker de Havilland) in Australia; maintenance agreements, with H+S Aviation in the UK, IPTN of Indonesia and FFV (Scania) in Sweden; and sales agreements, with FiatAvio and Alfa Romeo Avio in Italy, Derlan in Canada, and IPTN in Indonesia.

By 30 April 1999, more than 3,300 helicopters were in service powered by T700 engines. These engines had flown more than 13 million hours. Including commercial CT7 versions, over 11,000 engines are in use, having flown over 30 million hours. Mean time between unscheduled shop visits was approximately 2,000 hours for the -700, 3,000 for the -401, 4,000 for the -701 and 5,000 for the -701C.

Type
Ungeared free turbine turboshaft engine.

Intake
Annular, with anti-iced separator designed to remove 95 per cent of sand, dust and foreign-object ingestion. Extracted matter discharged by blower driven from accessory gearbox. T700/T6E has airframe composite bellmouth with primary and secondary screens.

Compressor
Combined axial/centrifugal. Five axial stages and single centrifugal stage mounted on same shaft. Each axial stage is one-piece `blisk' (blades plus disc) in AM355 steel highly resistant to erosion. Inlet guide vanes and first two stator stages variable. Pressure ratio about 15. Mass flow about 4.5 kg (10 lb)/s at

44,720 rpm. T700/T6A and T6E have an uprated compressor, mainly of AM355 and Inco 718, with a zero stage giving a mass flow of approximately 5.9 kg (13.0 lb)/s and pressure ratio of 18.6.

Combustion Chamber
Fully annular. Central fuel injection to maximise acceptance of contaminated fuel and give minimal smoke generation. Ignition power from separate winding on engine-mounted alternator serves dual plugs.

Compressor Turbine
Two stages, with speed at maximum power of 44,720 rpm.

Power Turbine
Two stages with tip-shrouded blades and segmented nozzles. Output speed at maximum power 21,000 rpm.

Accessories
Grouped at top of engine, together with engine control system. Integral oil tank, plus emergency mist lubrication. Torque sensor provides signal to electrical control.

Starting
Airframe-supplied electric or pneumatic, driving clutch/freewheel interface.

Control System
Hydromechanical control can be replaced in less than 12 minutes. Electrical control provides multi-engine speed and torque matching. T6E has Hamilton Standard FADEC.

Dimensions
Length:

T700-700, 701, 401	1,168 mm (46.0 in)
T700/T6A, T6E	1,204.5 mm (47.42 in)
Inlet diameter	385 mm (13.2 in)
Width (all)	Typically 660.4 mm (26.0 in)
Height:	
T700-700, 701, 401	584 mm (23.0 in)
T700/T6A, T6E	632.5 mm (24.90 in)

Weight, Dry (with particle separator)

T700-700	198 kg (437 lb)
T700-401	200 kg (441 lb)
T700-701C	207 kg (456 lb)
T700-401C	208 kg (458 lb)
T700/T6	220 kg (485 lb)

Performance Ratings (ISA, S/L, static):
T700-700:

Intermediate	1,210 kW (1,622 shp)

Continuous	987 kW (1,324 shp)

T700-701:

Contingency	1,285 kW (1,723 shp)
Intermediate	1,266 kW (1,698 shp)
Continuous	1,126 kW (1,510 shp)

T700-401:

Contingency	1,285 kW (1,723 shp)
Intermediate	1,260 kW (1,690 shp)
Continuous	1,072 kW (1,437 shp)

T700-701C, -401C:

Contingency	1,447 kW (1,940 shp)
Intermediate	1,342 kW (1,800 shp)
Continuous	1,239 kW (1,662 shp)

T700/T6A:

Contingency, intermediate	1,491 kW (2,000 shp)
Max continuous	1,327 kW (1,780 shp)

T700/T6E:

Emergency	1,952 kW (2,618 shp)
Max contingency	1,760 kW (2,360 shp)
Intermediate	1,652 kW (2,215 shp)
Max (30 min)	1,641 kW (2,201 shp)
Max continuous	1,539 kW (2,064 shp)

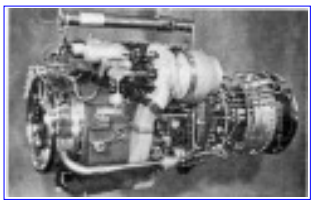
Specific Fuel Consumption (Continuous, as above)

T700-700	79.4 Ág/J (0.470 lb/h/shp)
T700-701	78.7 Ág/J (0.466 lb/h/shp)
T700-401	79.6 Ág/J (0.471 lb/h/shp)
T700-701, -401C	77.6 Ág/J (0.459 lb/h/shp)
T700/T6	74.8 Ág/J (0.443 lb/h/shp)

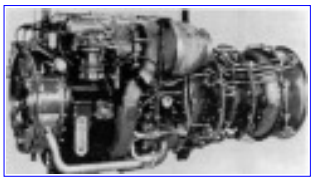
Contract Price

September 1987, US$21,422,782 for 36 T700-701C and US$200+ million for approximately 300 T700-401C/701C (multiyear award 1998-2000); December 1997, 16 T700-701C, US$9,363,888; June 1998, T700/T6A1 for 15 three-engined EH101, C$60 million; September 1998, 48 T700-401, US$32,240,826.

UPDATED



T700-700



T700-401

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Date Posted: 17 August 2000

Jane's Aero-Engines 08

SOLOY - SOLOY CORPORATION

450 Pat Kennedy Way SW, Olympia, Washington 98501-7298
Tel: (+1 360) 754 70 00
Fax: (+1 360) 943 76 59

CHAIRMAN: Joe I Soloy
PRESIDENT: David A Stauffer

Since 1969, Soloy has developed and sold over 500 gas turbine engine modification kits for OEM and aftermarket helicopter and fixed-wing aircraft. Among these are the Hiller 12E and Bell 47 helicopters and the Cessna 206 and 207 aircraft. Soloy's patented Dual Pac combining gearbox technology is used in the Tridair Gemini and Bell Twinranger helicopters and in Soloy's Pathfinder 21™ twin-engine single-propeller modification of the Cessna 208B Caravan.

UPDATED

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 17 December 1999

Jane's Aero-Engines 07

WILLIAMS INTERNATIONAL

WILLIAMS INTERNATIONAL FJ33

In October 1998, Williams announced a programme for a new turbofan, smaller than the FJ44, for commercial applications. The resulting FJ33 is an all-new engine sized to power a rapidly growing new class of light jet aircraft in the 2,268 to 4,082 kg (5,000 to 9,000 lb) MTOW category.

These aircraft will operate from the smallest licensed airports, cruise over the weather and, say Williams, ``have lower operating costs than many twin-piston aircraft''. The FJ33 incorporates technology and lessons learned from the FJ44, as well as manufacturing techniques employed in the NASA General Aviation Propulsion programme. It is designed for outstandingly good fuel economy and low noise.

FJ33-1

T-O rating 5.34 kN (1,200 lb st). In 1999, Williams had announced as customers the Century Aircraft CA-100 Century Jet and the Romanian Aerostar AerostarJet.

TYPE: Two-shaft turbofan.

FAN: Single-stage. Integrally bladed rotor with 20 wide-chord `swept' blades.

LP COMPRESSOR: Multistage core booster.

HP COMPRESSOR: Single-stage centrifugal.
COMBUSTION CHAMBER: Radial delivery from the compressor leads to folded annular combustor with effusion cooling.
HP TURBINE: Single stage, with uncooled inserted blades.
LP TURBINE: Two stages.
JETPIPE: Single fixed-area nozzle handling combined fan and core flows.
CONTROL SYSTEM: HMU or electronic to customer requirement.
DIMENSIONS: See GA drawing.

WEIGHT, DRY:	Less than 136 kg (300 lb).

PERFORMANCE RATINGS: See model listing.
SPECIFIC FUEL CONSUMPTION: Not yet announced.

General arrangement drawing of FJ33-1 (2000)

FJ33-1 (Paul Jackson) (2000)

FJ33 (2000)

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AERO-ENGINES - TURBOSHAFT, UNITED STATES OF AMERICA

Date Posted: 17 December 1999

Jane's Aero-Engines 07

WILLIAMS INTERNATIONAL

WILLIAMS INTERNATIONAL TSX

At the 1999 National Business Aircraft Association show Dr Williams disclosed that turboshaft and turboprop derivatives of the FJX-2 turbofan were in full development. These engines will be in the 224 kW (300 shp) class. The **TSX-1** turboshaft is planned to bring twin-turbine reliability to small helicopters which at present must use a single engine. The **TSX-2** turboprop will be almost identical except for the totally different front end, with a propeller reduction gear and relocated inlet and accessories. The gearbox is to start rig testing in early 2000.

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 26 October 1999

Jane's Aero-Engines 07

GENERAL ELECTRIC - GE AIRCRAFT ENGINES

GENERAL ELECTRIC F404

The F404 was derived from the YJ101 afterburning turbojet. The main difference is a slight increase in diameter of the fan (LP compressor). While the YJ101 powered the Northrop YF-17, the F404 was required to power the F/A-18 Hornet. This aircraft was selected by the US Navy in May 1975 as its VFAX, the next multirole attack fighter.

The first F404 engine test took place in December 1976. The PFRT was completed in May 1978, the first F/A-18 flight was in November 1978, the engine MQT in June 1979 and the first production engine delivery took place in December 1979. The following versions have been announced:

F404-GE-400

Baseline production engine, powering some 1,100 F/A-18A/B/C/D, including exports to Australia, Canada and Spain. Also fitted to Dassault Rafale A, Grumman X-29 and Rockwell/MBB X-31.Maximum rating 49.0 kN (11,000 lb st) dry, 71.2 kN (16,000 lb st) with afterburner. EIS February 1981.

F404-GE-400D

Derivative with plain jetpipe. Max T-O rating 48.0 kN (10,800 lb st). Powered Grumman A-6F prototype. No production contemplated.

F404-GE-100

Derivative with single-engine control system. Powered Northrop F-20A Tigershark.

F404-GE-100D

Derivative with single-engine control system and plain jetpipe. Max T-O rating 49.0 kN (11,000 lb st). Powers SIA A-4SU and TA-4SU Super Skyhawk.

F404-GE-102

Variant of GE-402 for South Korean KT-2. Single-engine features, with specific redundant systems, Leland IDG and new control system with F414-derived FADEC. GE has contracts for FSD and long-term production. Flight testing scheduled for 2001, with production due in 2005.

F404-F1D2

Derivative with plain jetpipe of special type for low observability. Max T-O rating 46.8 kN (10,540 lb st). Powers Lockheed F-117A.

RM12

Uprated engine developed with Volvo Aero of Sweden to power JAS 39 Gripen. Max afterburning rating 80.0 kN (18,100 lb st). Being further developed with radial flameholder, new monocrystal blades and FADEC. See Volvo, Sweden.

F404-F2J3

Higher thrust derivative of RM12, used to flight test Indian Light Combat Aircraft. First LCA flight expected July 1999, 45 months after prototype roll-out.

F404-GE-402

EPE (Enhanced-Performance Engine) combining a 2,000-hour hot-section inspection interval with an approximately 20 per cent increase in thrust ratings. Increased fan speed, improved design and materials

in compressor, turbines and afterburner. TET increased by about 65ºC.Max T-O ratings, see below. First flight July 1991, in production for F/A-18C/D since September 1991 for US Navy and Marine Corps, Finland, Kuwait, Malaysia, Switzerland and Thailand.

By June 1999 over 3,750 F404 engines were in service. At that time, all versions of F404 had flown more than 7.0 million hours. The high-time engine (a -400) was on 7,030 hours, and unscheduled SVR was 1.6 per 1,000 hours.

The following description applies to the F404-GE-402:

TYPE: Two-shaft low-BPR turbofan with afterburner.

FAN: Three stages, with variable inlet guide vanes. Bypass duct of PMR-15 graphite composites (previous versions had a ribbed light alloy duct). Mass flow 66.2 kg (146 lb)/s. Bypass ratio 0.27.

COMPRESSOR: Seven stages, with variable inlet guide vanes and stator stages 1 and 2. Overall pressure ratio 26.

COMBUSTION CHAMBER: Annular, with one-piece machined ring flame tube and casings with new thermal coatings; 18 fuel nozzles.

HP TURBINE: Single-stage with air-cooled single-crystal nozzles and blades.

LP TURBINE: Single-stage, with air-cooled single-crystal nozzles and blades.

AFTERBURNER: Close-coupled, with annular flameholder. Hydraulically actuated variable nozzle. Advanced thermal coatings.

ACCESSORIES: Engine-mounted gearbox with provisions for single PTO (Power Take-Off) and four drive pads for engine alternator, pumps for nozzle control and engine lubrication, main fuel pump and MFC (Main Fuel Control) and afterburner fuel pump and control.

STARTING: Provided from airframe via PTO shaft.

CONTROL SYSTEM: New control system incorporating advanced FADEC. On GE-400 self-contained ECU (Electrical Control Unit) and hydraulic MFC (Main Fuel Control). On GE-100, GE-100D, RM12 and F2J3 the ECU is replaced by a DEC (Digital Engine Control).

DIMENSIONS:

Length	4,030 mm (158.8 in)
Max diameter	880 mm (34.8 in)
WEIGHT, DRY:	1,035 kg (2,282 lb)

PERFORMANCE RATINGS:

Max afterburner	79.0 kN (17,700 lb st)
Intermediate	53.3 kN (11,950 lb st)

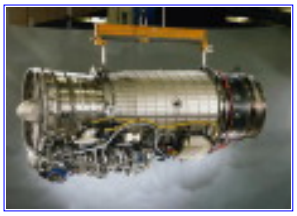
SPECIFIC FUEL CONSUMPTION:

Max a/b	49.29 mg/Ns (1.74 lb/h/lb st)
Intermediate	23.79 mg/Ns (0.84 lb/h/lb st)

CONTRACT PRICE: 36 F404-GE-402 for Finland and 18 for Thailand, US$118,120,093.

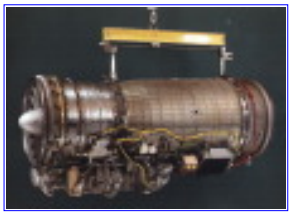
F404-GE-100D (1998)

Cutaway drawing of F404-GE-400 (1996)

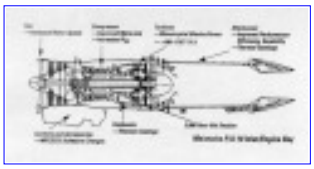


F404-GE-400D (2000)

F404-GE-402 (1996)



F404-F1D2 (jetpipe not shown) (1996)



Longitudinal section showing features of F404-GE-402, which also introduces the lightweight composite bypass duct (1996)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY PW7000

US military designation not yet allocated
This engine for supersonic military aircraft is being derived from the F119. It has a lower thrust rating, and is designed for minimum fuel burn and minimum emissions. It is the intended engine of the Lockheed Martin/Boeing A/F-22X two-seat attack fighter proposed for the US Navy as a replacement for the A-6E. This family of engines is being created around the XTC66 common demonstrator core. Under DOD contract, in late 1998 Pratt & Whitney Large Military Engines began testing its IHPTET Phase II Joint Technology Demonstrator Engine, the XTE66, which forms an initial prototype of the PW7000.

TYPE: Two-shaft augmented turbofan.
FAN: Derived from F119 with low-observable features in inlet and fan and enhanced resistance to birdstrikes.
COMPRESSOR: Derived from the F119 via the XTC66, matched to high pressure ratio in flight at all altitudes.
COMBUSTION CHAMBER: Modified from F119 for minimal emissions and stable combustion on catapult launch.
TURBINE: Single-stage HP and LP turbines rotating in opposition.
NOZZLE: Closely related to F119, with vectoring in vertical plane.

PERFORMANCE RATINGS:

Max T-O (provisional)	112.5 kN (25,300 lb st)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 17 September 1999

Jane's Aero-Engines 06

UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JTF22, F100

US military designation: F100
Stemming in part from the JTF16 demonstrator engine of 1965-66, the original JTF22 design was planned as an outstanding military augmented turbofan for supersonic fighters. Basic development was funded as a demonstrator programme for the US Air Force. In February 1970, this engine's core was selected as the basis for the engine for the winner of the FX programme, which became the F-15. Subsequently the following versions have been developed:

F100-PW-100 (JTF22A-25A)

Original engine of the twin-engined F-15A and F-15B. Some 3,000 hours of testing were accomplished between 1968 and the 60 hour PFRT in February 1972. The F-15 first flew on 27 July 1972. The 150 hour MQT was completed in October 1973. By 1980, 1,200 engines had flown 420,000 hours and TBO had reached 1,250 hours. By 1985, over 3,500 engines had flown nearly 2 million hours, and were cleared for 1,800 mission cycles without hot-section refurbishment.

F100-PW-200

Selected for the single-engined F-16, with back-up fuel control, modified control system cooling and a forward extension of the compressor inlet. Following loss of an F-16 on 26 March 1999 the USAF ``found hairline cracks in the engines of 17 F-16s at Luke AFB''. Pending rectification, all USAF F-16s powered by this engine were grounded. In May 1999 Pratt & Whitney was awarded a US$26,097,265 USAF contract for 19,555 second-stage stator vanes for Dash-200 and -220 engines, to be delivered at 1,200 per month.

F100-PW-220

Partly redesigned to incorporate later technologies, including digital electronic control, new materials and special coatings and heat-transfer concepts. Entered Alternate Fighter Engine competition for combined future F-15 and F-16 production. Entered service in October 1985 after an unprecedented 4,000-cycle Accelerated Mission Test, equivalent to nine years' normal operation. Powered all F-15C and F-15D aircraft and some F-16C and F-16D. In September 1997, Pratt & Whitney was awarded a US$399,900,416 fixed-price contract `to provide for 108 types of rotating fracture-critical components' for USAF-200/-220 engines. Under the F-16 CCIP (Common Configuration Improvement Program) the USAF was in June 1999 seeking an additional US$117 million, part of which would buy longer-life fan blades and an uprated augmentor duct giving increased thrust.

F100-PW-220E

Earlier engines converted by kit to provide all the advanced features of the PW-220. E stands for Equivalent. The USAF plans to upgrade all PW-100 and PW-200 engines during the 1990s. In June 1997, Pratt & Whitney was awarded a US$485 million contract to upgrade earlier USAF engines to this standard.

F100-PW-220LE

A further upgrade with improved materials, cooling and other features for extended life and reduced maintenance.

F100-PW-220P

A major upgrade of earlier engines to incorporate as much F100-PW-229 technology as possible, notably including the advanced fan, FADEC and new augmentor fuel management. Other designations for such engines are F100-PW-200+ and F100-PW-220E+.

F100-PW-229

This version represents an almost complete redesign to keep the F100 competitive through the 1990s. Major upgrades include an advanced fan and compressor handling greater mass flow with higher pressure ratio, increased TET, and many features giving better maintainability and longer life. Thanks to a shorter combustion chamber, the longer compression system was accommodated within the original frame size, making the PW-229 a bolt-on replacement for earlier versions. Qualification was completed in 1989, and the PW-229 entered operational service on the F-15 in 1991 and on the F-16 in 1993. On 28 May 1997, testing began of the VISTA/F-16D, the first to have the modular common inlet duct (also suitable for the F110). In April 1998, a contract was awarded to Pratt & Whitney's San Antonio office in the exceptional sum of US$458,882,277 for repair and overhaul of various PW-229 modules on USAF F-16 aircraft. When funding permits, the PW-229 in this aircraft will be fitted with a PYBBN (see MILITARY ENGINES `Various Nozzles`).

F100-PW-229A

Originally known as the PW-229IPE+ (Increased Performance Engine), the PW-229A has been intensively developed under Program Manager Bill Gostic to be ``an increased-thrust, affordable and exportable engine for the F-15 and F-16''. The main new feature is a redesigned fan (see drawing) based on F119 technology. This has a large-blade FOD-resistant first stage (but without the F119's sensitive-technology hollow blades) and wide-chord blisk second and third stages. This handles increased airflow, further enhanced by eliminating the variable inlet guide vanes of previous F100 engines. The IPE+ was intended to retrofit any F-15 or F-16, but the PW-229A requires larger airflow, as indicated by the figures below for Max T-O ratings. To keep overall engine length constant, the augmenter has been shortened, and according to Gostic the improved engine is designed for either an all-axis vectoring nozzle or an ejector nozzle with reduced IR signature.

The F100 has been co-produced under three licence agreements. Companies in Belgium, Denmark, Netherlands and Norway have produced PW-200 engines since 1976. In Japan, IHI, assisted by KHI, produces the PW-100 and -220E. In South Korea, Samsung leads a group which will produce the PW-229. In 1998, shipments of F100 engines exceeded 6,000, with a total flight time (all in fighters) exceeding 11 million hours.

TYPE: Two-shaft augmented turbofan.
INTAKE: Fabricated from titanium with fixed nose bullet (hemispherical in -100 and -200, conical in -220 and -229, small in IPE) carried on an integrally anti-iced vane frame. In -100 and -200, 21 IGVs with variable trailing flaps; in -220 and -229, 26 perforated VIGVs. Inlet diameter (all) 884 mm (34.8 in).
FAN: Three stages, with rotor blades solid titanium. In -100 to -220, all three rotors have blades of Ti 8-1-1 with part-span shrouds, in discs of Ti 6-2-4-6. Fan diameter 928 mm (36.5 in), max speed 10,400 rpm, BPR (-100) 0.7, (-220) 0.6. PW-229 has new blades with advanced aerodynamics, with part-span shrouds on the first two stages, held in a rotor EB-welded in 6AI-4V alloy, with a third stator ring added at the rear. BPR 0.36. IPE+ has further increased air flow and efficiency, see diagram below.
COMPRESSOR: PW-100 to -220, 10 stages, with variable stators over the first three stages. Discs 1-3, forged Ti; 4, forged PWA1016; 5, 7, 9, PWA1027; 6, 8, 10, Gatorised (isothermally squeeze-forged) IN-100. Blades 1-3, Ti 8-1-1; 4, Ti 6-2-4-6; 5-9, Incoloy 901; 10, PWA1005. Max speed 13,450 rpm, pressure ratio 8, giving OPR 24.5. PW-220P, pressure ratio 28.5. PW-229, a total redesign with new aerofoils in damage-tolerant Ti and superalloy discs inertially welded together. Variable stators in first four stages. Coated knife-edge and abradable tip seals. Split case facilitating blade replacement. OPR 32.2.
COMBUSTION CHAMBER: Annular, fabricated of nickel alloy with film cooling throughout. Large diameter

air-blast fuel nozzles. Capacitor-discharge ignition. PW-229 has a shorter chamber with a Floatwall flame tube, aft-mounted to improve the loadpath and ensure good sealing at the nozzle diaphragm. The flow splitter on earlier versions - modified to ameliorate stall-stagnation problems - is eliminated by advanced diffuser design.

HP TURBINE: Two stages. Discs forged in damage-tolerant IN-100. Blades and vanes (PW-100, -200) PWA1422 DS alloy with PWA73 coating, first stage impingement cooled, second convective (HP bleed), TET 1,399ºC; (PW-220) blades and vanes DS Mar-M200/Hf alloy with aluminide coating, first rotor transpiration convective, second HP bleed, same TGT; (PW-229) both stages monocrystal PWA1484, held in boltless discs, with outer airseal segments PW1485, increased TET.

LP TURBINE: Two stages, with discs forged IN-100. (PW-100, -200) blades uncooled cast IN-100 with PWA73 coating. (PW-220) blades cast IN-100 with aluminide coating. (PW-229) larger flow area, work load shifted to first stage (70/30 split), uncooled monocrystal PW1484, second IN-100.

AFTERBURNER: In all versions, high-intensity with outer duct fabricated in sheet steel and Ti stringers, with corrugated liner of perforated Haynes 188. (PW-100, 200, 220) five concentric spray rings with downstream flameholder assembly. High-energy capacitor ignition giving modulated light-up. Multiflap balanced-beam articulated nozzle; (PW-229) redesigned to reduce (almost eliminate) the transient pressure pulse seen by the fan on light-up, with seven spray rings and 11 combustion zones in place of five; internal pressure 20 per cent higher and temperature increased. New con/di nozzle with parts cast instead of fabricated. In `the late 1990s', the pitch/yaw balanced-beam nozzle should enter service. This retains all variable profile/area functions, whilst vectoring up to 20º in any direction (lateral force up to 1,814 kg, 4,000 lb) at up to 120º/s even in max a/b. The IPE+ is designed for vectoring, with integrated engine/airframe control.

CONTROL SYSTEM: (PW-100) unified hydromechanical fuel and nozzle control, with electronic supervisory control; (PW-200) added manual back-up fuel control; (PW-220) digital electronic control and a new gear-type main fuel pump; (PW-229) FADEC for engine and augmentor, giving `carefree' single-lever control and even faster snap acceleration.

DIMENSIONS:

Overall diameter (all)	1,181 mm (46.5 in)
Length (except PW-229A)	4,855 mm (191.2 in)
F100-PW-229A	4,844 mm (190.7 in)

WEIGHT, DRY:

F100-PW-100	1,428 kg (3,149 lb)
F100-PW-200	1,447 kg (3,190 lb)
F100-PW-220	1,452 kg (3,200 lb)
F100-PW-220E	1,467 kg (3,234 lb)
F100-PW-220P	1,544.5 kg (3,405 lb)
F100-PW-220LE	1,532 kg (3,378 lb)
F100-PW-229 (F-15)	1,680.5 kg (3,705 lb)
F100-PW-229 (F-16)	1,699 kg (3,745 lb)
F100-PW-229A	1,844 kg (4,065 lb)
F100-PW-229A (vectored)	1,860 kg (4,100 lb)

PERFORMANCE RATINGS:

Max T-O:

F100-PW-100, -200	100.53 kN (22,600 lb st)
F100-PW-220, -220E	105.72 kN (23,770 lb st)
F100-PW-220P	120.01 kN (27,000 lb st)
F100-PW-220LE	105.4 kN (23,700 lb st)
F100-PW-229	129.45 kN (29,100 lb st)
F100-PW-229A (F-16)	129.45 kN (29,100 lb st)
F100-PW-229A (F-16 `big mouth inlet')	142.0 kN (31,860 lb st)
F100-PW-229A (ideal inlet)	165.0 kN (37,100 lb st)

Max dry:

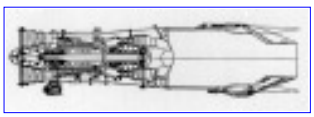
F100-PW-100, -200	65.26 kN (14,670 lb st)
F100-PW-220, -220LE	63.9 kN (14,370 lb st)
F100-PW-220E	64.9 kN (14,590 lb st)
F100-PW-220P	74.29 kN (16,700 lb st)
F100-PW-229, -229A	79.18 kN (17,800 lb st)
F100-PW-229A (ideal inlet)	up to 98 kN (22,000 lb st)

SPECIFIC FUEL CONSUMPTION:

Max T-O:

F100-PW-100, -200, -220, -220E	59.49 mg/Ns (2.10 lb/h/lb st)
F100-PW-220P	56.9 mg/Ns (201 lb/h/lb st)
F100-PW-220LE	59.20 mg/Ns (2.09 lb/h/lb st)
F100-PW-229	54.96 mg/Ns (1.94 lb/h/lb st)
F100-PW-229A	50.99 mg/Ns (1.80 lb/h/lb st)

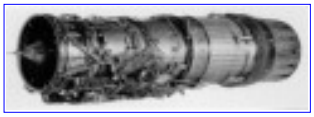
F100-PW-100 (1996)



Longitudinal section through F100-PW-100 (1996)

Cutaway drawing of F100-PW-100 (1996)

F100-PW-200 (1996)



F100-PW-220E (1996)

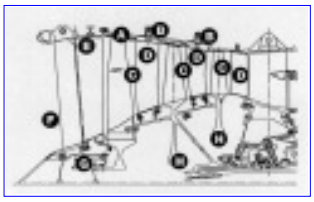
F100-PW-229 (1999)

Cutaway drawing of F100-PW-220 (1996)

Cutaway drawing of F100-PW-229 (1996)

Longitudinal section through F100-PW-229 (1996)

Cutaway drawing of F100-PW-229A/IPE+ (1996)



Features of the fan of the F100-PW--229A (a) active tip clearance control treatment; (b) on-wing, blending borescope ports; (c) FOD tolerant airfoils; (d) F-119-type bowed stators; (e) F119-type composite Inlet Guide Vane flap; (f) F119 2.5 lb bird-tolerant design; (g) F119 oil damped No 1 bearing; (h) F119-type second and third stage titanium Integrally Bladed Rotors (1996)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Date Posted: 17 September 1999

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UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JT3D

US military designation: TF33
By a remarkable mixture of good fortune and good engineering this highly competitive pioneer turbofan was produced in a very short time in 1959 by modifying the LP system of the JT3C turbojet. The first three stages of the LP compressor were replaced by a two-stage fan, and the second stage of the LP turbine was enlarged and a third stage added. The result was an engine handling 2.5 times the T-O mass flow, giving 1.5 times the T-O thrust, with dramatically reduced specific fuel consumption and noise.

Flight testing began with a B-52 and a 707 in 1960. The following are the main versions:

JT3D-1

Commercial engine rated at 75.56 kN (17,000 lb st). Original engine of Boeing 720B, 707-120B and DC-8-50. No longer operating.

JT3D-3B

Commercial engine rated at 80 kN (18,000 lb st). Option of water injection. Powers DC-8-50, DC-8-61 and Northrop Grumman E-8 Joint-STARS. The Quiet Technology Venture Stage-3 hushkit for the DC-8-50 was certified in June 1997, with that for the DC-8-61 being developed.

JT3D-7

Commercial engine rated at 84.4 kN (19,000 lb st). Option of water injection. Powers Boeing 707-320B and -320C and DC-8-62 and -63. Also powered Shanghai Y-10 prototypes.

TF33-P-3

Military engine rated at 75.56 kN (17,000 lb st). Powers B-52H, in twin-engine nacelles requiring fan ducts to discharge through curved `banana nozzles' on outboard side only.

TF33-P-7, P-7A (JT3D-8A)

Military growth version with a 16th stage of compression. T-O rating 93.4 kN (21,000 lb st). Powers Lockheed C-141A and B.

TF33-PW-100A (JT3D-8B)

Military engine rated at 93.4 kN (21,000 lb st). In January 1973 this engine was selected to power the Boeing E-3A. It is based on the TF33-7, with additional drives for high-power alternators.

TF33-PW-102, PW-102A

Military engine based on modernised JT3D-3B. Powers C-135E, KC-135E and C-18A. In 1997-2001 P&W will deliver 199 new inlet-guide-vane cases against a US$30.4 million contract.

TF33-PW-102B

Derived from PW-100A to power E-8C Joint-STARS surveillance aircraft.

TF33-PW-102C

Improved configuration for E-8C Joint-STARS, still with T-O rating of 80 kN (18,000 lb st) but augmented electrical output and new features. In 1999 Pratt & Whitney was redelivering 42 engines modified from PW-102B standard with kits purchased on a US$10,541,538 contract.

Production of new JT3D and TF33 engines was completed in 1984. Total shipments exceeded 8,550. In August 1997 Pratt & Whitney was awarded a US$154,750,001 contract for spare parts for USAF

engines.

Since 1990 BAC (Burbank Aeronautical Co) has been producing FAA-certified Stage-3 hushkits for the JT3D-3B and -7 engines of DC-8s. It expected by December 1996 to achieve FAA-certification for a similar kit for the Boeing 707, with production planned to follow immediately. BAC claims that its hushkit `does not add flight restrictions nor increase aircraft fuel burn'. Details of a competitor Stage-3 hushkit, by Quiet Nacelle Corporation, appear in the feature article `Environmental factors'.

TYPE: Two-shaft turbofan.
FAN: Two stages, with solid titanium blades with part-span shrouds, free to rock slightly in fir-tree sockets. Typical mass flow 209 kg (460 lb)/s. Typical bypass ratio 1.36.
LP COMPRESSOR: Six-stage spool, rotating with fan. Typical core mass flow 88.4 kg (195 lb)/s.
HP COMPRESSOR: Seven stages, originally identical with contemporary JT3C/J57 (TF33-7 and 100A, eight stages). Overall pressure ratio, (JT3D-3B, TF33-3) 13.0, (JT3D-7, TF33-102) 13.6, (TF33-7, -100A) 15.6.
COMBUSTION CHAMBER: Can-annular; basically identical to JT3C/J57.
HP TURBINE: Single-stage, identical to JT3C/J57. TET 871-930ºC.
LP TURBINE: Three stages bolted into single unit.
FAN DUCT: In TF33-3, single short duct discharging through curved `banana' nozzle on one side of the engine only. Most versions, short annular duct incorporating airframe-mounted reverser. TF33-7 and PW-100A, full-length duct, without core mixing.
ACCESSORIES: Grouped under HP compressor case, driven off HP spool via radial shaft.
STARTING: Usually pneumatic.
CONTROL SYSTEM: Hydromechanical with automatic starting and acceleration.
DIMENSIONS:

Length:

JT3D-3B, -7	3,429 mm (135 in)
TF33-3	3,454 mm (136 in)
TF33-PW-102A	3,480 mm (137 in)
TF33-7, PW-100A	3,607 mm (142 in)

Diameter:

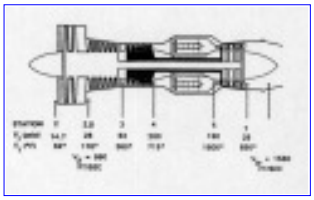
JT3D-3B, -7, TF33-3, PW-102A	1,346 mm (53 in)
TF33-7, PW-100A	1,372 mm (54 in)

WEIGHT, DRY:

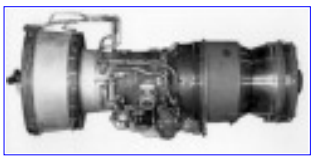
JT3D-3B, -7	1,932 kg (4,260 lb)
TF33-3	1,769 kg (3,900 lb)
TF33-7	2,088 kg (4,605 lb)
TF33-7A	2,109 kg (4,650 lb)
TF33-100A	2,173 kg (4,790 lb)
TF33-102A	1,969 kg (4,340 lb)

PERFORMANCE RATINGS: See model listing
SPECIFIC FUEL CONSUMPTION (T-O, S/L):

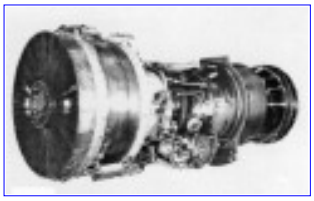
JT3D-3B	15.15 mg/Ns (0.535 lb/h/lb st)
JT3D-7, TF33-7, PW-100A	15.85 mg/Ns (0.560 lb/h/lb st)
TF33-3	14.72 mg/Ns (0.52 lb/h/lb st)
TF33-PW-102A	15.29 mg/Ns (0.54 lb/h/lb st)



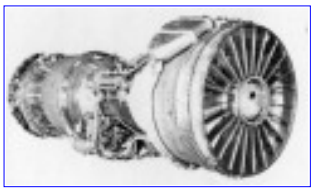
Typical pressures and temperatures in JT3D (18,000 lb rating) (1996)



JT3D-1 prototype, January 1960 (1996)



JT3D-3B (1996)



TF33-P-7 (1996)

Cutaway drawing of JT3D-3B (1997)

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Date Posted: 17 September 1999

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UNITED TECHNOLOGIES PRATT & WHITNEY

International participation

Several of the most important Pratt & Whitney engines involve partnerships with companies in other countries. Details are given in the table Collaborative Programmes in the Introductory features.

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UNITED TECHNOLOGIES PRATT & WHITNEY

V2500

Pratt & Whitney has a 33 per cent share in this civil turbofan. It is described under IAE in the International I-O section.

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UNITED TECHNOLOGIES PRATT & WHITNEY

JSF

In addition to the entry on the F119 details of JSF propulsion are given under JSF in the International I-O section.

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UNITED TECHNOLOGIES PRATT & WHITNEY

GP7000

This civil turbofan family is a 50/50 programme with GE Aircraft Engines. It is described in the USA section under Engine Alliance.

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GENERAL ELECTRIC - GE AIRCRAFT ENGINES

International Participation

Several of the most important GE engines involve partnerships with companies in other countries. Details are given in the table Collaborative Programmes in the introductory features.

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GENERAL ELECTRIC - GE AIRCRAFT ENGINES

JSF

In addition to the entry on the F120, details of GE participation in the JSF programme are given under the heading JSF in the International I-O section.

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GENERAL ELECTRIC - GE AIRCRAFT ENGINES

GP7000

This civil turbofan family is a 50/50 programme with Pratt & Whitney. Details are given in this (USA) section under Engine Alliance.

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GENERAL ELECTRIC - GE AIRCRAFT ENGINES

CFMI CFM56

Details of this family of turbofan engines are given under CFMI in the International I-O section.

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GENERAL ELECTRIC - GE AIRCRAFT ENGINES

CFE738

The civil turbofan is a 50/50 programme with AlliedSignal. Details are given in this (USA) section under CFE.

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AERO-ENGINES, UNITED STATES OF AMERICA

Jane's Aero-Engines 05

UNITED TECHNOLOGIES PRATT & WHITNEY

TF33

see JT3D

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AERO-ENGINES, UNITED STATES OF AMERICA

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UNITED TECHNOLOGIES PRATT & WHITNEY

TF30

see JTF10A

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UNITED TECHNOLOGIES PRATT & WHITNEY

T73

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Jane's Aero-Engines 05

UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JTF10A, TF30

US military designation: TF30
Development of this high-compression two-spool turbofan began in 1958 as a private venture, and resulted in testing of the first turbofan with afterburning. It was chosen subsequently as the power plant for the General Dynamics F-111, Grumman F-14 and LTV A-7 Corsair II. Numerous versions were produced, of which the following remain operational:

TF30-P-3, P-103 (JTF10A-21)

Afterburner. Powers F-111A, F-111C and modifications.

TF30-P-7, P-107 (JTF10A-27D)

Afterburner. Powers FB-111A.

TF30-P-108

Afterburner. Improved long-life engine for remaining RAAF F-111 aircraft.

TF30-P-9, P-109 (JTF10A-36)

Afterburner. Powers F-111E and EF-111A.

TF30-P-100, P-111

Afterburner. The ultimate mainstream development. Powers F-111F.

TF30-P-408 (JTF10A-16A)

No afterburner. Powers A-7P and TA-7P.

TF30-P-414, P-414A

Afterburner. Developed from P-412 with new first-stage compressor blades and redesigned surrounding case to ensure containment. TBO 2,400 hours. Powers F-14A.

TYPE : Two-shaft axial-flow turbofan.

INTAKE : Direct pitot annular type with 23 fixed inlet guide vanes (19 on P-408). Hollow vanes pass anti-icing air.

FAN : Three stages (two on P-408). Rotor and stator and casings all of titanium, except for steel containment case on P-414. Three rotor stages have 28 (with part-span shrouds), 36 and 36 blades, all dovetailed; stator stages have 44, 44 and 48 vanes, all rivet-retained. Pressure ratio 2.14 to 2.4. Mass flow typically 112 kg (247 lb)/s (P-100 118 kg; 260 lb/s).

LP COMPRESSOR : Six stages (seven on P-408), constructed integrally with fan to form nine-stage spool. Wholly of titanium construction, except stator blades of steel. Rotor blades dovetailed, stator vanes pinned butted and pinned.

HP COMPRESSOR : Seven stages, constructed mainly of nickel-based alloy. Rotor blades dovetailed, stator vanes pinned. Overall pressure ratio, (P-103) 17.1, (P-107) 17.5, (P-109) 18.1, (P-111) 22.0, (P-408) 15.9, (P-414A) 19.8.

COMBUSTION CHAMBER : Can-annular, with steel casing and eight Hastelloy X flame cans each held at the front by four dual-orifice burners. Spark igniters in chambers 4 and 5.

HP TURBINE : Single stage, with air-cooled nozzle guide vanes (stators) of cobalt alloy and air-cooled rotor blades of nickel-based alloy (P-100 vanes and blades of directionally solidified alloy, P-414A vanes single-crystal). Max gas temperature, early models, 1,137ºC, P-100 1,240ºC.

LP TURBINE : Three stages of nickel-based alloys. Rotor stages have 94, 96 and 80 (P-100, 72) fir-tree root blades. Gas temperature after turbine, typically 587ºC.

JETPIPE (non-a/b engine): Simple steel pipe where fan air flow and core gas mix before passing through fixed nozzle.

AFTERBURNER : Diffuser leads to combustion section comprising double-wall outer duct and inner liner carrying five-zone combustion system with three spraybars and rings upstream of flameholder. Fuel supplied by hydraulic pump up to 1,814 kg (4,000 lb)/h, and at higher rates by special centrifugal pump. Ignition by auxiliary squirt in a/b diffuser, coupled with main squirt in No 4 burner can which produces

hot-streak of fuel through the turbine (P-100 engine, fully moduated light-up by 4-joule electrical ignition system). Max gas temperature 1,677ºC.

NOZZLE (a/b engines): Primary nozzle has variable area, with six hinged segments actuated by engine-fuel rams (P-100 and P-414A, 18 iris segments translated along curved profile by six long-stroke rams). Ejector nozzle has six blow-in doors with free tail-feathers (P-414, 18 iris segments and no blow-in doors).

MOUNTING : Two-planar. Front peripheral pair of flanges absorb vertical, side and thrust loads; rear pair of peripheral flanges (in line with No 6 bearing behind LP turbine) absorb vertical and side loads.

ACCESSORIES : Main gearbox under compressor, driven by bevel shaft from HP spool. Contains major elements of lubrication and breather systems. Drive pads at front and rear for main and a/b fuel pumps, main oil pump, N_2 tachometer, starter, fluid power pumps and power take-off.

STARTING : Air-turbine starter on left forward drive pad of accessory gearbox.

CONTROL SYSTEM : HP system (above 69 bars; 1,000 lb/sq in), with conventional hydromechanical control. Main elements comprise fuel pump, filter, fuel control, P & D valves and nozzles. Separate afterburner system for a/b engines. No water injection.

FUEL SPECIFICATION : JP-4, JP-5, JP-8.

OIL SYSTEM : Self-contained dry-sump hot-tank system. Accessory gearbox housing forms 15 litre (4 US gallon; 3.3 Imp gallon) tank. Oil circulated at 3.10 bars (45 lb/sq in) through pump, filter, coolers (air/oil on airframe, fuel/oil on engine and a/b fuel/oil cooler) and three main bearing components; returned by scavenge pumps and de-aerator.

OIL SPECIFICATION : MIL-L-7808, MIL-L-23699.

DIMENSIONS :

Length:

TF30-P-103, P-108, P-111	6,139 mm (241.7 in)
TF30-P-107, P-109	6,116 mm (240.8 in)
TF30-P-408	3,254 mm (128.12 in)
TF30-P-414A	5,987 mm (235.7 in)

Diameter:

TF30-P-103, P-111	1,242 mm (48.88 in)
TF30-P-107, P-108, P-109, P-414A	1,293 mm (50.9 in)
TF30-P-408	1,068 mm (42.06 in)

WEIGHT, DRY :

TF30-P-103	1,850 kg (4,079 lb)
TF30-P-107	1,880 kg (4,144 lb)
TF30-P-108	1,890 kg (4,166 lb)
TF30-P-109	1,862 kg (4,105 lb)
TF30-P-111	1,824 kg (4,022 lb)
TF30-P-408	1,178 kg (2,597 lb)
TF30-P-414A	1,928 kg (4,251 lb)

PERFORMANCE RATINGS :

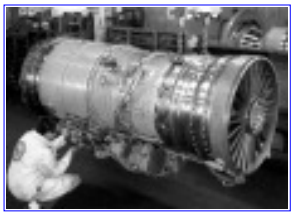
Max T-O:

TF30-P-103	82.3 kN (18,500 lb st)
TF30-P-107, P-108	90.52 kN (20,350 lb st)
TF30-P-109	92.7 kN (20,840 lb st)
TF30-P-111	111.7 kN (25,100 lb st)
TF30-P-408 (no afterburner)	59.6 kN (13,400 lb st)
TF30-P-414A	93.0 kN (20,900 lb st)

SPECIFIC FUEL CONSUMPTION :

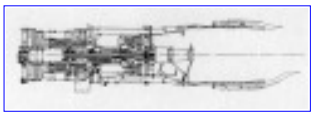
T-O, as above:

TF30-P-103	70.8 mg/Ns (2.5 lb/h/lb st)
TF30-P-107	85.26 mg/Ns (3.01 lb/h/lb st)
TF30-P-108	76.48 mg/Ns (2.70 lb/h/lb st)
TF30-P-109	74.22 mg/Ns (2.62 lb/h/lb st)
TF30-P-111	69.40 mg/Ns (2.45 lb/h/lb st)
TF30-P-408	19.39 mg/Ns (0.64 lb/h/lb st)
TF30-P-414A	78.75 mg/Ns (2.78 lb/h/lb st)



F30-P-6 predecessor of the P-408 in the A-7A (1996)

TF30-P-414 (1996)



Longitudinal section through TF30-P-414 (upper half, afterburner in operation) (1996)

Cutaway of TF30-P-111 (1996)

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UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JT12

US military designation: J60

Design of this small turbojet began in July 1957 at United Aircraft of Canada in Montreal (today Pratt & Whitney Canada). The prototype (J60-P-1, with USAF/Navy funding) ran in May 1958. In July 1959, it was delivered for the prototype Canadair CL-41 at a rating of 9.14 kN (2,900 lb st). The project was then transferred to the parent company in Connecticut, where the JT12 was further developed and also used as the basis for the JFTD12A (T73) free-turbine turboshaft engine to power the Sikorsky S-64 (CH-54 Tarhe) helicopter. Active JT12 versions include:

JT12A-5 (J60-P-3/5/6)

T-O rating 13.35 kN (3,000 lb st). J60-P-3 powers Rockwell T-39 versions. J60-P-6 powered Rockwell T-2B Buckeye.

JT12A-6A

T-O rating 13.35 kN (3,000 lb st). Powered Lockheed JetStar.

JT12A-7 (J60-P-4)

T-O rating 14.69 kN (3,300 lb st).

JT12A-8

T-O rating 14.69 kN (3,300 lb st). Powers Lockheed Dash-8 JetStar and Rockwell Sabreliner 40, 60 and 75, and derived T-39 versions.
TYPE : Single-shaft turbojet.
COMPRESSOR : Nine stages. Pressure ratio 6.5.
COMBUSTION CHAMBER : Can-annular type, with single burner nozzle in each flame tube.
TURBINE : Two stages, driving interconnected front and rear shafts running in three bearings.

DIMENSIONS :

Length	1,981 mm (78.0 in)
Diameter	556 mm (21.9 in)

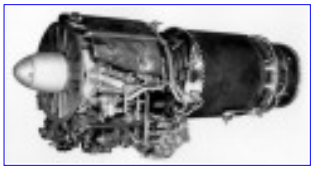
WEIGHT, DRY :

J60-P-3	203 kg (448 lb)
J60-P-6	225 kg (495 lb)
JT12A-6A	206 kg (453 lb)
JT12A-8, J60-P-4	212 kg (468 lb)

PERFORMANCE RATINGS :
T-O, S/L: See model listing
SPECIFIC FUEL CONSUMPTION :
T-O, as above:

J60-P-3, P-6, JT12A-6A	27.16 mg/Ns (0.96 lb/h/lb st)
JT12A-7, A-8, J60-P-4	28.15 mg/Ns (0.995 lb/h/lb st)

J60-P-4 (1996)


JT12A-6A (1996)

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AERO-ENGINES, UNITED STATES OF AMERICA

Jane's Aero-Engines 05

UNITED TECHNOLOGIES PRATT & WHITNEY

F117

see PW2000

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UNITED TECHNOLOGIES PRATT & WHITNEY

F105

see JT9D

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UNITED TECHNOLOGIES PRATT & WHITNEY

F100

see JTF22

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GENERAL ELECTRIC - GE AIRCRAFT ENGINES

F120

see YF120

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

Jane's Aero-Engines 05

GENERAL ELECTRIC - GE AIRCRAFT ENGINES

GENERAL ELECTRIC F118

This turbofan was designed specifically to meet the unique propulsion requirements of the USAF's Northrop Grumman B-2A Advanced Technology Bomber. It was derived from the F101 and F110, but with important modifications. Compared with the F110, it has increased mass flow and a higher pressure ratio, resulting in a higher dry thrust (no afterburner is fitted).

The technology of the B-2A remains classified. Speculation regarding this aircraft and its engines suggests unique use of exceedingly powerful electrostatic fields for both propulsive and `invisibility' reasons. This speculation is unofficial, and not confirmed by General Electric Aircraft Engines.

F118-GE-100

Baseline F118 for B-2A, described below. To reduce cost, maximum use was made of F101/F110 production tooling. This engine was first run in 1982, qualified in 1987, flying in the B-2A from 17 July 1989, and delivered to the USAF from December 1993. The present programme envisages 21 operational aircraft, 16 to be shared equally between 393 and 715 BS of 509 BW. Production terminated 1997.

F118-GE-101

Modified for flight at high altitudes. Powers U-2S, as a retrofit replacing the old technology J75-P-13B.

By 6 August 1997, 29 aircraft were flying with F118 engines, which had then logged 35,175 hours in operational service; the high-time engine had then logged 1,762 hours. Unscheduled SVR was then 1.68 per 1,000 engine flight hours.

TYPE : Unaugmented two-shaft turbofan.

FAN : Three stages, with one row of inlet guide vanes with variable-incidence trailing flaps. Reduced number of rotor blades of exceptionally long chord. Construction generally of titanium, with casing in upper/lower halves. Bleed air de-icing for F118-100.

COMPRESSOR : Nine stages, virtually identical with F110. Overall pressure ratio, F118-100, 35.1; F118-101, 32.2.

COMBUSTION CHAMBER : Annular, scroll type, based on F110. Modified for minimum emissions. Much research undertaken to reduce contrail formation.

HP TURBINE : Single stage, similar to F110, with air-cooled single-crystal rotor blades.

LP TURBINE : Two stages modified from F110.

FAN DUCT : Integrally ribbed metal in upper/lower halves.

JETPIPE : Details classified. In the B-2A the jets are cooled by mixing with a secondary air flow and appear to emerge across areas of black carbon composite.

ACCESSORIES : Grouped along the entire length of the engine's underside, driven by a radial shaft from the HP spool.

CONTROL SYSTEM : FADEC, derived from F110-GE-100B. In 1998, Hamilton Standard was awarded a contract to replace the existing system by an upgraded FADEC. The 120 new units are to be delivered from third quarter 2002.

DIMENSIONS :

Length:

F118-100	2,553 mm (100.5 in)
F118-101	2,794 mm (110.0 in)

Overall diameter :

F118-100	1,180 mm (46.5 in)
F118-101	1,194 mm (47.0 in)

WEIGHT, DRY :

F118-100	1,452 kg (3,200 lb)
F118-101	1,429 kg (3,150 lb)

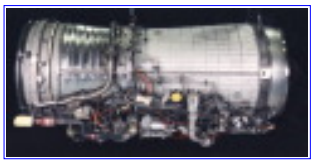
PERFORMANCE RATINGS (T-O, S/L):

F118-100	84.6 kN (19,000 lb st)
F118-101	75.7 kN (17,000 lb st)

SPECIFIC FUEL CONSUMPTION (T-O, as above):

F118-100	18.98 mg/Ns (0.670 lb/h/lb st)

F118-101	18.64 mg/Ns (0.658 lb/h/lb st)



F118-GE-100 (1996)

Showing inlet of F118 (1996)

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GENERAL ELECTRIC - GE AIRCRAFT ENGINES

CT58

see T58

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CF34

see TF34

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Jane's Aero-Engines 04

ALLISON ENGINE COMPANY

MANUFACTURER DETAILS

This company is now to be found in this section under Rolls-Royce. At present the Allison brand name is being continued for military engines only.

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2 Images

WRIGHT AERONAUTICAL

WRIGHT J65

The J65 was an American derivative of the Armstrong Siddeley Sapphire, made under licence by Wright Aeronautical. Small numbers are still in use of the following types:

J65-W-3

Bendix-Utica fuel/air gas-turbine starter. Fitted to F-84F.

J65-B-3

Similar to W-3 but made by Buick.

J65-W-5

Solid-propellant cartridge starter. Fitted to B-57.

J65-W-7

Fuel/air starter. Fitted to RF-84F.

J65-B-7

Similar to W-7 but made by Buick.

J65-W-16A

Pneumatic turbine starter. Fitted to F-1E and AF-1E.

J65-W-16C

Similar to W-16A. Fitted to early models of A-4.
TYPE: Single-shart turbojet.
INTAKE: Magnesium-alloy casting with front bearing supported by four struts anti-iced by hot bleed air. Fixed-incidence steel inlet guide vanes.
COMPRESSOR: Thirteen-stage axial, first seven stages forming LP section and last six HP section. Rotor blades secured in steel discs shrunk on hollow aluminium-alloy shaft. First three rows of blades are of steel and secured in discs by fir-tree roots. Remaining rows of blades, four of aluminium-alloy and last six of stainless steel, secured between pairs of discs by rivets. One row of inlet guide vanes and all stator blades secured by serrations in carrier rings, which are split for ease of assembly in compressor case. Inlet guide vanes and first seven rows of stator blades are shrouded. Compressor shaft carried on ball bearing (front) and roller bearing (rear). Compressor casing, an aluminium casting, is split on horizontal centreline. Mass flow at max 8,300 rpm typically 55.3 kg (122 lb)/s. Pressure ratio 7.1.
COMBUSTION CHAMBER: Single annular type with inner and outer stainless steel liners between which combustion takes place. Thirty-six `walking stick' fuel-air vaporisers. Fuel and primary air fed into hook-shaped vaporiser tubes whose open ends point upstream. Secondary air enters chamber through 36 air distributing cups and mix with combustion flame near vaporiser tubes.
TURBINE: Two-stage turbine. Steel discs coupled together by taper pins. First-stage disc has 110 blades, second-stage 75 blades, all secured by fir-tree roots and locking plates, and made of nickel-base alloy. Front turbine disc attached by taper pins to turbine rotor stub shaft. This is splined to turbine rotor shaft coupling, front end of turbine rotor shaft, in turn, being bolted to compressor rotor shaft drive coupling. Roller bearing in front of rotor disc assembly. TET 802ºC.
JETPIPE: Fixed type. Steel jetpipe and inner cone.
MOUNTING: Three-point suspension, one stabilising support at front and two trunnions at centre main bearing support on horizontal centreline.
ACCESSORIES: Off extension of compressor shaft in front main bearing support.
STARTING: Electric starter/generator in bullet fairing on nose (photo), or fuel/air, cartridge or pneumatic starter. Primers and igniters in combustion chamber. Electrical and ignition systems interconnected for automatic starting.
CONTROL SYSTEM: Low-presure fuel control.
FUEL SPECIFICATION: MIL-L-5624A (JP-4).

OIL SYSTEM: Closed system. Oil tank of 18.9 litres (5 US gallons; 4.2 Imp gallons) located on upper port side of compressor casing. One gear-type pressure pump, one gear-type scavenge pump and two piston-type metering pumps. Normal oil pressure 1.4 to 2.4 kg/cm^2 (20 to 35 lb/sq in).
OIL SPECIFICATION: MIL-L-7808.
DIMENSIONS:

Length:	
W-3, W-5, W-7	2,921 mm (115 in)
W-16	3,098 mm (122 in)
Diameter	953 mm (37.5 in)

WEIGHT, DRY:

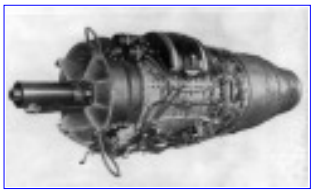
W-3, W-5	1,178 kg (2,595 lb)
W-7	1,268 kg (2,795 lb)
W-16	1,244 kg (2,742 lb)

PERFORMANCE RATINGS:
T-O, S/L:

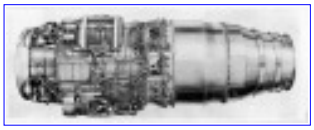
W-3	32.15 kN (7,220 lb st)
W-5	32.06 kN (7,200 lb st)
W-7, W-16A	34.74 kN (7,800 lb st)
W-16C	37.85 kN (8,500 lb st)

SPECIFIC FUEL CONSUMPTION:
T-O, as above:

all, typical	25.8 mg/Ns (0.91 lb/h/lb st)



J65 with electric starter/generator (1996)



J65-W-7 (1996)

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AERO-ENGINES - MANUFACTURER, UNITED STATES OF AMERICA

Jane's Aero-Engines 03

WRIGHT AERONAUTICAL - WRIGHT

MANUFACTURER DETAILS

At the end of the Second World War this was the world's largest aviation company. Today it no longer exists, but a small number of J65 engines are still flying, together with a much larger number of various Wright piston engines. They are supported by Curtiss-Wright Flight Systems, of Shelby, North Carolina 28150-8008.

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AERO-ENGINES - TURBOJET, UNITED STATES OF AMERICA

Jane's Aero-Engines 03

UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JT11D-20

US military designation: J58-P-1

This propulsion system stemmed from a Navy requirement of 1955 for a M3 attack aircraft. A photograph shows the JT11A-21. The redesigned J58 found a home in Lockheed SR-71 strategic reconnaissance aircraft. A diagram shows some of the features which vary the geometry of the installed engine. The following table shows how at high supersonic Mach numbers the contribution to propulsive thrust of each part of the installation changes completely:

Percentage contribution to propulsive thrust

Mach number	Inlet	Engine	Nozzle
0 (ie, static)	0	100	0
2.2	13.0	73.0	14.0
3.2	54.0	17.6	28.4

The SR-71 was withdrawn from USAF service, but two SR-71As and one dual-control SR-71B continued airworthy at NASA Dryden at Edwards AFB. In 1995 the US Air Force considered the capability of the SR-71 so valuable that these aircraft were returned to Air Force charge, and today remain operational.

TYPE: Single-shaft turbojet with bypass pipes and afterburner, designed for cruise at M3.2.

INTAKE: Sharp-edged circular inlet (part of airframe) with central conical centrebody (called the spike) with hydraulic translation to vary duct area and focus shock-waves. Forward bypass (`onion slicer') and aft bypass (`cabbage slicer') to control nacelle flows. Centrebody bleed linked to spike perforations.
COMPRESSOR: Nine axial stages. Variable inlet guide vanes and large bleed from fourth stage through six bypass pipes to prevent upstream stalling and downstream choking. Mass flow 166 kg (365 lb)/s. Pressure ratio 8.8.
COMBUSTION CHAMBER: Can-annular with interconnected Inconel flame tubes each with multiple downstream burners.
TURBINE: Two-stage, with air-cooled nozzle and rotor blades.
AFTERBURNER: Multiple spray rings and flameholders with hot-streak ignition.
NOZZLE: Multiflap primary nozzle convergent (minimum area) at low speeds and fully open to convergent/divergent profile at supersonic speeds. Surrounded by large ejector (airframe), again with variable profile and area.
CONTROL SYSTEM: Hydromechanical, with exceptional requirements (see diagram).
FUEL SPECIFICATION: JP-7 only.
DIMENSIONS:

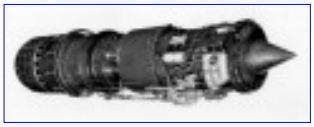
Diameter	1,407 mm (55.4 in)
Length	5,377 mm (211.7 in)
WEIGHT, DRY:	2,869.5 kg (6,326 lb)

PERFORMANCE RATINGS:

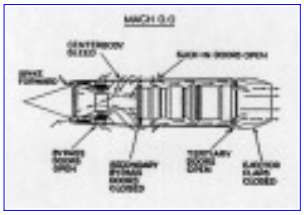
Max T-O	153.5 kN (34,500 lb st)

SPECIFIC FUEL CONSUMPTION:

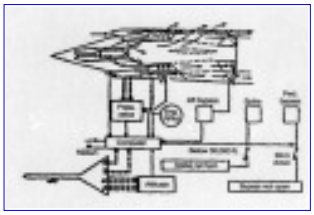
T-O as above	61.57 mg/Ns (2.174 lb/h/lb st)



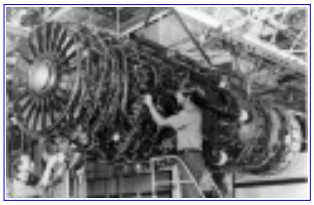
JT11A-21 following M3 demo, November 1958 (1996)



Simplified diagrams showing variable geometry of the J58 nacelle (1996)



Simplified diagram of the J58 control system. At lower left is the air-data sensor (1996)



J58-P-1 (1996)

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UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JT8B

US military designation: J52

Announced in December 1956, this two-shaft turbojet was designed under US Navy Bureau of Weapons contract to power the Grumman A2F, which became the A-6 Intruder. Other versions were adopted for later models of the Douglas A-4 Skyhawk, and the USAF selected a different version for the GAM-77 (AGM-28) Hound Dog supersonic cruise missile. This family of engines has had a very long active life, and also formed the basis for the JT8D series of commercial turbofans. The following are the principal models:

J52-P-6 (JT8B-1)

T-O rating 37.8 kN (8,500 lb st). Powers A-4E and derivatives.

J52-P-8 (JT8B-3)

T-O rating 41.4 kN (9,300 lb st). Powers all versions of A-6 and EA-6A, and A-4E, A-4F, TA-4F and TA-4J.

J52-P-408

T-O rating 49.8 kN (11,200 lb st). Powers EA-6B, A-4F re-engined, A-4M, OA-4M, A-4N and several A-4 export models.

J52-P-409

T-O rating increased to 53.4 kN (12,000 lb st), with reduced maintenance and longer life. Powers EA-6B ADVCAP and available for upgraded A-6E.
TYPE: Two-shaft turbojet.
INTAKE: Direct pitot type with front LP bearing held in 14 (in early versions 18) radial struts.
LP COMPRESSOR: Five stages, P-408 and -409 have two-position variable inlet guide vanes.
HP COMPRESSOR: Seven stages. Pressure ratio, P-6, 12.0; P-8, 12.6; P-408, 14.5; P-409, 14.6.
COMBUSTION CHAMBER: Can-annular type, with nine cans each with four dual-orifice fuel burners. In -408, -409, modified for reduced smoke.
HP TURBINE: Single stage, in P-408, -409 with air-cooled vanes and blades.
LP TURBINE: Single stage.
JETPIPE: Plain fixed-area type.
DIMENSIONS:

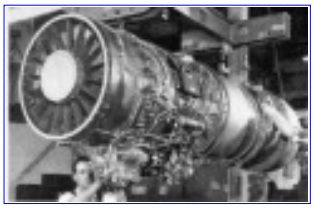
Length:	
J52-P-6, P-8	3,018 mm (118.5 in)
J52-P-408, P-409	3,020 mm (118.9 in)
Diameter:	
J52-P-6, P-8	766 mm (30.15 in)
J52-P-408, P-409	814.3 mm (32.06 in)

WEIGHT, DRY:

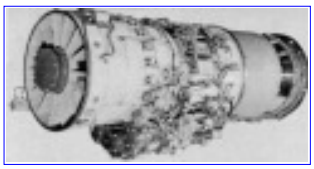
J52-P-6	933 kg (2,056 lb)
J52-P-8	961 kg (2,118 lb)
J52-P-408	1,052 kg (2,318 lb)

PERFORMANCE RATINGS: See model listing
SPECIFIC FUEL CONSUMPTION (T-O, S/L):

J52-P-6	23.2 mg/Ns (0.82 lb/h/lb st)
J52-P-8	24.35 mg/Ns (0.86 lb/h/lb st)
J52-P-408	22.9 mg/Ns (0.81 lb/h/lb st)



J52-P-6 (1996)



J52-P-408

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AERO-ENGINES - TURBOJET, UNITED STATES OF AMERICA

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UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JT4A

US military designation: J75

This larger turbojet was designed to power the USAF F-105, F-106 and F-107, and Navy F8U-3 (in each case fitted with an afterburner) and Navy P6M. The only aircraft to go into production were the F-105 and F-106. The commercial JT4A was produced, without an afterburner, for the 707-220 and -320 and the DC-8-20 and -30. The first commercial version, the JT4A-3, was FAA certificated in March 1957.

Small numbers of both military and commercial engines are still in use:

J75-P-9

Afterburner. Powers Lockheed Martin (Convair) F-106B, retained as chase aircraft.

J75-P-13B

No afterburner. Powers Lockheed Martin U-2 and TR-1 versions, but being replaced by F118-GE-101.

J75-P-17

Afterburner. Powers F-106A, retained as chase and research aircraft.

J75-P-19W

Afterburner and water injection. F-105D and later.

JT4A-9, -11

No afterburner. Powers remaining DC-8-30 series.
TYPE: Two-shaft turbojet, with or without afterburner.
COMPRESSOR: Eight-stage LP spool, seven-stage HP spool. Mass flow (typical) 136 kg (300 lb)/s. Pressure ratio 12.2. Max HP speed 8,000 rpm.
COMBUSTION CHAMBER: Can-annular, enlarged version of J57, with eight interconnected Inconel flame tubes each with six downstream air-blast burners.
HP TURBINE: Single-stage, with solid Waspaloy blades.
LP TURBINE: Two-stage, bolted to stub shaft and ball bearing each side of discs.
JETPIPE: JT4A versions have plain fixed-area jetpipe, as does J75-P-13B. Afterburning models have multi-petal nozzle driven by 12 rams.
DIMENSIONS:

Envelope diameter	1,092 mm (43.0 in)
Length:	
JT4A-9, -11	3,660 mm (144.1 in)
J75-P-13B	4,858 mm (191.1 in)
J75-P-19W	6,502 mm (256.0 in)

WEIGHT, DRY:

JT4A-9	2,290 kg (5,050 lb)
JT4A-11	2,315 kg (5,104 lb)
J75-P-13B	2,381 kg (5,250 lb)
J75-P-19W	2,703 kg (5,960 lb)

PERFORMANCE RATINGS (T-O, S/L, ISA):

JT4A-9	74.5 kN (16,800 lb st)
JT4A-11	78.1 kN (17,500 lb st)
J75-P-13B	75.9 kN (17,000 lb st)
J75-P-17	109.4 N (24,500 lb st)
J75-P-19W	118.3 kN (26,500 lb st)

SPECIFIC FUEL CONSUMPTION (T-O, as above):

JT4A-9, -11	22.66 mg/Ns (0.80 lb/h/lb st)
J75-P-17	62.30 mg/Ns (2.2 lb/h/lb st)

J75-P-9 (1996)

JT4A-9 (1996)

J75-P-19W (1996)

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UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JT3

Civil designation: JT3C
US military designation: J57

This two-spool turbojet was designed in 1947-49, first ran in January 1950 and flew under a B-50 testbed in March 1951. It set new high standards in pressure ratio and fuel economy, and was selected for the B-52, F-100 and many other USAF fighters, bombers, tankers and cruise missiles. Unexpectedly it also replaced the J40 in Navy fighters and bombers, and made possible the 707 and DC-8 jetliners. By 1960 Pratt & Whitney had delivered 15,024, and Ford Motor Co a further 6,202.

Very few of the afterburning fighter engines remain in service, mainly with private owners. A small number of JT3C-6 civil engines are still in commercial service, and the only substantial fleet remaining is made up of Boeing C/KC/RC/EC-135 aircraft of various sub-types still powered by the J57-P-59W. The J57 led to the larger JT4A (J75) and the refanned JT3D (TF33).

TYPE: Two-shaft turbojet.

INTAKE: Annular, with radial struts supporting centre housing for front main bearing. Fixed steel inlet guide vanes. Anti-icing by compressor bleed.

LP COMPRESSOR: Nine-stage axial-flow, on inner of two concentric shafts. Shaft runs on roller bearings fore and aft (thrust) and is splined at rear end to LP turbine shaft. Rotor made up of nine steel discs, and blades are all steel. Steel casing, split on horizontal centreline, carries nine rows of steel stator blades.

HP COMPRESSOR: Seven-stage axial-flow, on outer hollow shaft which is on two roller bearings and is splined to HP turbine shaft. Same type casing and rotor as for LP with steel stator and rotor blades. Pressure ratio 12.5. Mass flow 91 kg (200 lb)/s.
COMBUSTION CHAMBER: Outer annular steel casing encloses eight Inconel alloy interconnected flame tubes, each with six fuel nozzles. Combustion and cooling air introduced through central tube as well as through outer walls. Two igniter plugs.
TURBINE: Single-stage high-pressure and two-stage low-pressure turbines in tandem on concentric drive shafts, the outer shaft on a ball-bearing forward of the HP turbine wheel and the inner on a ball bearing aft of the LP turbine wheels. Steel casing with steel guide and stator vanes, steel turbine wheels and blades.
JETPIPE: Fixed type. Provision for thrust reverser and noise suppressor on tail pipe.
STARTING: Starter, either electric or air turbine, mounted on front of intake centre boss or on external gearbox.
CONTROL SYSTEM: Two-stage fuel pump (in P-59W, Chandler Evans 9400) with overspeed governor supplies fuel nozzles via Hamilton Standard flow-control unit. Provision for water injection.
FUEL SPECIFICATION: JP-1 or JP-4 (MIL-F-5616 or 5624).
OIL SYSTEM: Closed system. Main gear-type pressure pump feeds bearings through calibrated orifices. Scavenger pumps return oil through fuel-cooled heat exchanger to engine tank.
OIL SPECIFICATION: Synthetic Turbo Oil ETQ 15 (MIL-L-7808).
DIMENSIONS:

Length overall:	
J57-P-59W	4,289 mm (169. 0 in)
JT3C-6	4,250 mm (167.53 in)
Diameter	987 mm (38.88 in)

WEIGHT, DRY:

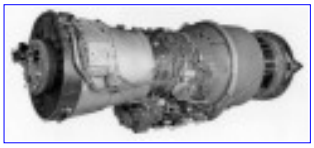
J57-P-59W	1,959 kg (4,320 lb)
JT3C-6	1,922 kg (4,234 lb)

PERFORMANCE RATINGS (T-O, S/L, ISA):

J57-P-59W (wet)	61.1 kN (13,750 lb st)
(dry)	51.1 kN (11,500 lb st)
JT3C-6 (wet)	57.8 kN (13,000 lb st)
(dry)	48.9 kN (11,000 lb st)
	all above ratings at 8,200 rpm

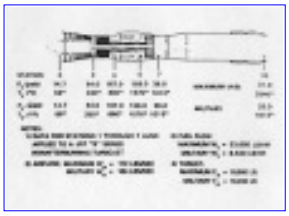
SPECIFIC FUEL CONSUMPTION (T-O):

J57-P-59W (wet)	28.16 mg/Ns (0.95 lb/h/lb st)
JT3C-6 (wet)	27.27 mg/Ns (0.92 lb/h/lb st)



JT3C-6 (1996)

J57-P-25 afterburning fighter engine (1996)



S/L static pressures and temperatures in a typical afterburning J57 (1996)

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UNITED TECHNOLOGIES PRATT & WHITNEY

PRATT & WHITNEY JFTD12

US military designation: T73
This turboshaft engine was developed to power the Sikorsky S-64 Skycrane (US Army CH-54 Tarhe) heavy-lift crane helicopter. The designation reflected the fact that the gas generator was essentially that of the JT12 turbojet. Three versions were produced:

JFTD12A-1

T-O rating 3,020 kW (4,050 shp). Powered S-64A. On test in September 1960, and flying in S-64 from 9 May 1962.

JFTD12A-4A (T73-P-1)

T-O rating 3,356 kW (4,500 shp). Powered S-64E and CH-54A.

JFTD12A-5A (T73-P-700)

T-O rating 3,579 kW (4,800 shp). Powers S-64F and CH-54B.

Deliveries totalled 351 engines. A small number remain in use.

TYPE: Free-turbine turboshaft.

COMPRESSOR: As JT12A. Pressure ratio 6.85.

COMBUSTION CHAMBER: As JT12A.

COMPRESSOR TURBINE: As JT12A.

POWER TURBINE: Two-stage free turbine running on its own two-bearing shaft.

JETPIPE: Short, fixed-area, handed (curved out to outboard side of aircraft).

OUTPUT: Direct drive from power turbine in cooled tunnel in jetpipe to reduction gear on helicopter.

ACCESSORIES: Grouped under the compressor casing, driven by a vertical shaft from the compressor.

STARTING: Usually electric.

CONTROL SYSTEM: Hydromechanical.

DIMENSIONS:

Length overall	2,730 mm (107.5 in)
Diameter	744 mm (29.3 in)
Height	864 mm (34.0 in)

WEIGHT, DRY:

JFTD12A-1	400 kg (882 lb)
JFTD12A-4A	417 kg (920 lb)
JFTD12A-5A	424 kg (935 lb)

PERFORMANCE RATINGS:

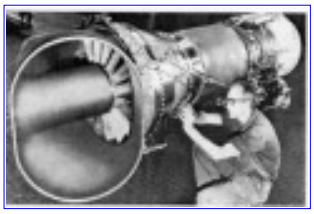
T-O: See model listing

SPECIFIC FUEL CONSUMPTION:

T-O rating:

JFTD12A-1	117.4 µg/J (0.695 lb/h/shp)
JFTD12A-4A	116.6 µg/J (0.690 lb/h/shp)
JFTD12A-5A	110.7 µg/J (0.655 lb/h/shp)

JFTD12A-5A (1996)



JFTD12A showing jetpipe and drive shaft tunnel (1996)

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AERO-ENGINES - TURBOJET, UNITED STATES OF AMERICA

Jane's Aero-Engines 03

UNITED TECHNOLOGIES PRATT & WHITNEY J75

see JT4A

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AERO-ENGINES - TURBOJET, UNITED STATES OF AMERICA

Jane's Aero-Engines 03

UNITED TECHNOLOGIES PRATT & WHITNEY J60

see JT12A

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AERO-ENGINES - TURBOJET, UNITED STATES OF AMERICA

Jane's Aero-Engines 03

UNITED TECHNOLOGIES PRATT & WHITNEY J58

see JT11D-20

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AERO-ENGINES - TURBOJET, UNITED STATES OF AMERICA

Jane's Aero-Engines 03

UNITED TECHNOLOGIES PRATT & WHITNEY J57

see JT3

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AERO-ENGINES - TURBOJET, UNITED STATES OF AMERICA

Jane's Aero-Engines 03

UNITED TECHNOLOGIES PRATT & WHITNEY J52

see JT8B

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GE AIRCRAFT ENGINES - GENERAL ELECTRIC

GENERAL ELECTRIC T64

The T64 was developed initially for the US Navy. Design began in 1954 but production did not start until 1964. It is available as a turboshaft or turboprop. Current versions are:

T64-GE-7A

Direct-drive turboshaft; produced under licence by MTU of Germany for CH-53G.

T64-GE-10

Turboprop with remote gearbox above inlet duct; powers DHC-5 Buffalo and produced under licence by IHI of Japan for P-2J and PS-1.

T64-GE-100

-7A with improved turbine. Powers MH-53J.

T64/T4C2

T64-GE-100 with changed HP turbine blades. Powers S-65C-3.

T64-GE-413

Powers CH-53D.

T64-GE-415

Improved combustion liner and turbine cooling. Powers RH-53D.

T64-GE-416

As -415 with changed fuel control. Powers CH/MH-53E.

T64-GE-416A

As -416, improved turbine.

T64-GE-419

As -416A, with integral fuel/oil heat exchanger, and OEI emergency power on a 32.2ºC day. Powers MH-53E since 1995.

CT64-820-4

Civil turboprop, powers DHC-5D.

T64/P4D

Turboprop, powers G222 and C-27A. Co-production by GE and FiatAvio, supported by Alfa Romeo Avio from 1975.
TYPE: Free turbine turboshaft/turboprop.
COMPRESSOR: Fourteen-stage axial flow, single-spool steel rotor for -820/-1/2/3, titanium and Inconel compressor for -100, T4C2, -413, -415, -416, -416A, -419, -P4D and CT64-820-4. Inlet guide vanes and first four stages of stator vanes variable, air mass flow per second: -100, T4C2, -413, -415, -416, -416A, -419, 13.3 kg (29.4 lb); -820-4, 11.9 kg (26.2 lb); P4D, 12.2 kg (27.0 lb). Pressure ratio: -820-4, 12.5; -100, T4C2, -413, -415/-416/-416A, -419, 14.0; P4D, 13.0.
COMBUSTION CHAMBER: Annular type. Double fuel manifold feeds 12 duplex type fuel nozzles.

COMPRESSOR TURBINE: Two-stage, coupled directly to compressor rotor by splined connection. Engines rated 3,266 kW (4,380 shp) or over have air-cooled first-stage blades.
POWER TURBINE: Two-stage, independent of gas generator.
OUTPUT: Remotely mounted gearbox for turboprop, offset and accessible for inspection and replacement. Ratio 13.44.
STARTING: Mechanical, airframe supplied.
DIMENSIONS:

Length: T64-GE-7A, -100, T4C2, -413, -415, -416, -416A, -419	2,006 mm (79 in)
T64-10, P4D, CT64-820-4	2,793 mm (110 in)
Width: T64-GE-100, T4C2, -413, -415, -416, -416A, -419	660 mm (26.0 in)
T64/P4D, CT64-820-4	683 mm (26.9 in)
Height: T64-GE-7A, -100, T4C2, -413, -415, -416, -416A, -419	825 mm (32.5 in)
T64-10, P4D, CT64-820-4	1,167 mm (46 in)

WEIGHT, DRY:

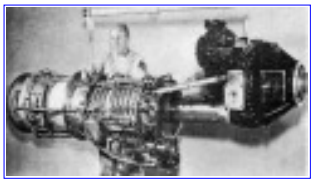
T64-GE-7A, -100, T4C2, -413, -415, -416, -416A	327 kg (720 lb)
T64-GE-419	343 kg (755 lb)
T64-GE-10, CT64-820-4	520 kg (1,145 lb)
T64/P4D	538 kg (1,188 lb)

PERFORMANCE RATINGS (max rating, S/L):

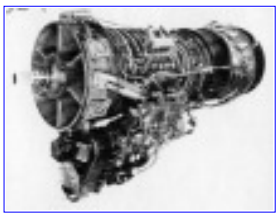
T64-GE-7A	2,935 kW (3,936 shp) to 28º
T64-GE-10	2,215 kW (2,970 shp) at 1,160 output rpm
T64-GE-100	3,229 kW (4,330 shp) to 29.4ºC
T64-GE-413	2,927 kW (3,925 shp)
T64-GE-415, -416, -416A	3,266 kW (4,380 shp)
T64-GE-419	3,542 kW (4,750 shp)
CT64-820-4	2,336 kW (3,133 shp)
T64/P4D	2,535 kW (3,400 shp) to 45ºC

SPECIFIC FUEL CONSUMPTION (cruise rating, S/L):

T64-GE-7A, -100, -413, -415, -416, -416A, -419	79.4 µg/J (0.47 lb/h/shp)
T64-GE-10, CT64-820-4, T64/P4D	84 µg/J (0.50 lb/h/shp)

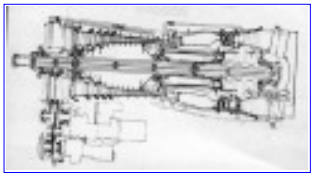


T64-10 (1996)



T64-GE-415/416 (1996)

T64-P4D (1997)



Longitudinal section through T64 core (1997)

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GE AIRCRAFT ENGINES - GENERAL ELECTRIC

GENERAL ELECTRIC CT58

The commercial version of the T58 is designated CT58 and was the first US helicopter turbine to receive FAA certification.

Current versions are as follows:

CT58-110

Rated at 932 kW; 1,250 shp (1,007 kW; 1,350 shp for 2.5 min) at 19,500 rpm. Mass flow 5.67 kg (12.7 lb)/s. Pressure ratio 8.2.

CT58-140

Rated at 1,044 kW; 1,400 shp (1,119 kW; 1,500 shp for 2.5 min) at 19,500 rpm. Mass flow 6.21 kg (13.7 lb)/s. Pressure ratio 8.4.

The CT58 powers the Sikorsky S-61 and S-62 and Boeing Vertol 107 Model II.

DIMENSIONS:

Max width	406 mm (16.0 in)

Length overall	1,500 mm (59.0 in)

WEIGHT, DRY:

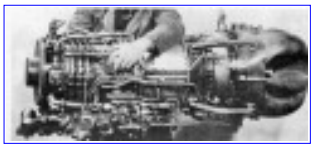
CT58-110	143 kg (315 lb)
CT58-140	154 kg (340 lb)

PERFORMANCE RATINGS (S/L, ISA):
2.5 min, see above

Cruise:	
CT58-110	783 kW (1,050 shp)
CT58-140	932 kW (1,250 shp)

SPECIFIC FUEL CONSUMPTION (cruise):

CT58-110	108 µg/J (0.64 lb/h/shp)
CT58-140	105 µg/J (0.62 lb/h/shp)



CT58-110 (1996)

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AERO-ENGINES - TURBOSHAFT, UNITED STATES OF AMERICA

Jane's Aero-Engines 03

GE AIRCRAFT ENGINES - GENERAL ELECTRIC

GENERAL ELECTRIC T58

The T58 is a small free-turbine turboshaft engine which was developed for helicopter propulsion for the US Navy Bureau of Weapons. A civil version, the CT58, was awarded a Type Certificate by the FAA on 1 July 1959 and is described immediately following the T58.

A hydromechanical constant-speed control system featured in the T58 maintains essentially constant rotor speed by regulating the engine power automatically, so eliminating the need for speed adjustment by the pilot during normal operation.

Rolls-Royce Ltd produced modified versions of the T58 under licence in the United Kingdom as the Gnome (which see). The T58 was also licensed for manufacture in Italy and Japan. Industrial and marine version of the T58 is the LM100. By the end of GE production in 1984 over 6,300 T58 and CT58 engines had been delivered (not including LM100 versions).

Versions currently in service are as follows:

T58-GE-3

Five-minute rating of 988 kW (1,325 shp). Powers Bell UH-1F.

T58-GE-5

Five-minute rating of 1,119 kW (1,500 shp). Powers Sikorsky CH-3E, HH-3E/F and NASA RSRA (Sikorsky S-72).

T58-GE-8E, F

Rated at 1,007 kW (1,350 shp). Powers Boeing Vertol CH-46A, Kaman SH-2, Sikorsky SH-3A/G and HH-52A.

T58-GE-10

Rated at 1,044 kW (1,400 shp). Powers Sikorsky SH-3D/H, and Boeing Vertol CH-46D/F.

T58-GE-16

Rated at 1,394 kW (1,870 shp). US military qualified. Air-cooled gas generator turbine and two-stage power turbine. Powers Boeing Vertol CH-46E.

T58-GE-100

Uprated T58-GE-5. Ten-minute rating 1,119 kW (1,500 shp) to 15ºC or 1,100 kW (1,475 shp) to 26ºC. Powers selected CH/HH-3E. Qualified 1976.
TYPE: Free-turbine turboshaft.
INTAKE: Annular intake casing with four hollow radial struts supporting central housing for starter drive clutch and front main roller bearing. Casing and struts anti-iced by air bled from compressor.
COMPRESSOR: Ten-stage axial-flow. Variable-incidence inlet guide vanes. First three of the 11 rows of stator blades also have variable incidence.One-piece steel construction for last eight stages of rotor hub. Casing divided into upper and lower halves. Pressure ratio 8.4. Mass flow 5.62 kg (12.4 lb)/s in T58-GE-3 and 8E, 6.21 kg (13.7 lb)/s in T58-GE-5, -10 and -100, 6.30 kg (13.9 lb)/s in T58-GE-16.
COMBUSTION CHAMBER: Annular type. Sixteen fuel nozzles (eight on each of two manifolds) mounted on front of inner liner. Dual capacitor discharge ignition unit. Outer casing in two halves to facilitate inspection.
COMPRESSOR TURBINE: Two-stage short-chord axial-flow type, coupled directly to compressor by hollow conical shaft. Centre ball thrust bearing, rear roller bearing. Cooling by air bled from compressor. T58-GE-16 has air-cooled first-stage turbine nozzle and blades and second-stage nozzle.
POWER TURBINE: Single-stage (two-stage in T58-GE-16) axial-flow type, mechanically independent of gas generator (compressor) turbine. Operated nominally at 19,500 rpm.
JETPIPE: Two positions (90º left or right) on all versions. T58-GE-16 can also be supplied with downward-ejecting or multiple-position exhaust.
OUTPUT: Engines with single-stage power turbine can have reduction gear giving output at 6,000 rpm. Power turbine accessory drive unit and flexible feedback cable provide a speed signal to the control. Optional gearbox with integral lubrication system also reduces power speed to 6,000 rpm. Assembly

includes an integral torque sensing system.
ACCESSORIES: Engine accessories driven from compressor shaft. Airframe accessories mounted on free-turbine reduction gearbox on rotor hub.
CONTROL SYSTEM: (except T58-GE-10 and -16): Free turbine constant-speed control. Hydromechanical controls. (T58-GE-10, -16): Integrated hydromechanical/electrical power control system for isochronous speed governing and twin-engine load sharing.
DIMENSIONS:

Max width:	
except T58-GE-16	526 mm (20.7 in)
T58-GE-16	607 mm (23.9 in)
Length overall:	
except T58-GE-16	1,499 mm (59.0 in)
T58-GE-16	1,626 mm (64.0 in)

WEIGHT, DRY:

T58-GE-3	140 kg (309 lb)
T58-GE-5, -100	152 kg (335 lb)
T58-GE-8E, F	138 kg (305 lb)
T58-GE-10	159 kg (350 lb)
T58-GE-16	201 kg (443 lb)

PERFORMANCE RATINGS:
5 min: See under model listings

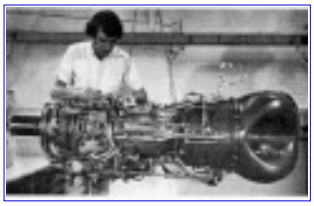
MIL:	
T58-GE-3	988 kW (1,325 shp) at 20,960 rpm
T58-GE-5, 10	1,044 kW (1,400 shp) at 19,500 rpm
T58-GE-8E, F	1,007 kW (1,350 shp) at 19,500 rpm
T58-GE-16	1,394 kW (1,870 shp) at 19,500 rpm
T58-GE-100	1,119 kW (1,500 shp) at 19,500 rpm
Cruise:	
T58-GE-3	798 kW (1,070 shp)
T58-GE-5, 10	932 kW (1,250 shp)
T58-GE-8E, F	857.5 kW (1,150 shp)
T58-GE-16	1,320 kW (1,770 shp)
T58-GE-100	1,015 kW (1,360 shp) at 19,500 rpm

SPECIFIC FUEL CONSUMPTION:

At MIL rating:	
T58-GE-3	103 µg/J (0.61 lb/h/shp)
T58-GE-5, 8E, F, 10, 100	101 µg/J (0.60 lb/h/shp)
T58-GE-16	89.5 µg/J (0.53 lb/h/shp)

At cruise rating:	
T58-GE-3	106.5 µg/J (0.63 lb/h/shp)
T58-GE-5, 100	103 µg/J (0.61 lb/h/shp)
T58-GE-8E, F	105 µg/J (0.62 lb/h/shp)
T58-GE-10	105 µg/J (0.62 lb/h/shp)
T58-GE-16	91 µg/J (0.54 lb/h/shp)

Left, T58-8B; right, T58-10 (1996)



T58-16 (1996)

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AERO-ENGINES - TURBOJET, UNITED STATES OF AMERICA

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GE AIRCRAFT ENGINES - GENERAL ELECTRIC

GENERAL ELECTRIC J85

In 1954 this simple single-shaft turbojet was launched as GE's second Small Engine division product, its initial application being a cruise missile. Production continued to late 1988, by which time deliveries exceeded 13,600. The following are current versions:

J85-4A

Powers the Rockwell International T-2C Buckeye trainer and Lockheed XST-1 (Have Blue 1001).

J85-5

Afterburning version with 6.6:1 thrust-to-weight ratio; powers Northrop T-38 Talon supersonic trainer.

J85-13

Developed from J85-5, with increased turbine inlet temperature for Northrop F-5A/B supersonic fighter. As the J85-13A, licence-built by Alfa Romeo, also powers Aeritalia G91Y.

J85-15

Version of J85-13 with improved turbine and hydraulically actuated exhaust nozzle to power CF-5 and NF-5. Manufactured under licence in Canada by Orenda.

J85-17A/B

Powers Saab 105G attack/reconnaissance aircraft and Cessna A-37B attack aircraft. Also used as take-off and climb booster for Fairchild C-123K and AC-119K.

J85-21

Higher air flow version with zero stage to give total of nine compressor stages. Equipped with afterburner for supersonic aircraft. Powers Northrop F-5E/F Tiger II.

The civil version of the J85 is the CJ610 turbojet, to which the aft-fan CF700 turbofan is closely related. Both are described separately.

The following data refer specifically to the J85-21:

TYPE: Single-shaft turbojet.

INTAKE: Annular type, surrounding central bullet fairing. Variable-incidence inlet guide vanes, with hot-air anti-icing.

COMPRESSOR: Nine-stage axial-flow type, with variable inlet guide vanes and first three stator stages. Titanium rotor blades, first two stages having part-span shrouds. Discs joined at periphery. Casing in upper and lower halves. Early versions with eight-stage compressor: mass flow 20 kg (44 lb)/s, pressure ratio 7. J85-21: mass flow 24.0 kg (53.0 lb)/s, pressure ratio 8.3.

COMBUSTION CHAMBER: Annular type with perforated liner. Twelve duplex fuel injectors. Ports in outer casing facilitate inspection of liner.

TURBINE: Two-stage axial-flow type. Casing is in halves, split horizontally. Turbine inlet temperature 977ºC.

AFTERBURNER: Consists of a diffuser and a combustor. A pilot burner with four spraybars and a main burner of 12 spraybars are located in the diffuser section. Combustion is initiated by a single igniter plug and is then self-sustained. Nozzle position governs exit area and is regulated automatically by the afterburner control system as a function of turbine exit temperature and throttle lever position.

STARTING: Air impingement starter on afterburning engines. Provision for starter/generator on non-afterburning engines.

OIL SYSTEM: Positive displacement, pressurised recirculating type.

DIMENSIONS:

Length:	
J85-4	1,029 mm (40.5 in)
J85-5	2,657 mm (104.6 in)
J85-13, -15	2,682 mm (105.6 in)
J85-17	1,039 mm (40.9 in)
J85-21	2,585 mm (112.5 in)

Diameter:	
J85-4, -17	450 mm (17.7 in)
J85-5, -13, -15, -21	533 mm (21.0 in)

WEIGHT, DRY:

J85-4	183 kg (404 lb)
J85-5	265 kg (584 lb)
J85-13	271 kg (597 lb)
J85-15	279 kg (615 lb)
J85-17	181 kg (398 lb)
J85-21	310 kg (684 lb)

PERFORMANCE RATINGS (S/L, ISA):

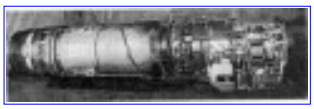
Max, with afterburner:	
J85-5	17.16 kN (3,850 lb st)
J85-13	18.19 kN (4,080 lb st)
J85-15	19.17 kN (4,300 lb st)
J85-21	22.29 kN (5,000 lb st)
Max dry (MIL):	
J85-4	13.15 kN (2,950 lb st)
J85-5	11.95 kN (2,680 lb st)
J85-13, -15	12.12 kN (2,720 lb st)
J85-17	12.70 kN (2,850 lb st)
J85-21	15.60 kN (3,500 lb st)

SPECIFIC FUEL CONSUMPTION (MIL rating):

J85-4	27.7 mg/Ns (0.98 lb/h/lb st)
J85-5, -13, -15	29.15 mg/Ns (1.03 lb/h/lb st)
J85-17	27.5 mg/Ns (0.97 lb/h/lb st)
J85-21	28.3 mg/Ns (1.00 lb/h/lb st)

J85-5 (1996)

J85-21 (1996)



J85-13 (1996)

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GE AIRCRAFT ENGINES - GENERAL ELECTRIC

GENERAL ELECTRIC J79

Development of the J79, America's first high-compression variable-stator turbojet, began in 1952. It was flight tested for the first time in 1955 and became the first production M2 engine when it was selected to power the General Dynamics B-58 Hustler bomber. In addition to production by General Electric, versions of the J79 have been manufactured by Orenda of Canada to power the Canadair CF-104/F-104G (MAP), by Ishikawajima-Harima in Japan for the licence-built F-104DJ, and by MTU of Germany, FiatAvio of Italy and TA (formerly FN) of Belgium for the European-built F-104G. The Italian production team, including Alfa-Romeo, produced the J79-GE-19.

The last version to remain in production was the J79-J1E, made under licence by Israel Aircraft Industries. Production was completed in 1993, bringing the total number of J79 engines produced to abut 16,990, of which 3,290 were made by licensees.

Derivatives of the J79 have been the CJ805-3 turbojet and CJ805-23 turbofan, powering the Convair 880 and 990 Coronado, respectively, as well as the LM1500 industrial and marine gas turbine.

Versions of the J79 are as follows:

J79-GE-7A

Powered the Lockheed F-104C and D Starfighters. Built under licence by Orenda (as J79-OEL-7) for

Canadair CF-104.

J79-GE-10

Advanced version powering McDonnell Douglas F-4J. Entered production in June 1966.

J79-GE-11A

For US-built Lockheed F-104G Starfighters. Built under licence in Japan (as J79-IHI-11A), Germany, Italy, Belgium and Canada.

J79-GE-15

Powered McDonnell Douglas F-4C, F-4D and RF-4C for USAF.

J79-GE-17

Similar to J79-GE-10, but for F-4E, F-4F and F-4EJ.

J79-GE-19

Advanced version designed to supersede J79-GE-11A in F-104. Used in F-104S and F-104A. Differs from J79-GE-10/17 only in external characteristics. Guided expansion jet nozzle derived from nozzles of J79-GE-5C and YJ93. Afterburner system provides continuous thrust modulation.

J79-MTU-J1K

Similar to J79-GE-11A but modified by MTU with steel compressor rotor and steel/Mg casing, Inconel jetpipe and other changes for long life and reliability.

J79-IAI-J1E

Derived from the J79-GE-19 to fit the airframe of the Kfir. Cleared to 102 per cent rpm at high Mach numbers. Smokeless combustor, repositioned accessories, titanium shield around afterburner. Bedek Combat Plus system to increase maximum thrust.

The following details cover the basic features of all J79 variants except where otherwise indicated:
TYPE: Variable-stator single-shaft axial-flow turbojet with afterburner.
INTAKE: Annular type, surrounding central bullet fairing. Struts and inlet guide vanes anti-iced with compressor discharge air. First-stage stator anti-icing on J79-GE-10 and -15.
COMPRESSOR: Seventeen-stage axial-flow. First six stator stages and the inlet guide vanes have variable incidence. Setting of variable incidence vanes adjusted by dual actuators moved by engine fuel

to achieve optimum air flow angles for each stage at all engine speeds. Rotor, which runs on two bearings, is made from Lapelloy, F5F5 and titanium. Type 403 stainless steel blades and vanes. Total of 1,260 stator vanes and 1,271 rotor blades. Variable stator vanes have a platform, trunnion and threaded stem arrangement for external attachment to the actuation system linkage. Fixed stator vanes are inserted into T-slots on rear casing. All rotor blades have dovetail roots. Front compressor stator casing is made from a magnesium-thorium casting or Chromolloy forging, depending on engine model. On those engines requiring an intermediate compressor casing this is made of either A286 or 321 SS. All models have a forged and machined rear compressor stator casing, constructed in two halves for ease of assembly and disassembly. Typical air flow 76.6 kg (169 lb)/s. Pressure ratio 13.5.
COMBUSTION CHAMBER: Cannular type consisting of 10 combustion cans. Outer casing of Chromolloy, flame tube of Hastelloy. Dual igniters in cans 4 and 5.
TURBINE: Three-stage. Stages 1 and 2 bolted to shaft, stage 3 integral with aft shaft. All three stages of J79-GE-10, -17 and -19 have intermediate aged V57. Other models have all stages of A286. J79-10, -17 and -19 first stage has 148 blades of Udimet 700 or René 80, second stage has 114 blades of Udimet 500, third stage has 84 blades of M252. All blades attached by fir-tree roots. Three-stage NGVs, first with 58 vanes of R41, second with 62 vanes of Hastelloy R235 and R41, third with 44 vanes of A286. Lightweight casing of fabricated A286 in two easily removable halves. TET (typical) 988ºC.
JETPIPE: Liner of N155 and L605 with ceramic coating. Jetpipe of A286.
AFTERBURNER: Short type, max 1,985ºC with fully variable nozzle of `petal' type. Actuation by hydraulic rams utilising engine lubricating oil. Three-ring, quadrant-burning on all models except J79-GE-10, -15, -17 and -19, which have core annulus burning with radial spraybars.
MOUNTING: Pads provided on front frame and turbine frame for a variety of mounting arrangements, depending on airframe requirements.
ACCESSORIES: All engine controls and accessories, aircraft hydraulic pumps, generators, alternators and constant-speed drives (as required) are driven by two gearboxes on bottom of engine and a nose inlet gearbox.
STARTING: J79-GE-7A, -11A and -19 have pneumatic turbine starter mounted on front frame of inlet gearbox. J79-GE-10 has turbine air impingement starter. J79-GE-15 and -17 have combination cartridge/pneumatic starter on transfer gearbox.
CONTROL SYSTEM: Hydromechanical range-governing control system composed of two separate and distinct systems, the main fuel system and afterburner fuel system. Main system is controlled by main fuel control, which is a flow-controlling unit. The afterburner system is controlled by an independent control, also of the flow-controlling type. Automatic acceleration control with exhaust temperature limiting. Gear-type main fuel pump. Engine-driven centrifugal afterburner fuel pump.
FUEL SPECIFICATION: JP-4 or JP-5.
OIL SYSTEM: Dry-sump system. Vane-type pumps. Sump pressure provided from compressor. Oil cooling from fuel. Sump capacity ranges from 15 to 19 litres (4 to 5 US gallons; 3.3 to 4.2 Imp gallons). Average normal oil supply pressure 3.5 kg/cm^2 (50 lb/sq in).
OIL SPECIFICATION: MIL-L-7808, MIL-L-23699.
DIMENSIONS:

Length overall:	
J79-7A, 11A, J1K	5,283 mm (207.96 in)
J79-10, 17, 19, J1E	5,301 mm (208.69 in)
Max diameter:	
J79-7A, 11A, 15, J1K	973 mm (38.3 in)
J79-10, 17, 19, J1E	992 mm (39.06 in)

WEIGHT, DRY:

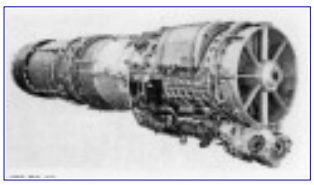
J79-7A	1,622 kg (3,575 lb)
J79-10	1,749 kg (3,855 lb)
J79-11A	1,615 kg (3,560 lb)
J79-15	1,672 kg (3,685 lb)
J79-19	1,740 kg (3,835 lb)
J79-J1E	1,699 kg (3,746 lb)
J79-J1K	1,685 kg (3,715 lb)

PERFORMANCE RATINGS (S/L, ISA):

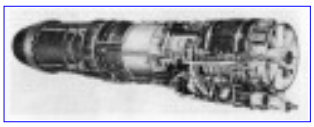
Max with afterburner:	
J79-7A, 11A	70.28 kN (15,800 lb st)
J79-15	75.62 kN (17,000 lb st)
J79-10, 17, 19	79.63 kN (17,900 lb st)
J79-J1E	83.40 kN (18,750 lb st)
J79-J1K	70.95 kN (15,950 lb st)
Dry (MIL):	
J79-7A, 11A	44.48 kN (10,000 lb st)
J79-15	48.49 kN (10,900 lb st)
J79-10, 17, 19	52.8 kN (11,870 lb st)
J79-1JE	49.4 kN (11,100 lb st)
J79-J1K	46.53 kN (10,460 lb st)

SPECIFIC FUEL CONSUMPTION:

T-O rating:	
J79-7A, 11A	55.8 mg/Ns (1.97 lb/h/lb st)
J79-15	54.97 mg/Ns (1.945 lb/h/lb st)
J79-10, 17, 19	55.53 mg/Ns (1.965 lb/h/lb st)
J79-J1K	58.5 mg/Ns (2.07 lb/h/lb st)
MIL rating:	
J79-15	24.3 mg/Ns (0.86 lb/h/lb st)
J79-7A, 10, 11A, 17, 19. J1K	23.74 mg/Ns (0.84 lb/h/lb st)



J79-GE-15 (1996)



J79-J1K licence-made by MTU (1996)

Longitudinal section through J79 (1996)

J79 combustor (1996)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

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GE AIRCRAFT ENGINES - GENERAL ELECTRIC

GENERAL ELECTRIC CJ805-23

In the early 1950s GE engineer Peter Kappus argued that Frank Whittle's aft-fan concept had much to commend it. This involves adding a completely independent free-running turbine behind an existing turbojet, with double-deck blades (GE called them `bluckets', from blade-buckets, or flades, from fan-blades). The inner part of each blade is rotated by the hot jet, carrying the outer part, the fan blade, with it. According to Kappus `The basic engine doesn't know what's going on behind it'.

The result was the aft-fan conversion of the J79 turbojet, initially designated X-220 and first tested on 26 December 1957. It entered production as the CJ805-23. No J79 engines were converted, but three types of commercial CJ805-23 were built:

CJ805-23

Prototypes, T-O rating 70.5 kN (15,850 lb st). Flight tested on RB-66 from 19 February 1960.

CJ805-23B

T-O rating 71.6 kN (16,100 lb st). Served in 37 CV-990 Coronados (first flight 24 January 1961), a few of which are still active.

CJ805-23C

Dash-23B modified to suit Caravelle demonstrator.
TYPE: Aft-fan turbofan.
FAN: Mechanically independent single-stage fan free-running in its own two bearings mounted on the aft end of a CJ805 (commercial version of the J79 turbojet). Mass flow 193.2 kg (426 lb)/s. BPR 1.54. Pressure ratio 1.65.
JETPIPE: The core exhausts through its own jetpipe in the centre of the concentric annular nozzle to the short fan duct.
DIMENSIONS:

Length with reverser	3,800 mm (149.6 in)
Length (basic engine)	3,325 mm (130.9 in)
Diameter	1,354 mm (53.3 in)

WEIGHT, DRY:

CJ805-23B with reverser	1,958 kg (4,316 lb)
CJ805-23B (basic engine)	1,708 kg (3,766 lb)

PERFORMANCE RATINGS: See model listing
SPECIFIC FUEL CONSUMPTION:

T-O, S/L	15.86 mg/Ns (0.56 lb/h/lb st)

CJ805-23B (1996)

CJ805-23B (1996)

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GE AIRCRAFT ENGINES - GENERAL ELECTRIC

GENERAL ELECTRIC CJ610

Announced in May 1960, the CJ610 is a power plant tailored for commercial and executive aircraft of 5,700 to 7,500 kg (12,500 to 16,500 lb) gross weight. It is essentially similar to the basic J85 turbojet, without afterburner.

By 1983, when production terminated, about 1,995 CJ610 engines had accumulated more than 10 million hours. TBO reached 4,000 hours in 1977.

There are seven versions:

CJ610-1, CJ610-4

Initial production versions, differing only in accessory gearbox locations.

CJ610-5, CJ610-6

Developed versions of -1 and -4 respectively, providing increased T-O thrust. Power Gates Learjet 24D, 25B and 25C, Hansa and IAI Westwind 1121.

CJ610-8, CJ610-9

Developed for production deliveries beginning in 1969. Power Hansa, IAI Westwind 1123 and NAL (Japan) experimental VTOL.

CJ610-8A

FAA certificated in April 1977 for operation at up to 15,500 m (51,000 ft), to give better economy and over-weather capability to Learjet Century III 24E, 24F, 25D and 25F. Main differences are longer life turbine and turbine-nozzle area change.
DIMENSIONS:

Length overall:	
CJ610-1, -5, -9	1,298 mm (51.1 in)
CJ610-4, -6, -8, -8A	1,153 mm (45.4 in)
Max flange diameter	449 mm (17.7 in)

WEIGHT, DRY:

CJ610-1	181 kg (399 lb)
CJ610-4	176 kg (389 lb)
CJ610-5	183 kg (402 lb)
CJ610-6	180 kg (396 lb)
CJ610-8, -8A	185 kg (407 lb)
CJ610-9	191 kg (421 lb)

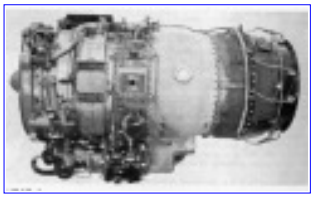
PERFORMANCE RATINGS (guaranteed):

T-O:	
CJ610-1, -4	12.7 kN (2,850 lb st)
CJ610-5, -6, -8A	13.1 kN (2,950 lb st)
CJ610-8, -9	13.8 kN (3,100 lb st)
Max continuous:	
CJ610-1, -4	12 kN (2,700 lb st)
CJ610-5, -6	12.4 kN (2,780 lb st)
CJ610-8, -9	13 kN (2,925 lb st)
CJ610-8A	12.7 kN (2,850 lb st)

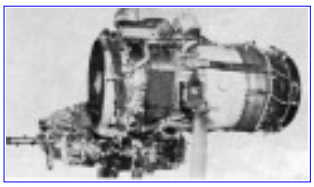
SPECIFIC FUEL CONSUMPTION:

At T-O rating:	
CJ610-1, -4	28.05 mg/Ns (0.99 lb/h/lb st)
CJ610-5, -6, -8, -9	27.75 mg/Ns (0.98 lb/h/lb st)

CJ610-8A	27.5 mg/Ns (0.97 lb/h/lb st)
At max continuous rating:	
CJ610-1, -4, -8A	27.5 mg/Ns (0.97 lb/h/lb st)
CJ610-5, -6, -8, -9	27.2 mg/Ns (0.96 lb/h/lb st)



CJ610-4 (1996)



CJ610-5 (1996)

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AERO-ENGINES - TURBOFAN, UNITED STATES OF AMERICA

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GE AIRCRAFT ENGINES - GENERAL ELECTRIC

GENERAL ELECTRIC CF700

Like the CJ610 turbojet, the CF700 is also derived from the J85 engine. Utilising the same gas generator, it is an aft-fan turbofan suitable for military and commercial aircraft. Since it can be tilted while in steady-state operation and operate vertically, it afforded lift/cruise capability in VTOL aircraft.

FAA certification of the original version was received on 1 July 1964. The uprated CF700-2D was certificated in early 1968. The CF700-2D has an improved compressor turbine with higher thermodynamic efficiency. The CF700-2D2 incorporates a new design of tailpipe. The main applications are the Dassault Falcon 20 and Rockwell Sabre 75.

By December 1974 the TBO had reached 3,000 hours. Production was completed in 1981 at just over 1,100.

The general description of the J85 turbojet applies also to the CF700, with the following additional assembly:

FAN: Single-stage free-floating fan. Bypass ratio 1.6:1. Mass flow 39.9 kg (88.0 lb)/s.

DIMENSIONS:

Overall length, compressor nose to tailcone tip	1,912 mm (75.57 in)
Diameter	913 mm (36.1 in)
Diameter (core engine)	449 mm (17.7 in)

WEIGHT, DRY:

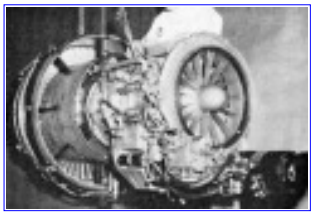
CF700-2C	329 kg (725 lb)
CF700-2D, -2D2	334 kg (737 lb)

PERFORMANCE RATINGS (S/L, ISA):

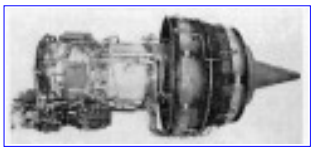
T-O:	
CF700-2C	18.68 kN (4,200 lb st) to 30ºC
CF700-2D, -2D2	18.91 kN (4,250 lb st)
Max continuous:	
CF700-2C	17.80 kN (4,000 lb st)
CF700-2D, -2D2	18.33 kN (4,120 lb st)

SPECIFIC FUEL CONSUMPTION (T-O, as above):

CF700-2C, -2D	18.46 mg/Ns (0.652 lb/h/lb st)
CF700-2D2	18.21 mg/Ns (0.643 lb/h/lb st)



CF700-2C (1996)



CF700-2D (1996)

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